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Reliability and Safety Assessment of Urban Air Mobility Concept Vehicles

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The Boeing Company
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EXECUTIVE SUMMARY

The following work was commissioned by the National Aeronautics and Space Administration (NASA) to guide industry and future regulation related to urban air mobility (UAM). Prior studies compared the relative safety of NASA concept vehicles, Figure ES1, designed for UAM and provided recommendations for industry research, aircraft architectural improvements, and regulatory updates. After the prior study completed, the European Aviation Safety Agency (EASA) released regulatory guidance in the form of a special condition (SC), SC-VTOL-01, for multirotors with distributed propulsion and flight controls (DPFC).

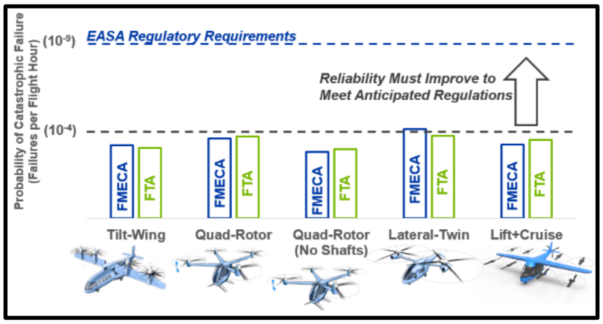

Figure ES1: Safety Level Results from Prior Work. Safety Must Improve to Comply with SC-VTOL-01

The objective of the current work was to develop DPFC architectures that will comply with SC-VTOL-01. Vehicle designs, DPFC architectures, and stability & control (S&C) models were developed to find limitations and trends to guide industry. To guide this task, NASA developed quad, hex, and an octorotor to better define vehicle attributes and trade space. Aircraft for study included electric, hybrid-electric, and turboshaft powerplants and collective and RPM control schemes. Assessments are in terms of the safety level achieved, and/or aircraft component/features needed to meet SC-VTOL-01. The most challenging criteria being the catastrophic failure rate, ≤10^{-9} catastrophic failures per flight hour, and that no single failures may result in a catastrophic event.

A disciplined process was followed, similar to that in Aerospace Recommended Practice (ARP) 4761. A preliminary system safety assessment (PSSA) leveraged prior work as a basis of creating failure rate budgets for system design teams. System designs were updated and iterated upon, working with reliability and safety subject matter experts to develop SC-VTOL-01 compliant designs. Design changes were reflected in updated PSSAs for initial verification of compliance.

The DPFC architecture was broken into four system design teams, the (1) flight control system (FCS), (2) drive and power system, (3) thermal management system (TMS), and (4) electrical power and distribution system. The FCS including elements necessary to control the aircraft, drive and power including elements necessary to generate and transmit shaft power, the TMS including elements necessary to maintain temperature limits in all operating environments, and electrical power and distribution including equipment necessary to store and transmit electrical energy.

Results found that all aircraft evaluated may have paths to comply with SC-VTOL-01, Figure ES2. However, S&C models showed large power transients that must be addressed and PSSA results show that future work is needed in single load path structures, high voltage power storage and distribution, and in motor/rotor overspeed protection.

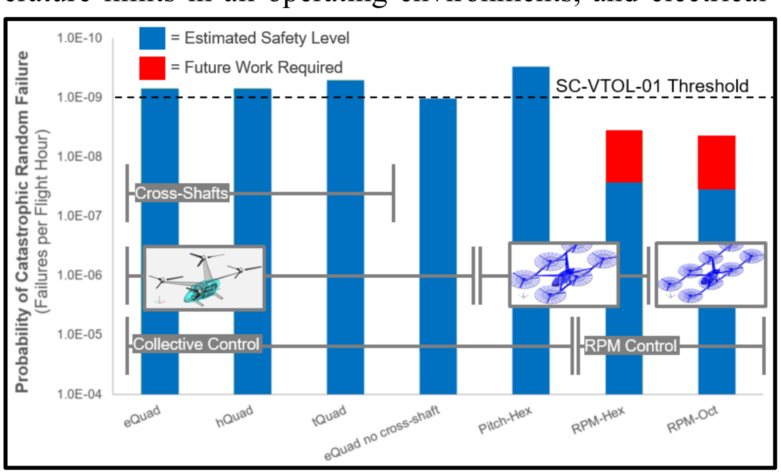

Figure ES2: Summary of Propulsion Safety Level Results per Vehicle

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LIST OF ACRONYMS

Acronym	Expansion
AAM	Advanced Air Mobility
AC	Advisory Circular
AC	Alternating Current
AFI	Arc Fault Interrupters
AGMA	American Gear Manufacturer's Association
AMC	Accepted Means of Compliance
ARMD	Aeronautics Research Mission Directorate
ARP	Aerospace Recommended Practice
CAD	Computer Aided Design
CB	Circuit Breaker
CMP	Cubic Mean Power
COTS	Commercial Off The Shelf
CP	Compensating Provision
CRP	Contingency Rated Power
CS	Certification Specification
DC	Direct Current
DE/HEP	Distributed Electric / Hybrid-Electric Propulsion
DPFC	Distributed Propulsion and Flight Control
DSR	Derived Safety Requirements
EASA	European Aviation Safety Agency
eHex	All-Electric Hexarotor
EHPS	Electric / Hybrid Propulsion Systems
EMA	Electromagnetic Actuator
EMF	Electromotive Force
EMP	Electric Motor Pump
eOct	All-Electric Octorotor
eQuad	All-Electric Quadrotor
ESC	Electronic Speed Controller
ETL	Effective Translational Lift
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Controllers
FAR	Federal Aviation Regulation
FBD	Functional Block Diagram
FCC	Flight Control Computer
FCS	Flight Control System
FDM	Failure Mode / Mechanical Distributions
FHA	Functional Hazard Analysis
FMECA	Failure Modes and Effects Criticality Analyses
FMI	Failure Mode Index

FOFS	Fail Operative, Fail Safe
FS	Fail-Safe
FSR	Flight Safety Report
FT	Fault Tree
FTA	Fault Tree Analysis
GFCI	Ground Fault Current Interrupter
HP	Horsepower
hQuad	Hybrid-Electric Quadrotor
HV	High Voltage
HVDC	High Voltage Direct Current
IGBT	Insulated-Gate Bipolar Transistor
IRP	Intermediate Rated Power
ISC	Internal Short Circuit
Li-ion	Lithium-Ion
LQR	Linear Quadratic Regulator
LVDC	Low Voltage Direct Current
MCP	Maximum Continuous Power
MOSFET	Metal–Oxide–Semiconductor Field-Effect Transistor
MRP	Maximum Rated Power
MTBO	Mean Time Before Overhaul
MTBR	Mean Time Before Repair
NASA	National Aeronautics and Space Administration
NDARC	NASA Design and Analysis of Rotorcraft
NPRD	Non-Electric Parts Database
OMI	One Motor Inoperative
OML	Outer Mold Line
PE	Power Electronics
PFH	Per Flight Hour
PMSM	Permanent Magnet Synchronous Motor
PRV	Pressure Regulating Valves
PSSA	Preliminary System Safety Assessment
PTC	Positive Temperature Coefficent
RPM	Rotations Per Minute
RVLT	Revolutionary Vertical Lift Technology
RWB	Reliability Workbench
S&C	Stability and Control
SAE	Society of Automotive Engineers
SDP	Shaft-driven Hydraulic Pump
SoC	State of Charge
SSPC	Solid State Power Controllers
TCV	Temperature Control Valve

TMS	Thermal Management System
tQuad	Turboshaft Quadrotor
TR	Thermal Runaway
UAM	Urban Air Mobility
VDC	Volt Direct Current
VTOL	Vertical Takeoff and Landing
WCA	Warning / Caution / Advisory

1 INTRODUCTION

The National Aeronautics and Space Administration (NASA) Revolutionary Vertical Lift Technology (RVLT) Project, under the Aeronautics Research Mission Directorate (ARMD), performs research in the area of integrated electric propulsion systems for Vertical Takeoff and Landing (VTOL) Urban Air Mobility and Advanced Air Mobility (AAM) vehicles. UAM, also referred to as urban air taxi operations, is a concept for flexible air transportation within a metropolitan area consisting of passenger-carrying operations and is the focus of this work. Vehicles for this mission will be capable of VTOL and desire zero-emissions operation and reduced noise compared to conventional helicopters. Zero-emissions operation can include all-electric power plants or turbine-based power plants combined with advanced fuel sources like hydrogen or net-neutral fuels.

To date, there are 400+ configurations being built/proposed for the UAM market (ref. 1). The NASA RVLT team has designed concept vehicles to identify crucial technologies, define research requirements, and explore a range of propulsion systems. The aircraft were designed using a NASA conceptual design and sizing tool for vertical lift, NASA Design and Analysis of Rotorcraft (NDARC) (ref. 2); the concept vehicles are described by Johnson and Silva, et al (ref. 3, 4). While the NASA-developed configurations in this statement of work may not be exact representations of every aircraft in this space, this work is intended to be generally applicable to all aircraft proposed for the UAM market and serve as a guide for developing the subject aircraft.

In a recently completed study funded by NASA and executed by Boeing (ref. 5), the failure modes and hazards associated with some RVLT concept vehicles were identified and functional hazard analyses (FHA), failure modes and effects criticality analyses (FMECA) were performed. A fault tree analysis (FTA) was created for each of the concept vehicles to compare the relative safety between vehicle configurations. In addition, conceptual designs of a notional powertrain configuration for each of the study vehicles were developed to support the reliability and safety analysis for the study. Hazards were identified and the severity of each were categorized in the FHA. Guidelines for reliability targets for both the air vehicle and the operation in the UAM mission were provided. The study provides a methodology for evaluating the safety and reliability of an architecture than can be applied to other UAM vehicles.

This statement of work expands on the previous work by refining it and developing more detailed propulsion system architectures. This work focuses on more similar vehicle configurations, permutating control scheme, number of rotors, and propulsion architecture. The prior work was the first step in a systems engineering approach to aircraft development, focused on the concept development phase. The process from prior work took system requirements and concept schematics to create an initial safety assessment and compared the relative safety of each vehicle against each other, prior to the development of bespoke safety standards for UAM. This work takes the process a step farther down the systems engineering "V" and delves into finer details of the aircraft and system design, in order to assess their impacts on reliability and safety against publically available regulatory guidance, SC-VTOL-01. Figure 1 shows the process used for this effort, which is similar to Society of Automotive Engineers (SAE) ARP 4761 (ref. 6), except that it is tailored for the specific application and means of compliance (ref. 7, 8).

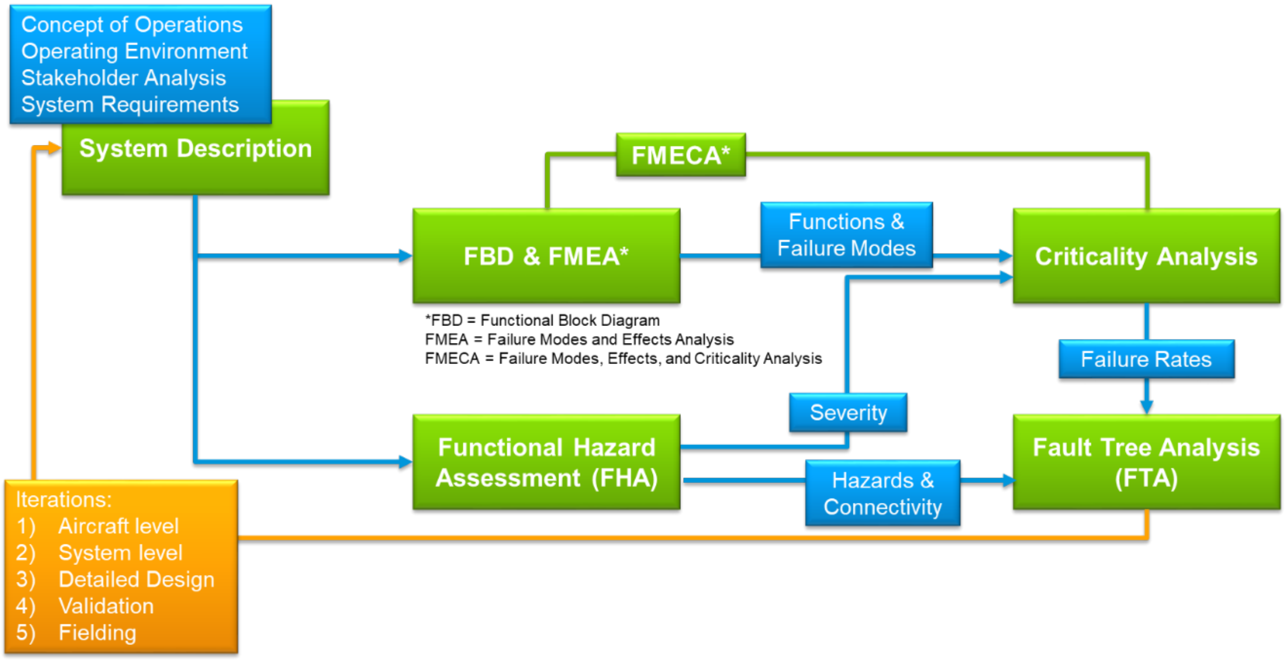


Figure 1: Process flow map followed in this statement of work

1.1 Research Objectives

The objective of this task was to contrast major design differences of UAM aircraft and explore the safety and reliability implications from the perspective of vehicle design, propulsion architecture, and flight dynamics & control. NASA developed concept vehicles to better define vehicle attributes and trade space and also framed a series of trade studies. Trade studies were separated by tables identified in the statement of work with specific questions to address. Appendix A includes the statement of work questions and responses, along with the specific sections of interest for each question.

Both quantitative and qualitative assessments of the safety of the major design attributes of the aircraft are shown in each table. The assessments are in terms of the safety level achieved, and/or aircraft components/features needed to comply with SC-VTOL-01. The two most challenging criteria being the catastrophic failure criteria, requiring $\leq 10^{-9}$ catastrophic failures per flight hour, and that no single failures may result in a catastrophic event.

NASA provided NDARC models and linearized stability and control derivative models of the bare-airframe dynamics for the configurations listed in Table 1. This work focused on comparing different vehicle design attributes: namely number of rotors, control scheme, and propulsion architecture. Additionally, architectural attributes were evaluated within each aircraft system in accordance with the statement of work.

Table 1: Concept vehicles configurations included

General Configuration	Number of Rotors	Number of Motors	Cross-shafting	Propulsion	Collective or RPM control?
Hexacopter	6	1 or more per rotor	No	Electric	RPM
Octocopter	8	1 or more per rotor	No	Electric	RPM
Hexacopter	6	1 or more per rotor	No	Electric	RPM
Hexacopter	6	1 or more per rotor	No	Electric	Collective
Quadrotor	4	1 or more per rotor	Yes	Electric	Collective
Quadrotor	4	Turbine, generator and 1 or more motors/rotor	Yes	Hybrid Electric	Collective
Quadrotor	4	1 turboshaft engine	Yes	Turboshaft	Collective

1.2 Technical Approach

The technical approach consists of defining requirements, designing the conceptual DPFC architecture, and assessing the reliability and safety of each aircraft under consideration. Prior work (ref. 5) was used as a basis to begin a preliminary system safety assessment (PSSA), Sections 6 and 7; initial fault tree architectures and failure rates were also incorporated into the PSSA to guide DPFC architecture development. In parallel, transient S&C models were developed, Section 5. The statement of work, PSSA, and S&C models were used to derive requirements that were fed back into the DPFC architecture, Sections 8 through 11. The updated DPFC architecture was then reevaluated against derived requirements as initial verification of compliance to SC-VTOL-01 safety targets, Sections 12 and 13.

The distributed propulsion and distributed flight control systems are the primary, novel features of the NASA RVLT concept vehicles. Functional systems that are similar to the state-of-the-art, such as rotors, blades, pilot and crew interfaces, and landing gear, were not included in this study because of the availability of guidance needed to safely develop these systems.

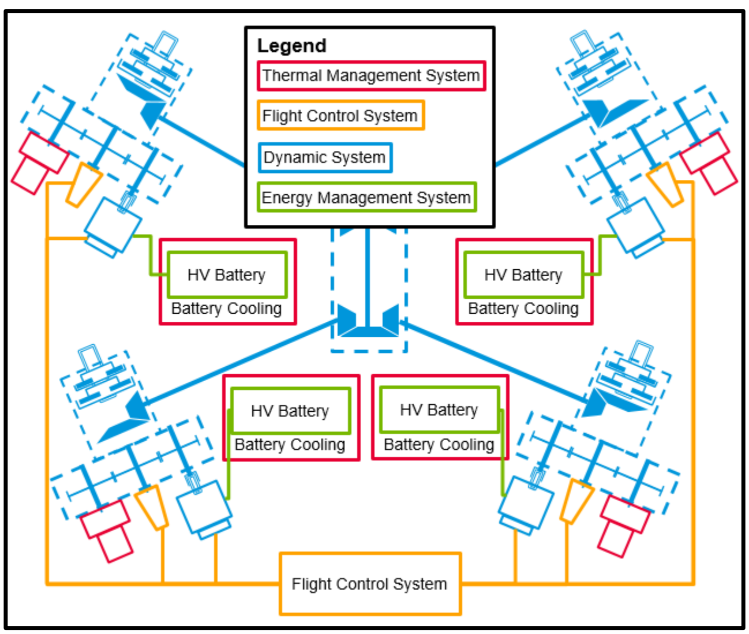


Figure 2: DPFC architecture Elements

Helicopter certification specifications and supporting documentation were used for an initial basis of comparison when needed because of the NASA RVLT concept vehicles VTOL capability. Other certification specifications for other aircraft types, such as fixed-wing aircraft, should be reviewed in the future for their applicability to VTOL aircraft and compared against SC-VTOL-01 and other VTOL certification specifications.

Following EASA guidance under Certification Specification (CS) CS-27, the "power plant" includes all elements of the rotorcraft that affect the safety of the propulsive units. As shown in Figure 2, the multi-rotor DPFC architecture analysis herein includes elements of the (1) flight control system (FCS), (2) drive and power system, (3) thermal management system (TMS), and (4) electrical power and distribution system.

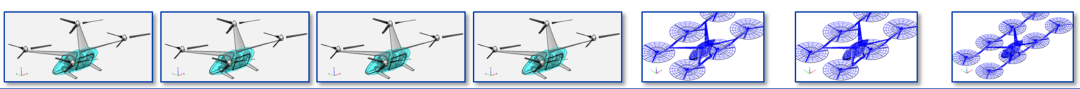
The unique attributes of the NASA RVLT quadrotor DPFC architecture mean that obtaining a type certificate under CS-27 would be difficult, at best. EASA has recognized this and has published SC-VTOL-01 which establishes certification criteria for vertical takeoff and landing vehicles with unique propulsion and control architectures, including distributing the flight controls and electric propulsion elements.

SC-VTOL-01 attempts to increase the safety metrics from CS-27, and even CS-29. Some of the CS-27 or CS-29 regulations requiring fail safety or redundancy include exceptions for specific cases. For example, Advisory Circular (AC) 29-2C allows for dual independent hydraulic system for power boosted control systems; which will not meet the overarching CS-29 10^{-9} catastrophic failures per flight hour requirement.

1.3 Vehicle Configurations

Three primary trade studies were conducted during this statement of work. The propulsion architecture trade study compared three different quadrotor vehicles: an all-electric quadrotor (eQuad), a hybrid quadrotor (hQuad), and a turboshaft quadrotor (tQuad). Control schemes were compared in the second trade study with two all-electric hexarotors (eHex), one with variable speed and fixed pitch control and the other with control blade pitch and fixed speed control. The final trade study compared the number of rotors, comparing an eHex and all-electric octorotor (eOct), both with a variable speed control. Table 2 details the specific information for each aircraft.

Table 2: NASA RVLT Concept Vehicle Configurations



	eQuad	hQuad	tQuad	eQuad	eHex	eHex RPM	eOct
Propulsion Type	All-Electric	Series Hybrid-Electric	Twin Turboshaft	All-Electric	All-Electric	All-Electric	All-Electric
Control Scheme	Collective Blade Pitch	Collective Blade Pitch	Collective Blade Pitch	Collective Blade Pitch	Collective Blade Pitch	Rotor Speed	Rotor Speed
Mech. Interconnect	Yes	Yes	Yes	No	No	No	No
Number of Rotors	4	4	4	4	6	6	8
Design Gross Weight	6,469 lb	5,115 lb	3,735 lb	(a)	6,656 lb	6,212 lb	6,847 lb
Rotor Diameter	26.2 ft	21.6 ft	18.4 ft	(a)	21.7 ft	20.9 ft	19.0 ft
Nominal Rotor Speed	401 rpm	487 rpm	570 rpm	(a)	484 rpm	501 rpm	551 rpm
Battery Capacity	368.1 kWh	34.7 kWh	-	(a)	413.1 kWh	386.3 kWh	454.6 kWh
Fuel Capacity	-	245 lb	179 lb	-	-	-	-

Note: (a) Sizing data was assumed to be similar to the eQuad with mechanical interconnection for this study.

The all-electric propulsion system studied stores energy in a battery network and providing power to the rotors via electric motors. The hybrid-electric system stores energy in the form of liquid or gaseous fuel, converted to electrical energy via a turbogenerator, and provides power to the rotors via electric motors. The turboshaft system stores power in liquid or gaseous fuel and provides power to the rotor through turboshaft engines. The vehicles compared in the propulsion type trade study were all variations of a quadrotor with collective blade pitch control. This study also includes an evaluation of replacing the eQuad mechanical interconnection (cross-shafting) with component level redundancy, such as multi-winding motors. Table 2 includes the details for each vehicle. Quadrotor comparisons aimed to address the impact of propulsion architectural decisions on vehicle safety and reliability. Due to the different propulsion systems, system functions, failure modes, and severity were addressed to draw fair comparison of safety and reliability.

The control scheme trade study compares the impact of rotor thrust control on vehicle safety and reliability. The difference in control scheme affects the per-flight hour failure rate of the power system components and the overall safety and reliability of the vehicle. In addition, the peak transient power levels are identified along with their impact to component reliability.

The final trade study compares the number of rotors on the vehicle safety and reliability on the aircraft and component design is studied along with its impact of per-flight hour failure rate of power system components. This research was performed by comparing the RVLT eHex and eOct, both of which utilize rotor speed control. The peak transient power levels required per propulsion unit were identified. Component redundancy of both the hexarotor and octorotor was evaluated with respect to overall system reliability.

2 WORK SCOPE

This work builds upon Boeing's prior work with NASA (ref. 5). The work herein focuses on a six passenger quadrotor RVLT concept vehicle, Figure 3, (ref. 4) and derivations thereof. PSSA and S&C simulations were performed to define requirements that were used by system conceptual design teams. Conceptual designs developed DPFC concept architectures against EASA special condition guidance. This study attempted to develop SC-VTOL-01 compliant DPFC architectures for each of the NASA RVLT concept vehicles under study, in the UAM mission.

Four main engineering capability teams were responsible for the development and analysis of the DPFC architecture, (1) flight control, (2) drive and power, (3) thermal management, and (4) electrical power and distribution. Conceptual designs from prior work focused on elements of each capability area that are novel for the vehicles under study. Conceptual designs from prior work (ref. 5) were used to update PSSA elements in order to comply with SC-VTOL-01. Functional block diagrams were created that show the connectivity between primary functions. FHAs were performed using the functions and connectivity from the functional block diagrams. Historical data was used to assess the reliability of groups of components, where available. Sub-systems or groups of components that do not have historical data available were configured with enough detail in an attempt to accurately assess their reliability. The reliability of each component or sub-system was then plugged into the PSSA for initial verification of compliance with SC-VTOL-01.

This process was carried out for NASA RVLT concept vehicles (refer to Table 1 and Table 2):

1. Electric quadrotor (eQuad) with variable pitch, collective control;
2. Hybrid-electric quadrotor (hQuad) with variable pitch, collective control;
3. Turboshaft quadrotor (tQuad) with variable pitch, collective control;
4. eQuad without interconnecting shafts with variable pitch, collective control;
5. Electric hexarotor with variable pitch, collective control (Pitch eHex);
6. eHex with variable speed control (RPM eHex);
7. Electric octorotor (eOct) with variable speed control.

Using the RVLT concept vehicles under consideration, each engineering capability team developed comparisons, limitations, and trends in order to guide industry research, aid smaller, startup companies on the development of safe eVTOL, and potentially guide future regulation updates related to UAM.

3 BACKGROUND

NASA research has pioneered distributed electric/hybrid-electric propulsion (DE/HEP) through concepts like GL-10 and X-57. NASA has also previously worked with transportation companies, such as Uber, to develop gap assessments for UAM. UAM is envisioned as networks of small VTOL DE/HEP aircraft. These aircraft differ from conventional fixed-wing aircraft because of their VTOL runway independence and differ from rotary-wing airplanes because they typically use multiple rotors for vertical lift. Specifically, DE/HEP systems have grown out of a surge in remote control multi-rotor helicopters and are believed to make air travel safer because of multiple lift/thrust units and the potential for added redundancy and segregation by design (ref. 8). Designing a fail-safe system is an accepted practice to improve safety in any number of design cases; however, due to the complex flight dynamics associated with VTOL aircraft, adding additional propulsion systems and/or rotor systems does not necessarily make the system or aircraft fail-safe.

Safety and reliability per vehicle are paramount for the UAM mission as these aircraft will be carrying people, operating over densely populated areas, and are expected to operate 40 hours per week, year round (ref. 9). These factors require higher safety standards than traditional, small rotorcraft to mitigate social and economic losses. The number of aircraft required to build a network to fulfill the UAM mission requires even higher safety standards per vehicle to keep injuries, fatalities, and economic losses per year extremely low. Boeing's prior work (ref. 5) has shown how incorporating redundant propulsion and rotor systems may not result in a vehicle that meets Federal Aviation Administration (FAA) or European Aviation Safety Agency (EASA) regulatory guidance. Results found that proposed propulsion system architectures resulted in top-level, vehicle catastrophic failure rates of approximately 10^{-4} failures per flight hour, see Figure 3.

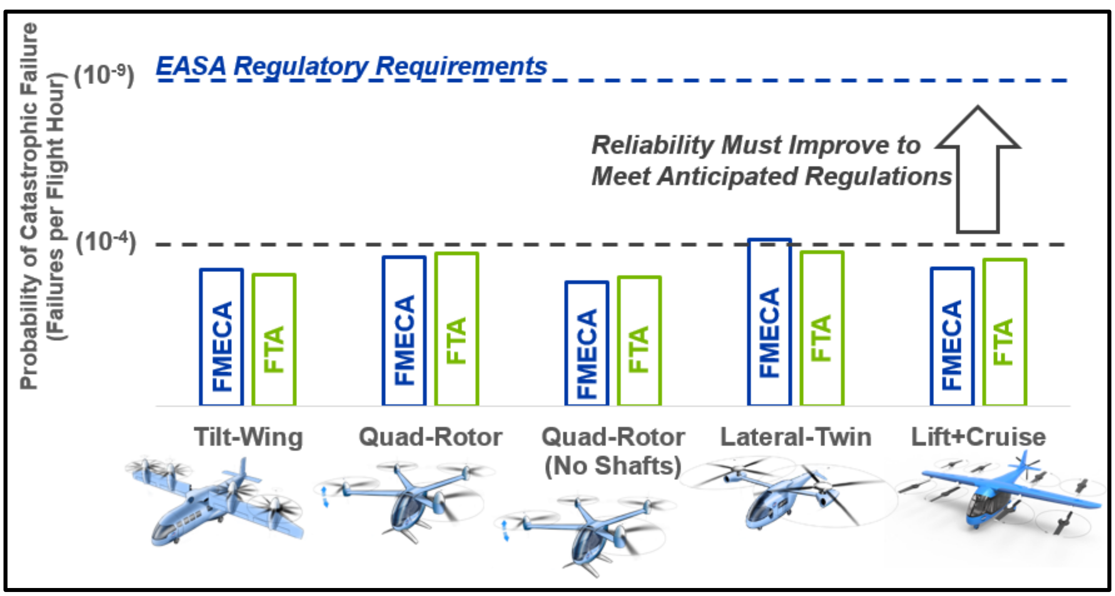


Figure 3: Top-Level Safety Results from Boeing Study (ref. 5)

In July, 2019, EASA issued SC-VTOL-01, focused on these emerging aircraft for the UAM mission. SC-VTOL-01 is the first regulatory document released that is intended to specifically govern the emerging multi-rotor, DPFC aircraft (ref. 8). It is not intended to govern aircraft with more conventional propulsion and flight control architectures which have less than three rotors to create lift in hover or low-speed flight (ref. 8). It includes vehicle reliability requirements based on vehicle size, number of people onboard, personal or commercial use, and geographic location. However, any UAM vehicle carrying passengers for commercial use falls into the most stringent category, Category Enhanced.

SC-VTOL-01 requires of Category Enhanced vehicles that "a single failure must not have a catastrophic effect upon the aircraft," but establishes that a future accepted means of compliance (AMC) "will include considerations on what constitutes single failures in the context of single and multiple loads paths" (ref. 10). SC-VTOL-01 also establishes that catastrophic failures of equipment, systems, and installations are "extremely improbable and does not result from a single failure," and establishes a threshold for Category Enhanced vehicles of 10^{-9} catastrophic failures per flight hour for equipment, systems, and installations (ref. 10).

Additionally, on January 27, 2020, EASA proposed SC E-19 (ref. 11) which was developed to support type certification of electric and/or hybrid propulsion systems (EHPS). EASA SC E-19 requires a safety analysis to be performed that accounts for all failure conditions that are reasonably expected to occur. The safety analysis should consider secondary failures, latent failures, and, "multiple failures that result in hazardous EHPS effects, hazardous aircraft effects, or catastrophic aircraft effects," (ref. 11). Additionally, a summary of the safety analysis must be provided, including the probability of occurrence of failure conditions that result in major aircraft effects, hazardous EHPS effects, hazardous aircraft effects, or catastrophic aircraft effects.

If a propulsion system for some aircraft configurations is all-electric and/or utilizes a variable speed control scheme, a designer will need to consider the safety effect of a propulsion thruster failure within the guidance of the functional hazard analysis. Such consideration should include the assignment of the motor as a key element of flight control, making the loss of a propulsor tightly coupled to loss of control.

DPFC architectures may see a large functional overlap between systems. In one scenario, control is provided by a variable pitch rotor system creating a light functional coupling to the propulsion system; in another example, control is provided by varying the speed of the rotor system, which tends to create a tight functional coupling between airplane control and propulsion. Figure 4 uses some example airplane functions to depict the overlap of typical variable pitch and variable speed DPFC architectures.

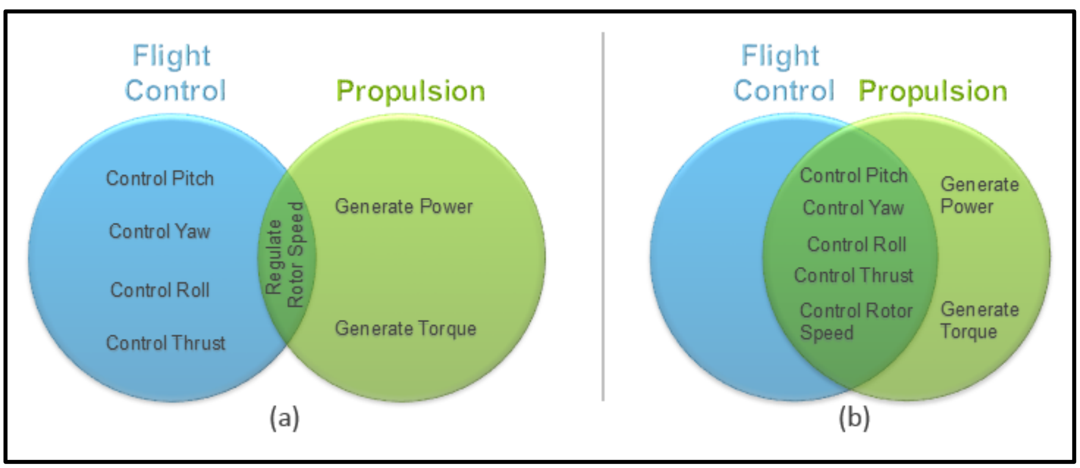


Figure 4: Flight Control and Propulsion Systems' Functional Overlap for (a) Typical Variable Pitch DE/HEP Systems, and (b) Typical Variable Speed DE/HEP Systems

Because of the tight coupling between flight control and propulsion in DPFC architectures, the aircraft special condition, SC-VTOL-01, and the propulsion system special condition, SC E-19, both must be considered. Proposed SC E-19 applies to the propulsions system, specifically, but does not include rotors or other aircraft attributes, such as vehicle control. SC-VTOL-01 applies to rotors, vehicle control, and some elements of the propulsion system pertinent to continued safe flight and landing for VTOL aircraft. The study herein is still largely focused at system level attributes, and therefore focuses on compliance with SC-VTOL-01, but compliance with SC E-19 was also considered and will have a larger role in future, more detailed system development.

4 DESIGN MISSION AND VEHICLE POWER SPECTRUMS

Design mission and vehicle power spectrums were created for each vehicle configuration provided in Table 2. Design mission and vehicle power spectrums lay the foundation for subsystems analysis, namely the electrical power and distribution system, flight control system, drive and power system, and thermal management system. The UAM mission by Silva, et al was used as the design mission for this study, Figure 5 (ref. 4). The power spectrums are based on sizing and performance data provided by NASA in the form of NDARC model output. The power spectrums show step changes in power between mission segments for simplicity. Operationally, there would be smooth transitions in power between the mission segments.

Segment	1	2	3	4	5	6	7	8	9	10
Initial Alt. (MSL ISA)	6,000	6,000	6,050	6,050	10,000	6,050	6,050	6,050	6,000	10,000
Final Alt. (MSL ISA)	6,000	6,050	6,050	10,000	10,000	6,050	6,050	6,000	6,000	10,000
Time (sec)	15	30	10	t_{climb}	t_{cruise}	10	30	30	15	1200
Distance (nmi)	-	0	0	D_{climb}	37.5-D_{climb}	0	0	0	0	-
Speed	-	-	0	V_y	V_{br}	0	0	-	-	V_{br}
ROC (ft/min)	-	100	0	≥ 900	0	0	0	-100	-	0
Percent of Max Power	10%	100%	100%	P_{climb}	P_{cruise}	100%	100%	100%	10%	P_{cruise}

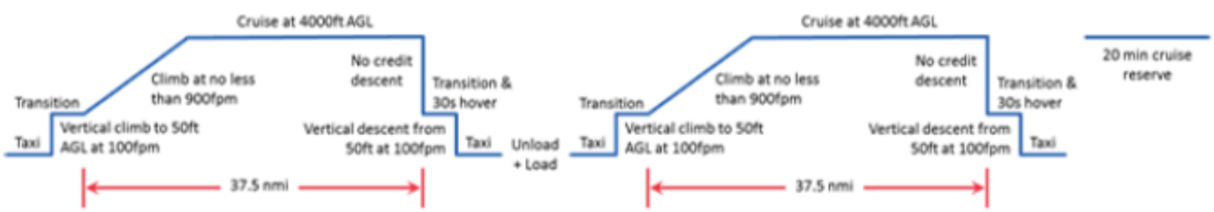


Figure 5: Design Mission

Figure 6 shows the power required per rotor vs. mission type for three quadrotors, each with different propulsion architectures. The power requirements for the turboshaft quadrotor are significantly lower than the electric and hybrid quadrotors, driven by a lighter weight propulsion system and sized gross weight of the vehicle. The all-electric quadrotor cruises at a lower speed compared to the turboshaft and hybrid-electric vehicles, which impacts the economic value of the mission.

Figure 7 shows the power required per rotor vs. mission type for three all-electric vehicles: the quadrotor, the hexarotor, and the octorotor. As expected, the power required per rotor is lower for the vehicles with higher numbers of rotors. However, the overall vehicle size, weight and total power and energy requirements are increasing with number of rotors as shown previously in Table 2, driven by the increasing complexity of the system.

The power spectrums were used to define derived requirements used for drive and power system sizing, thermal management system cooling analysis, and a comparison of component usage spectrums. Future work would develop usage spectrums, incorporating maneuver loads and various gross-weights to better characterize mission parameters for system definition.

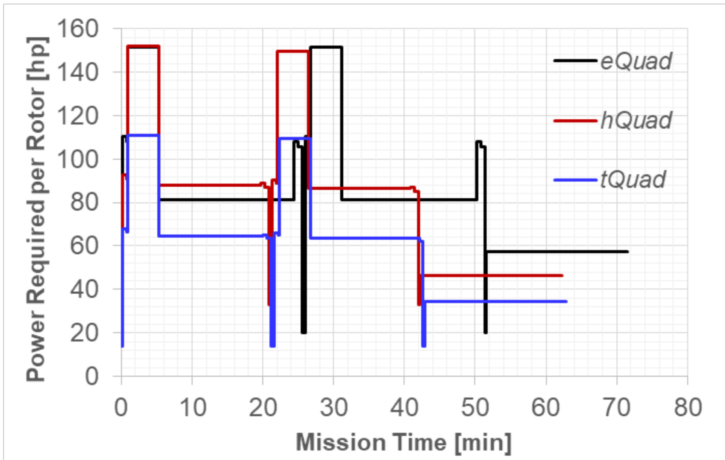


Figure 6: Vehicle power spectrum of the all-electric quadrotor, hybrid-electric quadrotor, and the turboshaft quadrotors

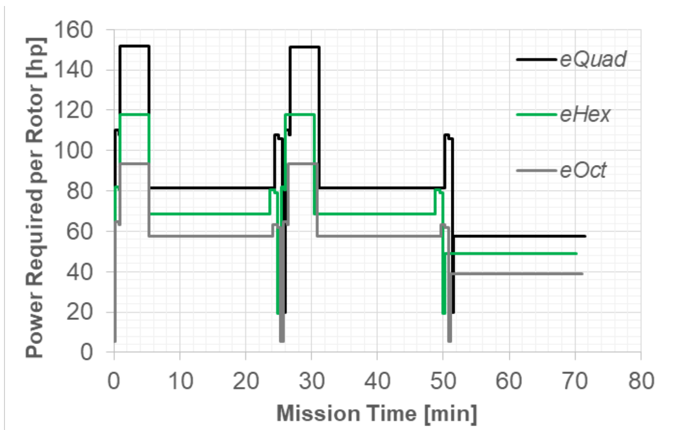


Figure 7: Vehicle power spectrum of three all-electric vehicles: a quadrotor, hexarotor, and octorotor vehicles

5 STABILITY AND CONTROL SIMULATION

In order to assess vehicle safety, operation in nominal and failure conditions must be considered in all operating environments. To help characterize safety in all operating environments, transient power requirements are investigated in the presence of atmospheric disturbances representative of an urban canyon environment. The primary focus of this investigation is the sensitivity of transient power requirements with respect to vehicle control scheme and number of rotors. Performance in atmospheric disturbances is particularly important to urban air mobility vehicles due to the highly turbulent environment in which they are intended to operate and because vehicle size may make them particularly sensitive to small-scale winds.

Simulation analyses are presented in order to assess transient power requirements for the quadrotor, hexarotor, and octorotor vehicle configurations for disturbance rejection in an urban canyon environment. A representative atmospheric disturbance framework is developed, and significant fidelity is added to the government-furnished bare airframe models in order to analyze transient power requirements in nominal and pertinent failure conditions. This disturbance rejection analysis additionally serves to show margin around the steady state (i.e. trim condition) that would be required for maneuvering in this environment. The analysis in this section is primarily conducted using MATLAB and Simulink.

The simulation framework and processes are first applied to the analysis of the all-electric quadrotor with and without cross-shafting. Additional fidelity is then added to the simulation models for the analysis of the hexarotor and octorotor vehicle configurations. Nominal and failure conditions are simulated for all configurations in order to assess the impact of drive system configuration, thrust control scheme, and number of rotors on transient power requirements.

5.1 Model Description and Analysis Approach

5.1.1 Atmospheric Disturbance Framework

The atmospheric disturbance framework used to evaluate vehicle design attributes consists broadly of three classes of atmospheric motion: steady winds, discrete gusts, and continuous turbulence. A survey of existing regulatory guidance for rotorcraft and fixed-wing aircraft informs parameters of these atmospheric motion profiles. Gust and turbulence model parameters are further tailored based on a survey of existing literature of atmospheric flow in urban environments. The transient power required for disturbance rejection is evaluated using this analysis framework in dynamic simulation models for several trim conditions in hover/low speed and forward flight.

All vehicle configurations are assumed to have on-board pilots and therefore disturbance rejection criteria are established using publicly available documents associated with piloted vehicles, some of which have a history of operating in urban canyon-type environments. FAA AC 25.341-1, which establishes means of compliance for Federal Aviation Regulation (FAR) Part 25 Gust and Turbulence Loads, informs discrete gust and continuous turbulence profiles used in this analysis (ref. 12, 13). These profiles, as well as steady wind profiles, are tailored based on relevant guidance in FAA AC 27-1B and 29-2C, which establish means of compliance with FAR Part 27 and 29, respectively (ref. 14, 15, 16, 17). These profiles are further tailored by existing military guidance including MIL-HDBK-1797 and MIL-F-8785C, and EASA Proposed Means of Compliance with Special Condition for small-category Vertical Take-Off and Landing (VTOL) aircraft (SC-VTOL-01) (ref. 18, 19, 7). Turbulence model parameters are tailored in accordance with low-altitude wind models developed by Boeing for the FAA, which have been used by the FAA, Boeing

Commercial Aircraft, and Boeing Vertical Lift for evaluation of turbulence profiles near ground (ref. 20).

5.1.1.1 Steady Winds

Steady wind conditions used in the analysis of each configuration are primarily defined based on guidance related to controllability and maneuverability in FAA AC 27-1B, and 29-2C, as well as the atmospheric disturbances related to minimum acceptable handling qualities ratings in the EASA Proposed MOC with SC-VTOL-01 (ref. 7). In accordance with this guidance, 29 fps (17 knot) linear gust velocities are injected into the bare airframe models at azimuths from 0 to 360 degrees in increments of 45 degrees. The steady wind magnitude ramps linearly from zero to maximum velocity in order to avoid incurring transient power spikes from step-like winds.

The atmospheric disturbance framework is refined for evaluation of the hexarotor and octorotor vehicle configurations, for which steady winds with 50 fps magnitude are also injected. This value is commensurate with the mean wind speed near ground implied by continuous turbulence used in this analysis, described in a subsequent subsection. In addition, this extended steady wind envelope is intended to be representative of UAM operational scenarios in which added pressure to maintain schedule in off-nominal conditions may be expected.

5.1.1.2 Discrete Gusts

Discrete gust profiles injected for the purposes of disturbance rejection analysis follow a one-minus-cosine velocity profile in accordance with guidance from AC 25.341-1. In the context of an urban canyon environment, these discrete gust profiles are intended to capture wind velocity profiles that could be expected to result from airflow jetting through gaps between buildings as well as vortices within street canyons (ref. 21).

Discrete gusts with a range of gust widths from 26.2 to 327.5 feet are evaluated, based on the rotor diameter of the all-electric quadrotor. The maximum amplitude of these gust profiles is 50 fps, in accordance with guidance from the EASA Proposed MOC with SC-VTOL-01 for gust conditions in VTOL mode (ref. 7). The linear velocities associated with these discrete gust profiles are input to the bare airframe models for each vehicle configuration in the ±x, y, and z directions in the North-East-Down coordinate frame.

These discrete gust profiles are defined as a function of gust penetration distance. Thus for the purposes of assessing comparable atmospheric motion profiles in the hover trim condition, a steady velocity is applied to the gust field, analogous to injection of the gust as a function of penetration distance in forward flight.

5.1.1.3 Continuous Turbulence

Continuous turbulence, also known as stochastic or random turbulence, comprises the third class of atmospheric disturbance injected for the purposes of disturbance rejection analysis. The Dryden wind turbulence model has been deemed acceptable for turbulence evaluation in the context of assessing flying qualities of piloted aircraft and therefore is used for the purposes of this analysis (ref. 18, 22). Furthermore, the discrete Dryden wind turbulence model is available as a block in Simulink that is convenient for integration into the discrete-time simulation setup used to conduct this analysis.

A limited number of light, moderate, and severe turbulence cases are evaluated for the all-electric quadrotor. The altitudes and airspeeds used in this evaluation are based on the sizing mission profile and summarized in Table 3.

Table 3: Altitude and airspeed combinations used for evaluation of continuous turbulence for the all-electric quadrotor.

Trim Condition	50 ft. AGL	1000 ft. AGL	2000 ft. AGL	4000 ft. AGL
Hover	Light, moderate, severe	-	-	-
50 KTAS	-	Light, moderate, severe	-	-
110 KTAS	-	-	Light, moderate, severe	Light, moderate, severe

The primary focus of the disturbance rejection analysis is to evaluate the sensitivity of transient power requirements with respect to vehicle control scheme and number of rotors. Therefore, the atmospheric disturbance framework is refined for evaluation of the hexarotor and octorotor vehicle configurations, for which turbulence intensities are tailored for a range of surface roughness lengths. Surface roughness lengths from 0.15 to 15 feet are evaluated in order to capture turbulent conditions representative of urban city centers as well as flat terrain appropriate to most airports (ref. 20, 22, 23). The altitudes chosen for continuous turbulence evaluation are based on the sizing mission profile, ranging from 50 to 4000 feet AGL. Turbulence intensity is tailored based on the range of surface roughness lengths for altitudes below 2000 feet (i.e. the applicable low-altitude regime). Unrealistic low-altitude, high-speed flight conditions (i.e. 100 KTAS, under 100 ft. AGL) are excluded for the purposes of this analysis.

The Dryden turbulence model treats linear and angular velocity components as spatially varying stochastic processes. In order to inject a pseudo-random turbulence stream conforming to Dryden velocity spectra in the hover trim condition, a steady velocity is applied to the turbulence field. In the hover trim condition, then, it is appropriate to consider the aircraft hovering in a steady wind with the turbulence superimposed, analogous to the turbulence evaluation in forward flight.

5.1.2 Vehicle Model Setup

The simulation models for all vehicle configurations share a common set of components. These components include the wind model described in the previous section as well as the vehicle controller, engine controllers, and government-furnished bare airframe models. This section details the simulation model as tailored for each vehicle configuration.

5.1.2.1 Vehicle Controller Setup

The vehicle controller for each configuration is a linear quadratic regulator (LQR) tuned in order to meet specific disturbance rejection criteria in the presence of a discrete gust profile representative of the range of discrete gusts described previously. The discrete gust profile to which the controller is tuned in each axis has a 50 fps maximum gust velocity and 157 foot gust width.

This simple controller serves to provide a baseline for comparison of transient power requirements for each vehicle configuration. In hover, this controller implements rate, attitude, velocity, and position feedback and in forward flight implements rate, attitude, and velocity feedback. The disturbance rejection metrics to which the controller is tuned are based on vehicle size and desired ADS-33 criteria for hover and speed control mission task elements, detailed in Table 4 (ref. 24). The vehicle controller is tuned to stay just within the desired performance metrics in the presence of the representative discrete gust profile so as to meet the criteria without exceeding realistic control ranges or overly burdening the power system.

Table 4: Disturbance rejection metrics for tuning vehicle controller 3

Performance Metric	Desired Value (Hover)	Desired Value (Forward Flight)
Lateral Position	±3 feet	N/A
Longitudinal Position	±3 feet	N/A
Altitude	±2 feet	±100 feet
Airspeed	N/A	±2.5 knots
Heading	±5 degrees	±5 degrees

5.1.2.2 All-Electric Quadrotor

A schematic of the model setup for the all-electric quadrotor vehicle configuration is presented in Figure 8. The aircraft control system is a LQR tuned to an acceptable output state error band described in Section 5.1.2.1.

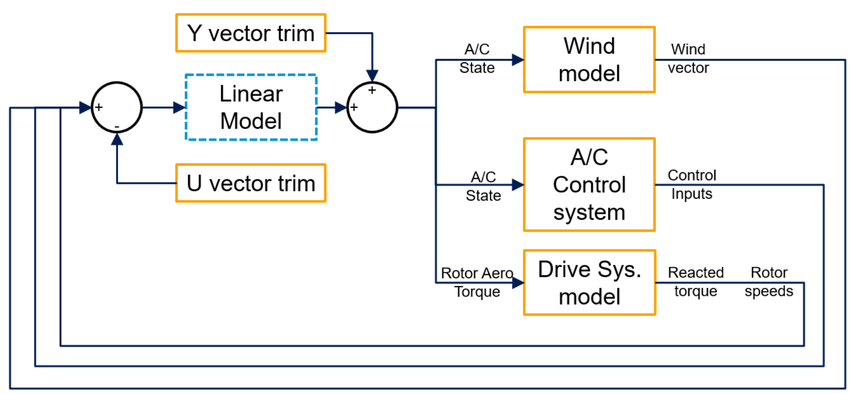


Figure 8: All-electric quadrotor model schematic. Government-furnished components are outlined in blue dashed lines and Boeing-developed components in yellow.

The linear model represents the government-furnished bare airframe model trimmed in a particular hover or forward flight condition. The states and output signals have been rearranged in order to facilitate the drive system model described below, and to add position states to facilitate disturbance rejection analysis.

The rigid drive system models individual cross-shaft load conditions based on rotor aerodynamic torque computed from rotor accelerations from the bare airframe model and torque outputs of individual engine models. The engine models maintain trim rotor speed using proportional-integral control tuned to a prescribed operational bandwidth of approximately 3 Hz. The 3 Hz bandwidth is representative of typical helicopter power plant response. Resulting reacted torques and rotor speeds are fed back as inputs to the bare airframe model. The drive system is set up to facilitate individual zero-torque engine and cross-shaft failures.

The wind model produces a wind vector consisting of body-axis rates and velocities as applicable based on the atmospheric disturbance framework described in the previous section. Steady winds, discrete gusts, and continuous turbulence are generated and applied to the bare airframe model separately.

5.1.2.3 Collective-Controlled Hexarotor

A schematic of the model setup for the collective-controlled hexarotor vehicle configuration is presented in Figure 9.

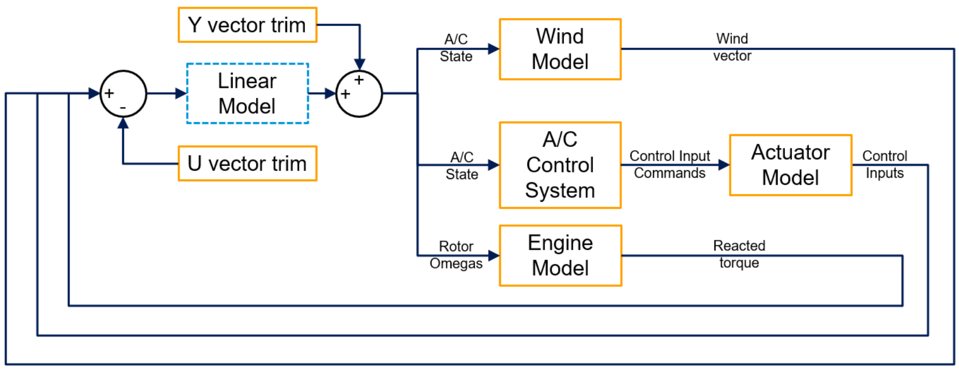


Figure 9: Collective-controlled hexarotor model schematic. Government-furnished components are outlined in blue dashed lines and Boeing-developed components in yellow.

Similar to the all-electric quadrotor, the linear model represents the government-furnished bare airframe model trimmed in a particular hover or forward flight condition with added position states to facilitate disturbance rejection analysis. The wind model, aircraft control system, and engine models are consistent with those described for the all-electric quadrotor, without implementation of a cross-shafted drive system.

The collective-controlled hexarotor model differs from the all-electric quadrotor in that collective actuator models provide added fidelity for the purposes of assessing transient power requirements, particularly in the presence of actuator failures. Individual rotor collective pitch commands are computed from the control input commands from the vehicle controller. The individual actuators are modeled as transfer functions with a prescribed operational bandwidth of 7 Hz. The output collective pitch values are used to compute actual control inputs to the bare airframe model. The actuator models are set up to facilitate individual failure capabilities in a fail-fixed condition.

5.1.2.4 RPM-Controlled Hexarotor and Octorotor

A schematic of the model setup for the RPM-controlled hexarotor and octorotor vehicle configurations is presented in Figure 10. The difference in the model setup between these configurations is simply the number of engines and associated inputs, states, and outputs within the linear model.

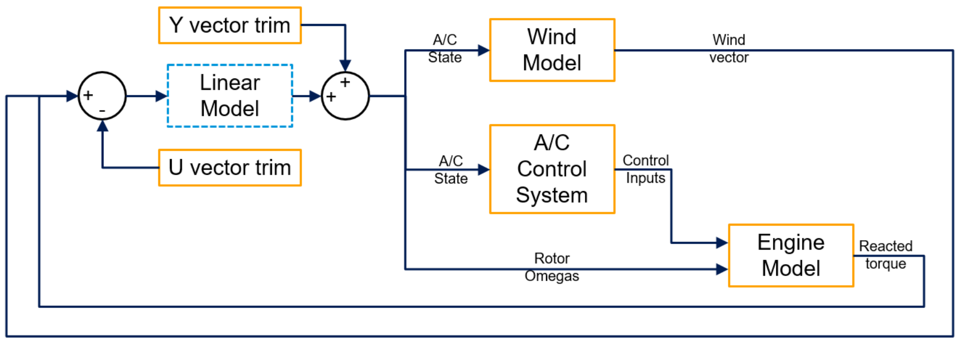


Figure 10: RPM-controlled hexarotor and octorotor model schematic. Government-furnished components are outlined in blue dashed lines and Boeing-developed components in yellow.

Similar to the previous vehicle configurations, the linear model represents the government-furnished bare airframe model trimmed in a given hover or forward flight condition and position states have been added to facilitate disturbance rejection analysis. The wind model and aircraft control system are consistent with those described for the all-electric quadrotor.

3 and 7.25 Hz bandwidths were evaluated for use with the RPM-controlled hexarotor. Results showed a trend of lower power transients when a lower (3 Hz) engine bandwidth was used, but the difference between maximum power transients for each bandwidth considered was relatively small. The engine controllers implemented in the RPM-controlled vehicle configurations are tuned to a 7.25 Hz bandwidth, deemed appropriate and most widely used for these vehicle configurations. The engine models compute reacted torque based on control input commands from the vehicle controller as opposed to previous vehicle configurations in which reacted torques were computed to maintain trim rotor speeds.

5.2 Trade Studies

5.2.1 Transient Power Sensitivity to Quadrotors

5.2.1.1 Transient Power Sensitivity to All-Electric Quadrotor Drive System Configuration

Several simulation scenarios are used to evaluate transient power requirements of the all-electric quadrotor vehicle configuration. The disturbance rejection analysis is conducted for vehicle configurations both with and without the cross-shafted drive system with all engines operating. Additionally, single engine failures are simulated for the cross-shafted quadrotor configuration. In general, the cross-shafted drive system results in lower peak transient power requirements in gust and turbulence conditions.

In the presence of 17 knot steady winds prescribed by regulatory guidance identified in Section 5.1.1.1, benign transient power requirements are seen for the all-electric quadrotor with and without the cross-shafted drive system. However, the maximum transient power values with the cross-shafted drive system are lower than those without cross-shafting due to the load sharing nature of the drive system. Figure 11 shows the distribution of maximum transient power required in the presence of steady winds.

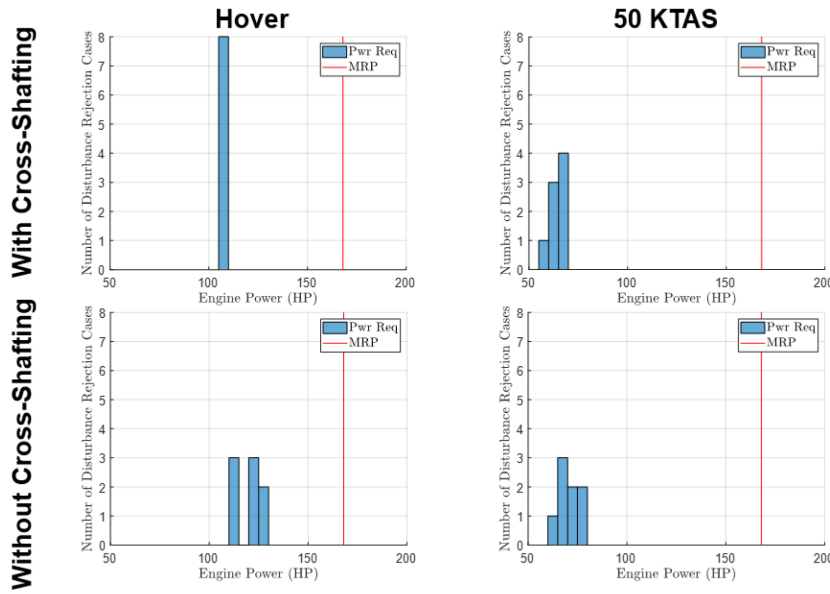


Figure 11: Distribution of maximum engine power required for disturbance rejection in the presence of steady winds.

Figure 12 shows the distribution of maximum engine power required in the presence of discrete gust profiles described previously (i.e. a range of gust widths injected in the ±x, y, and z directions). Results are shown in hover, approximate maximum endurance, and high speed forward flight trim conditions for vehicle configurations with and without cross-shafting. The maximum rated power (MRP) value for the all-electric quadrotor is 168 horsepower (HP) per motor (ref. 4).

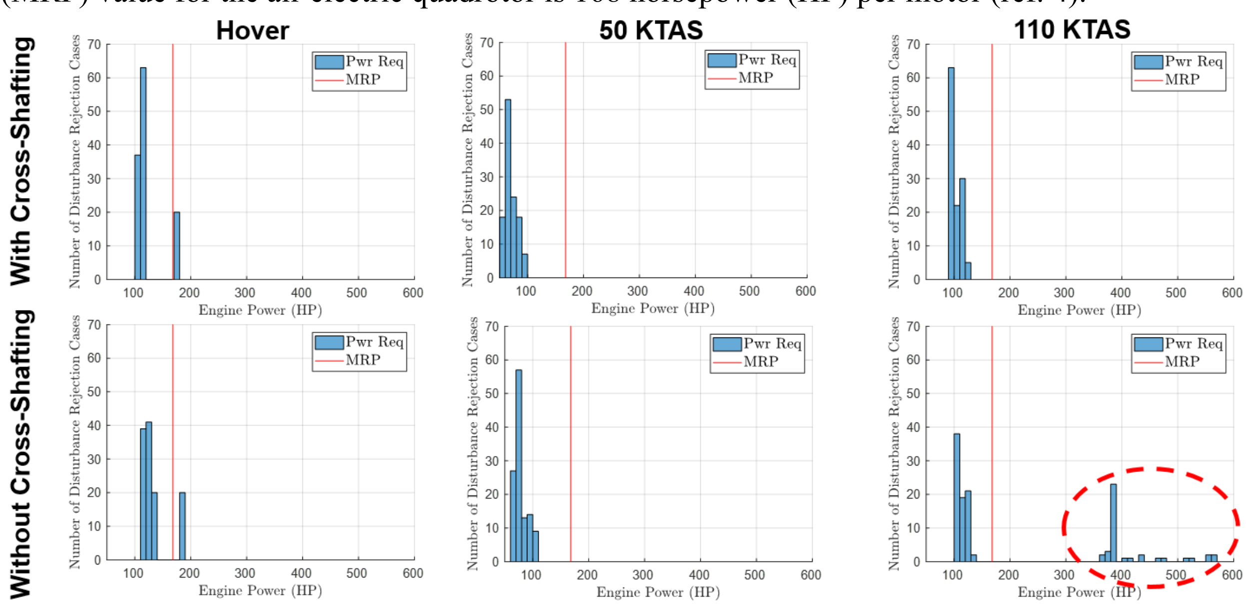


Figure 12: Distribution of maximum engine power required for disturbance rejection in the presence of discrete gusts. Large transient power requirements are highlighted in red.

Figure 13 shows the distribution of maximum engine power required in the presence of continuous turbulence conditions for the same vehicle configurations and trim conditions as for discrete gusts. As described in Section 5.1.1.3, a limited number of continuous turbulence cases are evaluated for this vehicle configuration.

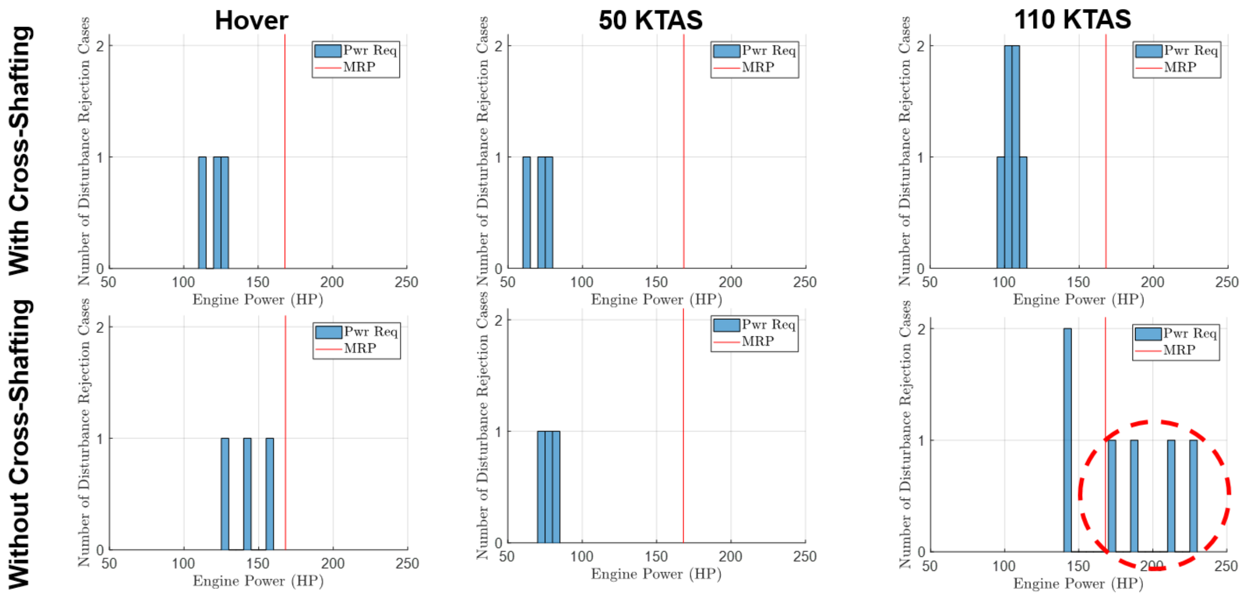


Figure 13: Distribution of maximum engine power required for disturbance rejection in the presence of continuous turbulence. Large transient power requirements are highlighted in red.

Injection of discrete gusts and continuous turbulence both result in high transient power requirements for the non-cross shafted vehicle configuration, particularly in high speed forward flight. Given the simple vehicle controller and tuning scheme implemented for the purposes of this analysis, transient power requirements are shown to exceed MRP particularly for trim conditions near V_{max}, the maximum continuous power forward flight speed. However, such transient power exceedances of MRP may be seen at lower airspeeds with increases in the gross weight of the vehicle. The cross-shafted drive system significantly reduces the maximum transient power required to reject these discrete gust disturbances. Power can be provided to highly loaded rotors via the cross-shafted drive system in order to reduce transient power requirements at each engine.

Figure 14 shows a comparison of maximum transient power requirements in both nominal and single engine failure conditions in the presence of discrete gusts and continuous turbulence. This analysis was performed for the all-electric quadrotor with a cross-shafted drive system. Note that with this rigid drive system, the single engine failure results are independent of which engine fails. In discrete gust conditions, engine failures are injected at the time of peak gust velocity. In continuous turbulence conditions, engine failures are injected at a predetermined simulation time.

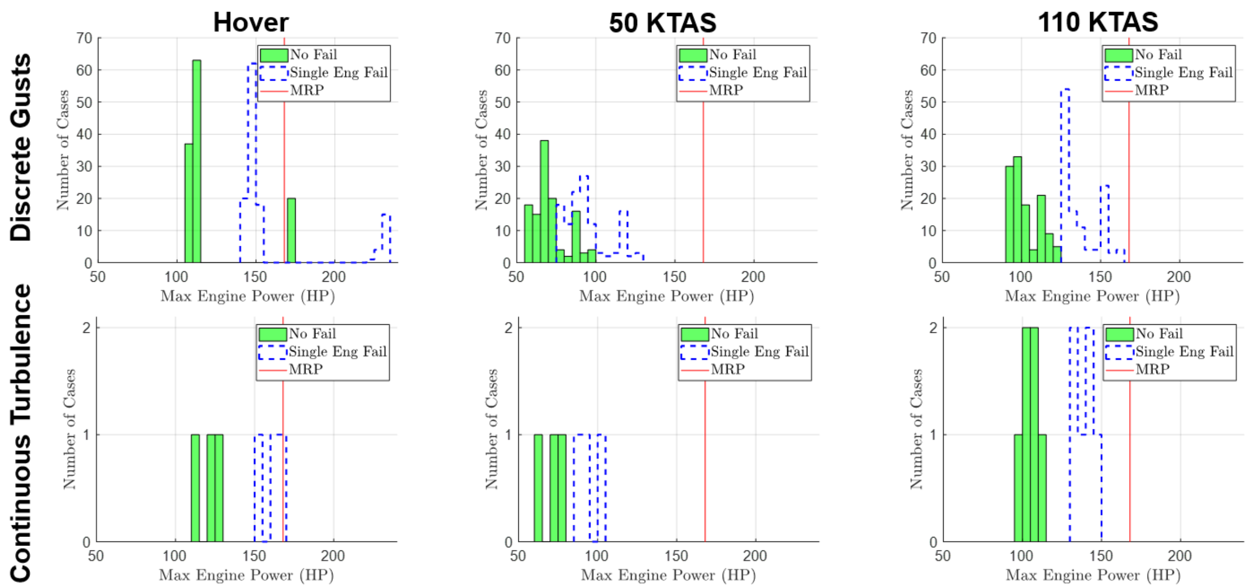


Figure 14: Distribution of maximum engine power required for disturbance rejection in the presence of discrete gusts and continuous turbulence for the all-electric quadrotor with cross shafting in both nominal and single engine failure conditions.

The trends in maximum transient power required are roughly consistent in nominal and single engine failure conditions. However, with three engines operable the transient power values increase proportionally. Particularly in hover, top-down discrete gusts result in maximum transient power requirements exceeding MRP given the current vehicle controller configuration.

5.2.1.2 Extension of Results to Hybrid and Turboshaft Quadrotors

The model setup described previously is at the level of fidelity of total power draw at each rotor; the vehicles are not modeled with sufficient fidelity to make a distinction in the source of power. Therefore, the simulation model architecture is equivalent for all quadrotor variants. Given this similarity, separate analysis results are not presented for the hybrid and turboshaft quadrotor vehicle configurations. The trends presented for the all-electric quadrotor therefore can be extended to the hybrid and turboshaft vehicle configurations. Variations in the linear model and trimmed state of the vehicle due to changes in inertial properties may result in differences in exact maximum transient power values, but trends are expected to remain similar across configurations.

5.2.2 Transient Power Sensitivity to Hexarotor Control Scheme

Several simulation scenarios are used to evaluate transient power required for disturbance rejection for the collective- and RPM-controlled hexarotor vehicle configurations. Steady winds, discrete gusts, and continuous turbulence are used for the purposes of this analysis. Additionally, single engine failures are injected in continuous turbulence conditions. Notably, as described previously, the continuous turbulence conditions evaluated for these vehicle configurations have been developed significantly over those used to evaluate the all-electric quadrotor, including analysis of varying surface roughness values typical of urban environments.

The subsequent transient power results (Table 5) are shown with respect to MRP values estimated for these vehicle configurations. The MRP value for the all-electric quadrotor is approximately 156% of the average engine power required for trim in hover (ref. 4). The MRP values for the hexarotor configurations are similarly estimated as 156% of engine power required for trim in hover, summarized in the table below.

Table 5: Estimated MRP values for collective- and RPM-controlled hexarotor configurations.

Collective Hex MRP		RPM Hex MRP	
Per Engine (HP)	**Total (HP)**	**Per Engine (HP)**	**Total (HP)**
115.2	691.2	107.5	645.0

In the presence of steady winds, the vehicle response for both configurations is benign with respect to both the engine power required and vehicle output state errors. Figure 15 shows the distribution of maximum transient power required in steady wind conditions. Note that the engine power required is shown with respect to the estimated MRP values described previously.

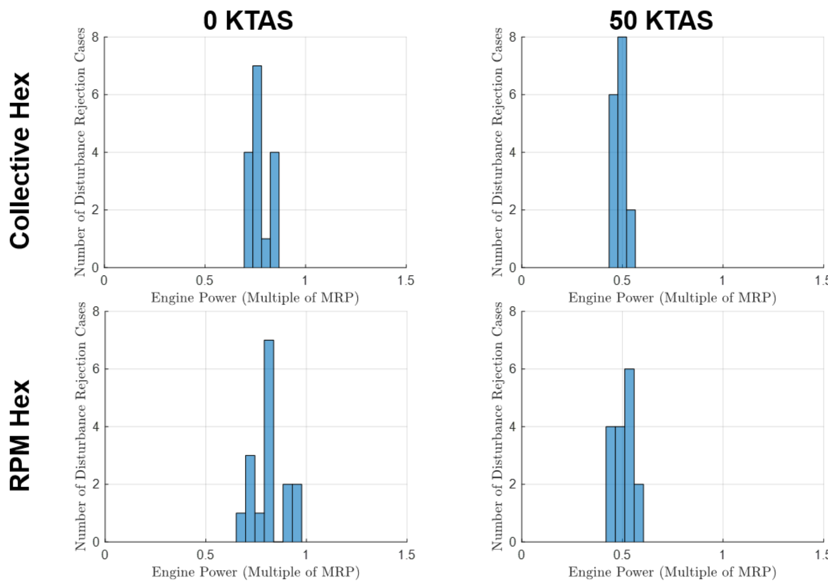


Figure 15: Distribution of maximum engine power required for disturbance rejection in the presence of steady winds for the collective- and RPM-controlled hexarotor vehicle configurations.

Discrete gust injection results in substantial differences in transient power requirements between the collective- and RPM-controlled hexarotor configurations. Figure 16 shows the distribution of maximum transient power required for disturbance rejection in the presence of discrete gust profiles representative of an urban canyon environment.

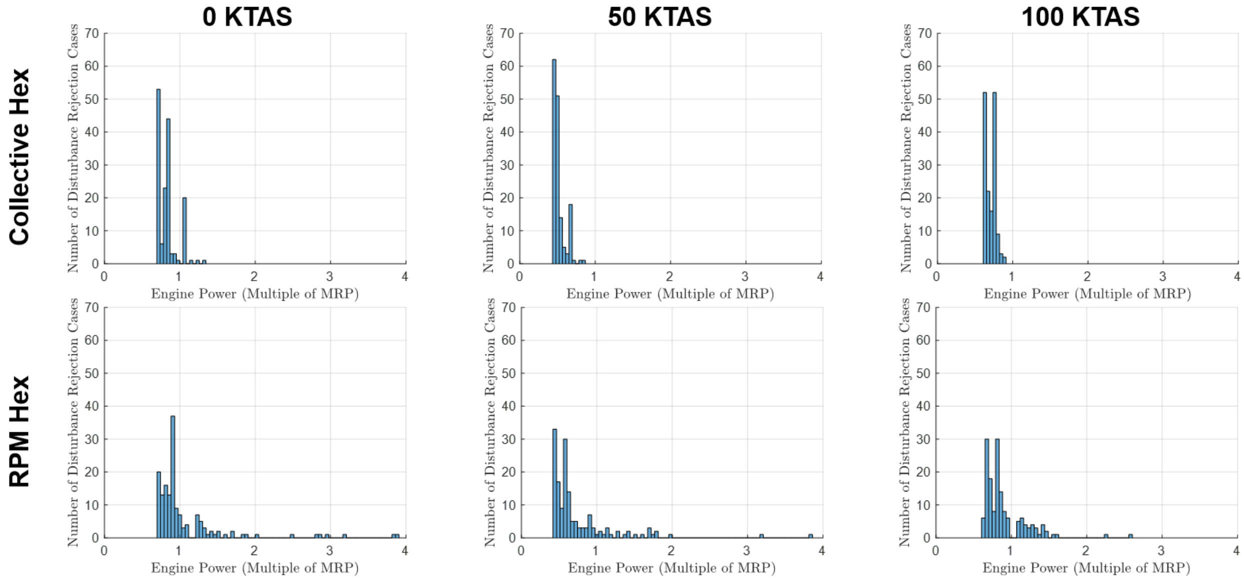


Figure 16: Distribution of maximum engine power required for disturbance rejection in the presence of discrete gusts for the collective- and RPM-controlled hexarotor vehicle configurations.

Likewise, Figure 17 shows the distribution of transient power required for disturbance rejection in the presence of continuous turbulence for both nominal and single engine failure conditions. For the purposes of this analysis, a failure in Engine 1 was injected at a prescribed simulation time for the single engine failure scenarios.

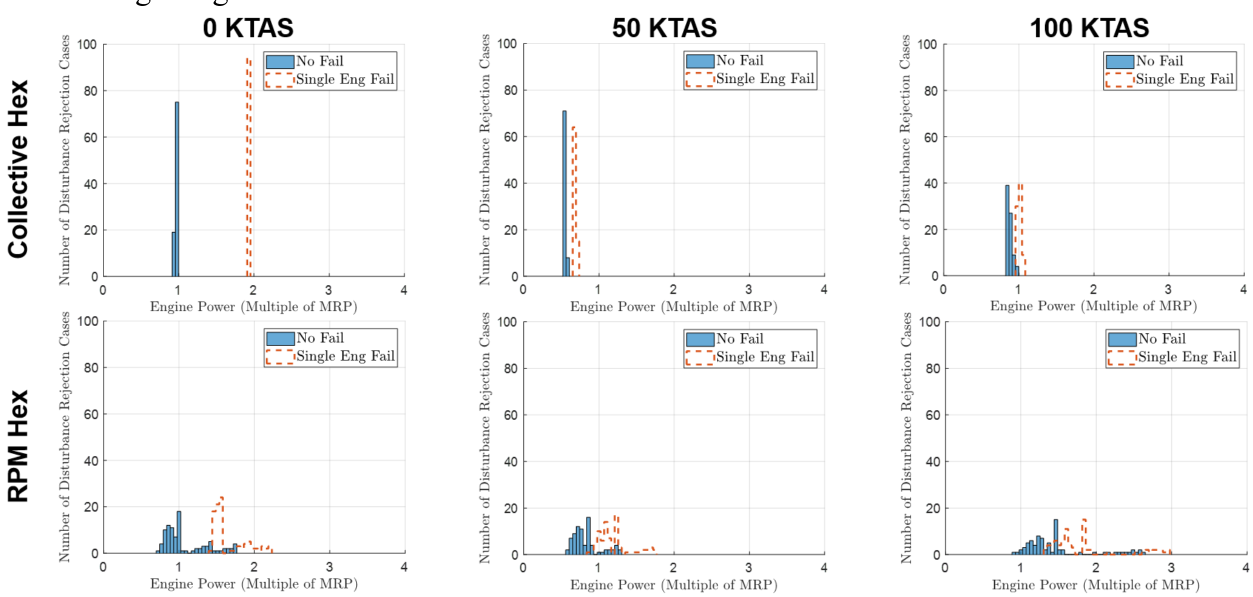


Figure 17: Distribution of maximum engine power required for disturbance rejection in the presence of continuous turbulence for the collective- and RPM-controlled hexarotor vehicle configurations in both nominal and single engine failure conditions.

In both discrete gust and continuous turbulence conditions, the distribution of maximum transient power requirements is much tighter for the collective-controlled hexarotor than the RPM-controlled hexarotor. The average maximum transient power required over the distribution of gusts and turbulence injected in the model is higher for the RPM-controlled hexarotor than the collective controlled hexarotor, particularly in forward flight, and furthermore the RPM-controlled hexarotor shows a more heavy-tailed distribution in worsening atmospheric disturbance conditions with respect to transient power requirements. In single engine fail conditions, the trends in transient power requirements remain consistent with the trends in nominal operating conditions. However, the transient power required in the five remaining engines increases proportionally.

Figure 18 shows transient power requirements as a function of surface roughness length in order to assess vehicle performance across a variety of urban environments within, and outside of, city centers. Note that the unrealistic 50 ft. AGL/100 KTAS flight condition is not included in this analysis.

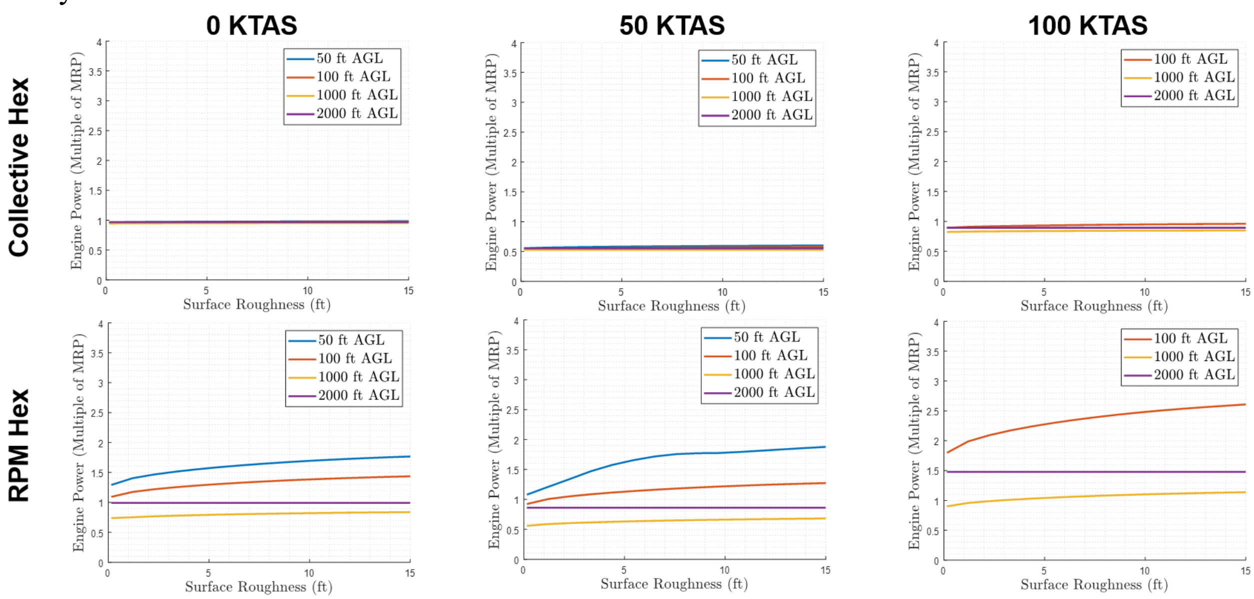


Figure 18: Maximum engine power required for disturbance rejection in the presence of continuous turbulence as a function of surface roughness length for the collective- and RPM-controlled hexarotor vehicle configurations.

The distribution of transient power required for the collective-controlled hexarotor is much tighter with respect to variations in surface roughness. Furthermore, as shown previously, the worst-case transient power requirements for the collective-controlled hexarotor are substantially lower than for the RPM-controlled hexarotor. This suggests that the atmospheric conditions of the target operating environment do not have a significant impact on engine sizing for the collective-controlled hexarotor, but may be significant for engine sizing the RPM-controlled hexarotor.

5.2.2.1 Hexarotor Collective Actuator Failures

In order to investigate the impact on transient power requirements of collective actuator failures, such failures are simulated for the collective-controlled hexarotor over a range of failure durations. The actuator failure duration represents the time required to detect and correct a collective actuator failure, for example, the time to hand off from one actuator to another as a brake engages. Simulations are run for failures in place and ±10% of actuator position in order to approximate the transient power required in a fail free condition. Figure 19 shows maximum transient power requirements in continuous turbulence conditions with an actuator failure injected at a predetermined simulation time. Results are shown for hover, maximum endurance, and high speed forward flight trim conditions.

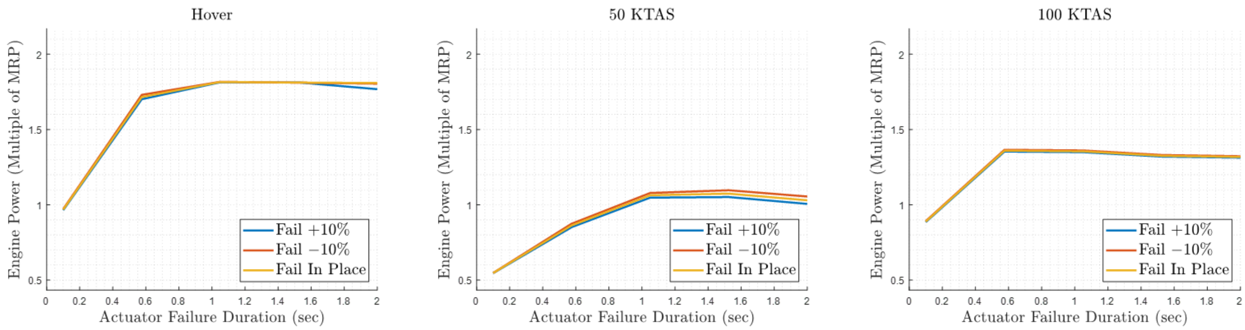


Figure 19: Maximum engine power required for disturbance rejection in the presence of continuous turbulence as a function of actuator failure duration for the collective-controlled hexarotor vehicle configuration.

The transient power required to handle a collective actuator failure with respect to estimated MRP does not have a significant impact on the maximum transient power requirements of the vehicle in these conditions if the actuator failure duration is sufficiently small, on the order of a few tenths of a second. As the failure duration increases past a threshold of approximately half a second in hover and high-speed flight and one second near maximum endurance airspeed, a longer failure duration does not lead to higher transient power required. Put another way, if this threshold is exceeded, a faster failure detection and correction does not lead to lower maximum transient power requirements in these conditions. These results can help inform future vehicle design attributes in the areas of actuator configuration and sizing.

5.2.2.2 Hexarotor Vehicle Controller Improvements

In order to improve vehicle performance of the RPM-controlled hexarotor in the presence of single engine failures, a method of reconfiguring the vehicle controller is presented. Upon injection of the (zero torque) failure, the mixing matrix between stick commands from the vehicle controller and individual rotor speed commands is switched such that the engine complementary to the failed engine is shut down (zero torque). The mixing matrix is recalculated such that the same primary axis response is generated by the remaining four engines/rotors for a given stick command. Figure 20 shows a comparison of the vehicle attitude and position with respect to disturbance rejection metrics in the hover trim condition in the presence of continuous turbulence. This example shows a failure in Engine 1 leading to a shutdown of Engine 6.

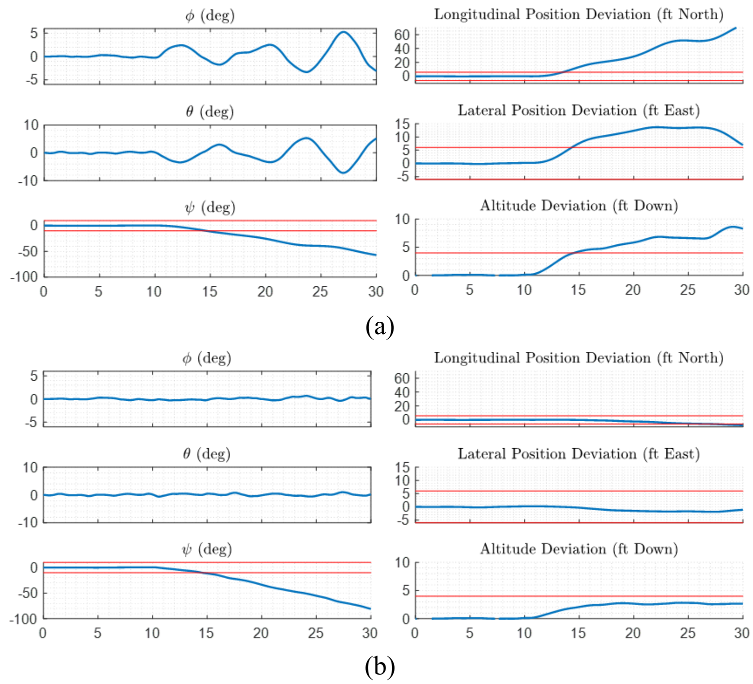


Figure 20: Vehicle performance with respect to disturbance rejection metrics (red lines) in the presence of continuous turbulence and a single engine failure injected at t=10 seconds (a) without reconfiguring the vehicle controller, and (b) with the controller reconfigured to shut down the engine complementary to the failure.

Clearly, modification to the baseline vehicle controller offers a means to greatly improve position response with respect to disturbance rejection metrics for the RPM hexarotor vehicle configuration. However, these results illustrate the need for further control law development in order to handle off-nominal operating conditions, particularly to improve heading response in the single engine failure condition of this example.

5.2.3 *Transient Power Sensitivity to Number of Rotors*

Several simulation scenarios are used to evaluate transient power required for disturbance rejection for the RPM-controlled octorotor in comparison to the RPM-controlled hexarotor vehicle configuration. The same steady wind, discrete gust, and continuous turbulence described in the previous section are analyzed. In addition, single engine failures are injected in continuous turbulence conditions.

The subsequent transient power results are shown with respect to MRP values estimated as 156% of engine power required for trim in hover as described previously and summarized in Table 6.

Table 6: Estimated MRP values for RPM-controlled hexarotor and octorotor configurations.

RPM Hex MRP		RPM Oct MRP	
Per Engine (HP)	**Total (HP)**	**Per Engine (HP)**	**Total (HP)**
107.5	645.0	89.1	712.8

In the presence of steady winds, the vehicle response for both configurations is benign with respect to both the engine power required and vehicle output state errors. Figure 21 shows the distribution of maximum transient power required by an individual engine. Note that the engine power required is shown with respect to the estimated MRP values described previously.

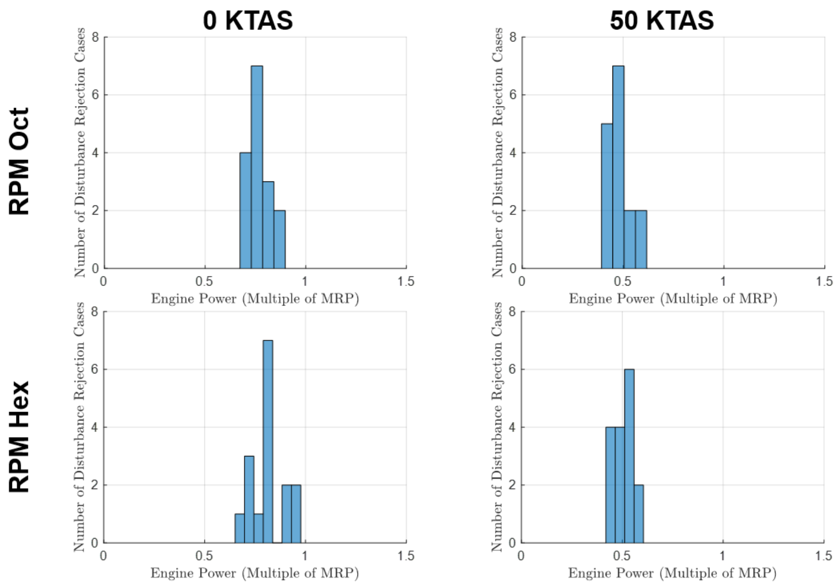


Figure 21: Maximum engine power required for disturbance rejection in the presence of steady winds for the RPM-controlled hexarotor and octorotor vehicle configurations.

Discrete gust injection results in very similar distributions of transient power required for both the RPM-controlled hexarotor and octorotor vehicle configurations. Figure 22 shows the distribution of maximum transient power required for disturbance rejection in the presence of discrete gust profiles representative of an urban canyon environment.

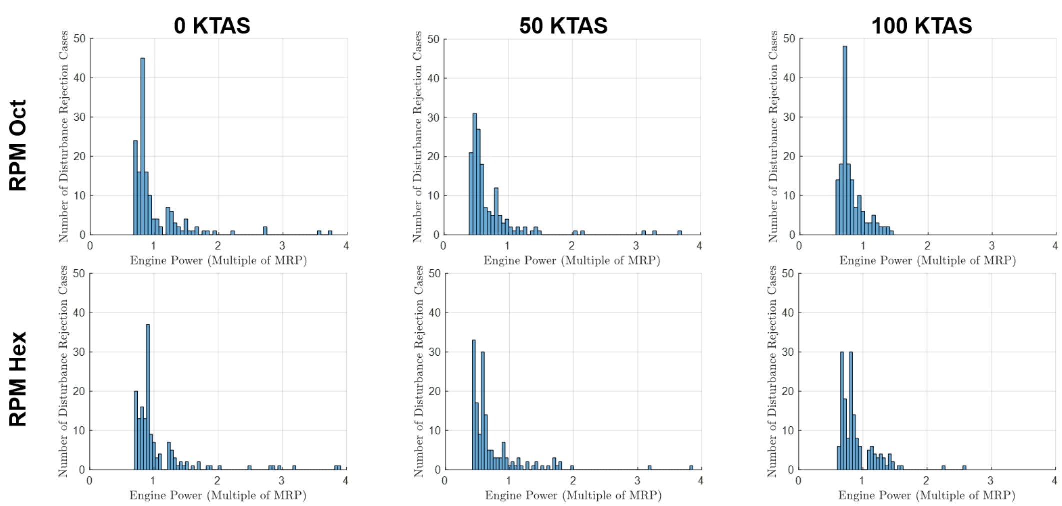


Figure 22: Maximum engine power required for disturbance rejection in the presence of discrete gusts for the RPM-controlled hexarotor and octorotor vehicle configurations.

Likewise, Figure 23 shows the distribution of transient power required for disturbance rejection in the presence of continuous turbulence for both nominal and single engine failure conditions. For the purposes of this analysis, a failure in Engine 1 was injected at a prescribed simulation time for the single engine failure scenarios.

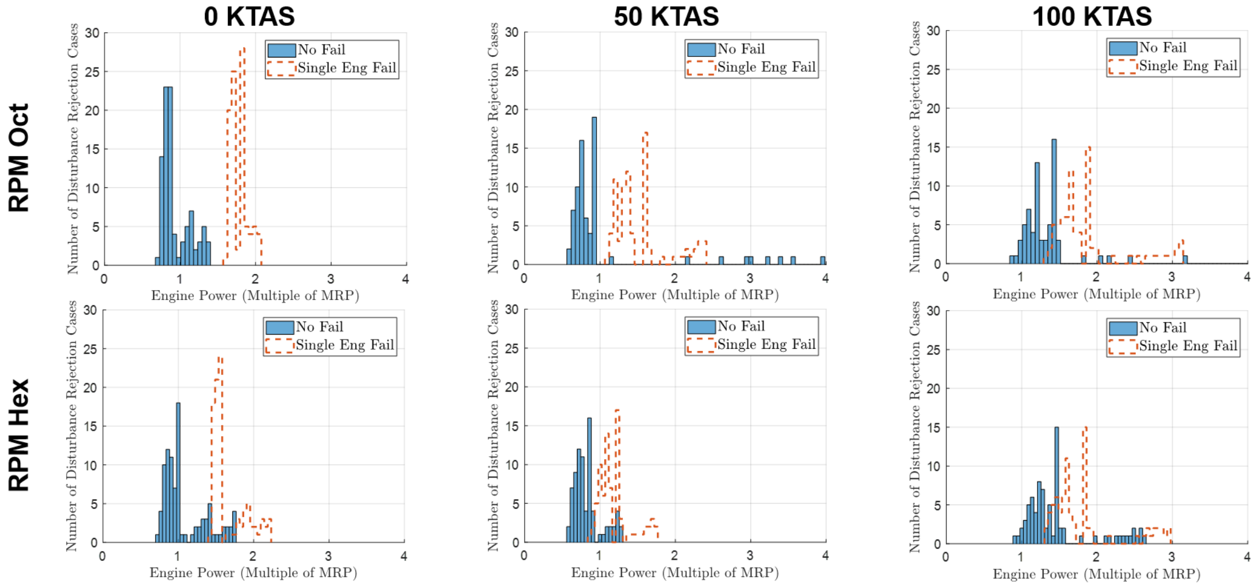


Figure 23: Maximum engine power required for disturbance rejection in the presence of continuous turbulence for the RPM-controlled hexarotor and octorotor vehicle configurations.

In both gust and turbulence conditions, the distribution of maximum transient power required is similar between the RPM-controlled hexarotor and octorotor vehicle configurations with respect to estimated MRP values in the majority of conditions evaluated. Note that the abscissa in Figure 23 is limited to 4×MRP. In forward flight, however, several worst-case low-altitude turbulence conditions result in larger transient power requirements for the octorotor than the hexarotor, which are addressed in more detail with respect to surface roughness below. In single engine fail conditions, transient power requirements are increased with respect to MRP in the octorotor as compared to the hexarotor. However, it is clear that transient power requirements are much less sensitive with respect to the number of rotors than with respect to RPM versus collective control schemes in the presence of atmospheric disturbances expected in an urban canyon environment.

Figure 24 shows transient power requirements as a function of surface roughness length in order to assess vehicle performance across a variety of urban environments outside of and within city centers. The unrealistic 50 ft. AGL/100 KTAS flight condition is not included in this analysis.

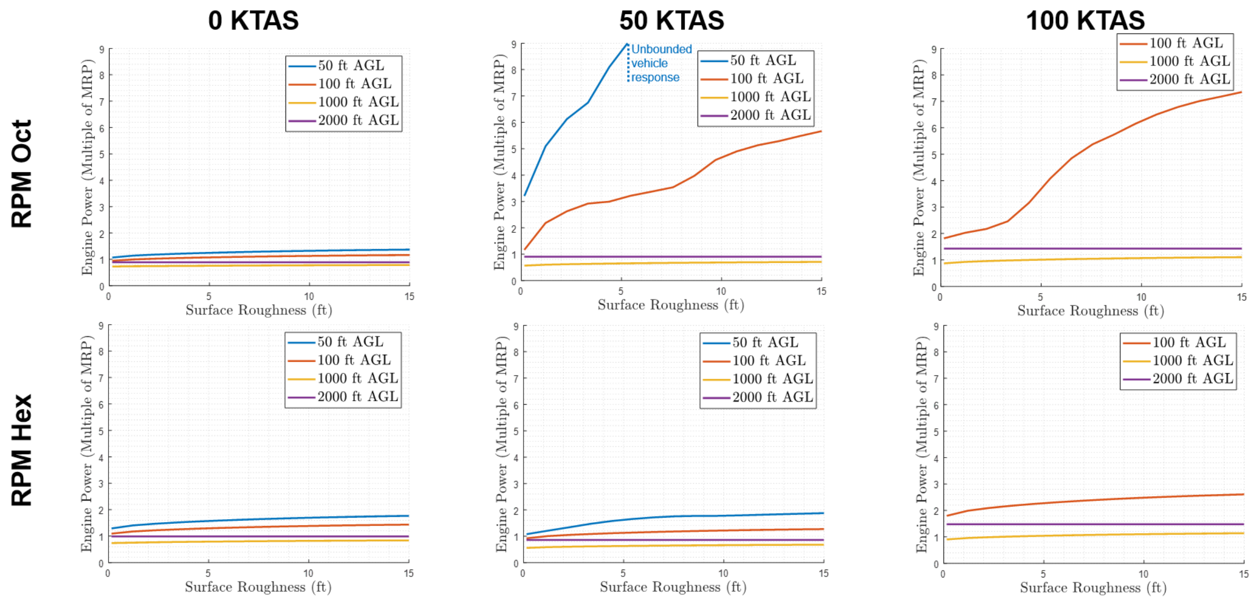
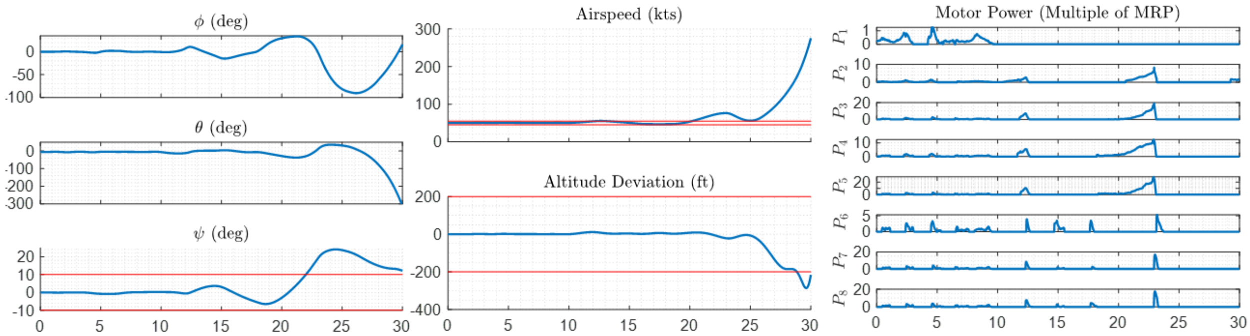


Figure 24: Maximum engine power required for disturbance rejection in the presence of continuous turbulence as a function of surface roughness length for the RPM-controlled hexarotor and octorotor vehicle configurations.

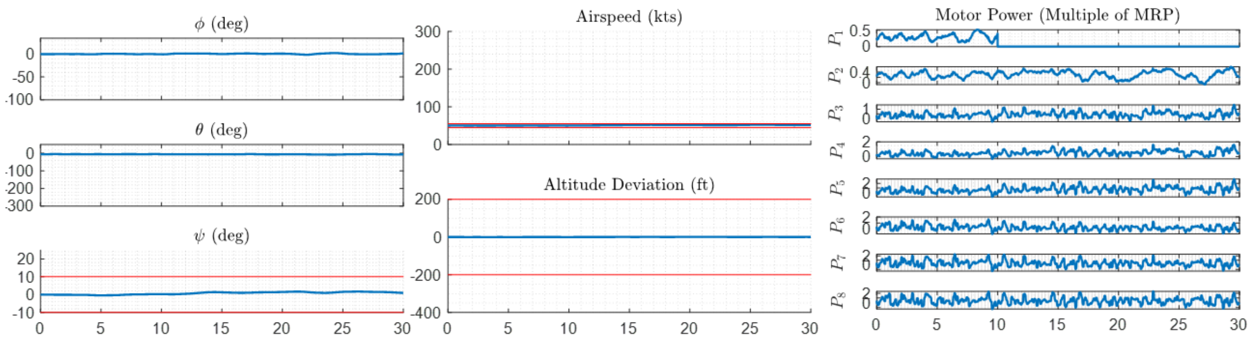
In forward flight trim conditions, the maximum transient power required is substantially higher for the octorotor than the hexarotor, which is driven by outlying worst-case turbulence conditions as seen in Figure 24. In forward flight, the hexarotor sees much lower variation in power requirements with increasing surface roughness than the octorotor. This may be due to inertial properties of the vehicle such as increased pitch and yaw inertia of the octorotor in comparison to the hexarotor. The simple vehicle controller implemented for the purposes of generating direct comparisons between vehicle configurations is not adequate to ensure a bounded response in all turbulence conditions, as seen in the 50 KTAS/50 ft. AGL flight condition for the octorotor in Figure 24. Additional vehicle design efforts are required to ensure a bounded response in highly turbulent environments in such conditions. Worst-case transient power requirements may be reduced with further development of the vehicle controller or by other design decisions such as active rotor braking.

Active Rotor Braking

Active rotor braking offers a potential mechanism by which the large maximum transient power requirements seen for the RPM-controlled octorotor configuration may be mitigated. Figure 25 shows a comparison of vehicle performance with respect to disturbance rejection metrics with and without active rotor braking capability in the 50 KTAS trim condition in the presence of continuous turbulence conditions and an engine one failure. Without active rotor braking capability, worst-case turbulence conditions can result in the vehicle essentially performing as though flying open-loop after sufficient time has elapsed.



(a)



(b)

Figure 25: Vehicle performance with respect to disturbance rejection metrics (red lines) and power required in the presence of continuous turbulence and a single engine failure injected at t=10 seconds (a) without active rotor braking, and (b) with active rotor braking.

It is clear that active rotor braking capability offers significant improvement in vehicle performance with respect to disturbance rejection metrics as well as power requirements. Future design work would determine the mechanism by which this braking is achieved, although regenerative motor braking may be attractive from an energy recovery standpoint in addition to controllability implications discussed here.

5.3 Summary and Conclusions

The simulation results presented here provide several key insights with regard to transient power requirements of vehicle design attributes. Transient power requirements for gust and turbulence conditions are significantly lower for the collective over RPM control, particularly in forward flight. Although the power required to vary collective pitch is not explicitly captured in these results, this is expected to be small in comparison to primary lift power, with typical tandem rotor helicopters requiring roughly 1% of total lifting power for actuation. Furthermore, the collective control scheme is less sensitive to variations in surface roughness typical of urban environments than RPM control. These results indicate that transient power requirements are largely a function of rotor inertia that must be overcome in RPM control schemes for disturbance rejection in this environment. The sensitivity of transient power requirements to the number of rotors in the vehicle configuration is not nearly as great as that of the vehicle control scheme, as evidenced in the comparison between RPM-controlled hexarotor and octorotor configurations. However, larger transient power requirements are seen in the octorotor in the presence of single engine failures, as well as worst-case turbulence conditions in forward flight. Finally, cross-shafting in quadrotor configurations is shown to provide a means to significantly reduce transient power requirements, particularly in high-speed flight.

With regard to failure conditions, trends in transient power requirements for all vehicle configurations remain similar to nominal conditions with proportional increases in power requirements in the remaining operable engines. If collective actuator failures are able to be detected and corrected quickly (on the order of 0.1 seconds), little impact is seen in transient power requirements of the vehicle. Once the failure duration exceeds a threshold (on the order of 0.5 to 1 second), longer failure durations do not lead to higher power requirements. In other words, faster failure correction does not lead to lower transient power requirements once this threshold is exceeded. In RPM control schemes, reconfiguration of the vehicle controller in order to command zero torque from the engine complementary to the failure can significantly improve vehicle response with respect to disturbance rejection metrics, and active rotor braking offers a potential mechanism to reduce the large transient power requirements, including in the presence of engine failures.

Further design work is needed in order to avoid the large transient power requirements present in these simulation results for all vehicle configurations. Power requirements are highly dependent on the implementation of the aircraft control system, and development beyond the simple LQR vehicle controller implemented for the purposes of this analysis offers significant room to improve with respect to power required for disturbance rejection. In addition, design decisions such as power clipping offer mechanisms by which these transient power spikes can be reduced.

Future work should focus on simulation and analysis of urban air mobility concept vehicles using nonlinear vehicle models, including appropriate control law development to ascertain the impact of failure injection and pilot reactions. Nonlinear models will be able to facilitate varying mission parameters such as multiple gross weight conditions (e.g., low gross weight versus low power margin in hover) in addition to modeling rotor-to-rotor interference. Pilot-in-the-loop simulations using such nonlinear models will help determine how pilots respond to failure conditions as well as design decisions made to reduce maximum transient power requirements such as power clipping. Furthermore, these simulations allow opportunities to investigate the tradeoff between reduction in transient power requirements for disturbance rejection and pilot workload and ride quality. Pilot-in-the-loop simulation activities also offer opportunities to assess these metrics in critical phases of flight such as climb-outs and descents. In addition, pilot-in-the-loop simulation could be used to perform comparisons between vehicle architectures in the context of how easy or costly it would be to train pilots and keep them sufficiently proficient.

Further refinement of wind models should be investigated in order to improve fidelity with regard to surface roughness variations, surface heating, and wake modeling in representative urban canyon environments. The mean wind profile and turbulence intensities, for example, are influenced not only by roughness of the terrain immediately below, but also by upwind topography. Added fidelity of these wind models will further the benefits of nonlinear simulation models described previously.

6 FUNCTIONAL BLOCK DIAGRAMS

A Functional Block Diagram (FBD) is a schematic representation of the aircraft functions analyzed. The FBD is used as input to the FHA and FMECA. The FMECA uses the functional block diagram as a basis to postulate failure modes and analyze effects. The FBD is used as a basis for the FHA to develop the list of failure conditions based on the functional configuration of the air vehicle.

The FBDs are organized by two aircraft level functions and five main sub-functions to highlight the key differences between air vehicle configurations. These aircraft-level functions and main sub-functions are:

<u>Aircraft Level Function: Transmit Adequate Power to Rotors</u>

Function 1: Power plant

Function 2: Convert electrical energy to shaft torque

Function 3: Transmit torque to rotors

<u>Aircraft Level Function: Collective Control of Rotors</u>

Function 4: Actuation

Function 5: Flight Control System (collective pitch control or RPM control functions)

Each block is representative of a *function* and is not indicative of a single *component*. A block may represent an individual system, a system of systems needed to achieve the desired function, or it may represent the components as redundancy designed into that function. For example, the drive systems are represented by larger, all-encompassing blocks (e.g., “Gearbox #1” includes all gears, shafting, etc. associated with the #1 rotor gearbox) and the “ESC” blocks represent the electronic speed controller (ESC) function, not a single ESC component.

Redundancy that was anticipated prior to detailed design, based on knowledge of current technology reliability, is incorporated in the FBD. For example, the quadrotor configurations were anticipated to require triplex hydraulic systems, represented by three hydraulic pumps, shown in Figure 27. As another example, the final design for the hexarotors includes two ESCs per motor, but they are illustrated in the FBD as a single function. Equipment such as rotors and fuel systems are illustrated with dashed lines on the FBD, but not assigned functions as they are not considered in the analysis. Similarly, the quadrotors utilize overrunning clutches so that the rotors can continue to operate if one motor needs to be shut down, while the hexarotors and octorotor allow for one rotor to be shut down if the corresponding motor needs to be deenergized.

The following subsections describe the essential functions of the propulsion system and summarize the main differences between configurations. The eQuad aircraft level functions and essential sub-functions are color coded in the FBD as shown in Figure 26.

Aircraft Level Functions	**Main Functional Blocks**
Transmit Adequate Torque to Rotors	1. Powerplant 2. Convert electrical energy to shaft or motor torque 3. Torque to rotors
Collective Control of Rotors	4. Actuation 5. Flight Control System

Figure 26: Aircraft Level Functions and Main Functional Blocks

6.1 Electric Quadrotor

The eQuad with cross-shafting FBD is shown in Figure 27. There are all five essential functions, described below.

- Function 1: Power plant. This consists of the High Voltage (HV) Battery Network, HVDC Power Management and Distribution, low voltage direct current (LVDC) power, and battery cooling.
- Function 2: Convert electrical energy to shaft torque. This function consists of the Electronic Speed Controllers (ESCs), electric motors, associated motor/ESC cooling, FCS control signals and feedback sensors, and overrunning clutches.
- Function 3: Transmit torque to rotors. This function consists of the rotor gearboxes and interconnecting cross shafting and combiner gearboxes.
- Function 4: Actuation. This function consists of hydraulic actuators at each rotor and three hydraulic pumps driven by the combiner gearboxes.
- Function 5: Flight Control System. This function consists of the flight control computing, inertial data, air data, rotor sensing, and collective pitch actuation sub-functions.

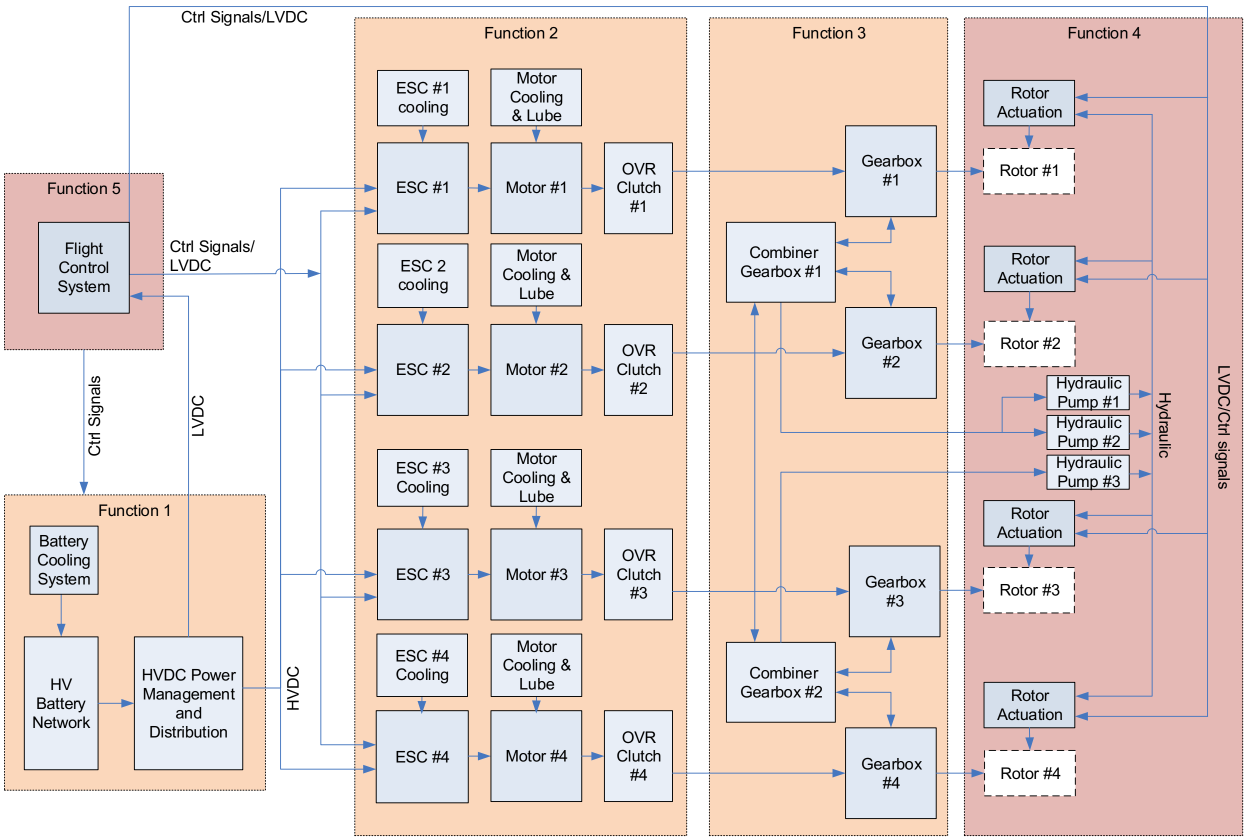


Figure 27: Baseline eQuad FBD (w/cross shafting)

6.2 Hybrid-Electric Quadrotor

The hybrid-electric quadrotor FBD is shown in Figure 28. There are all five essential functions, described below.

- Function 1: Power plant. This consists of the HV battery network, associated battery cooling, fuel system, engine, engine gearbox, generator, HVDC power management and distribution, and LVDC power.
- Function 2: Convert electrical energy to shaft torque. This function is identical to the eQuad, consisting of the ESCs, electric motors, associated motor/ESC cooling, FCS control signals and feedback sensors, and overrunning clutches.
- Function 3: Transmit torque to rotors. This function is identical to the eQuad, consisting of the rotor gearboxes and interconnecting cross shafting and combiner gearboxes.
- Function 4: Actuation. This function is identical to the eQuad, consisting of three hydraulic actuators at each rotor and hydraulic pumps driven by the combiner gearboxes.
- Function 5: Flight Control System. This function is identical to the eQuad, consisting of the flight control computing, inertial data, air data, rotor sensing, and collective pitch actuation sub-functions.

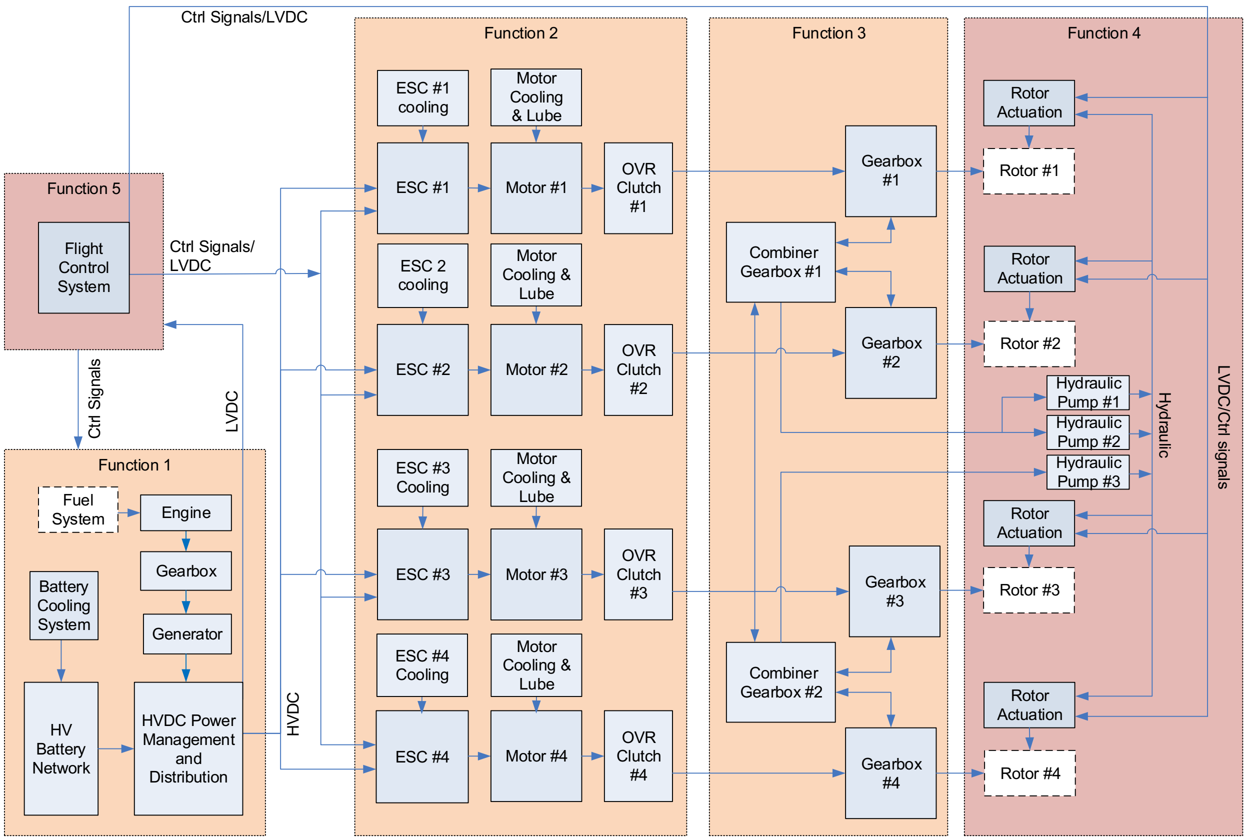


Figure 28: hQuad FBD

Relative to the eQuad FBD, the only change to the hQuad is found within Function 1. The hQuad power plant consists of a HVDC battery network, but also a turbo-generator system providing the primary source of electric propulsion power, represented by the “fuel system”, “engine”, “gearbox”, and “generator” blocks. The fuel system was excluded from this analysis because of its similarity to existing, fielded designs.

Since the only change from the eQuad was within the function supplying electricity to the motors, the remaining functional blocks represented by Functions 2, 3, 4, and 5 are identical to the eQuad.

6.3 Twin Turboshaft Quadrotor

The tQuad FBD is shown in Figure 29. There are four essential functions described below, noting the elimination of Function 2 associated with ESCs/Motors, etc.

- Function 1: Power plant. This consists of dual redundant engines, engine gearboxes, over running clutches, lubrication and cooling systems, and fuel system.
- Function 3: Transmit torque to rotors. This function is identical to the eQuad, consisting of the rotor gearboxes and interconnecting cross shafting and combiner gearboxes.
- Function 4: Actuation. This function is identical to the eQuad, consisting of three hydraulic actuators at each rotor and hydraulic pumps driven by the combiner gearboxes.
- Function 5: Flight Control System. This consists of the flight control computing, inertial data, air data, rotor sensing, and collective pitch actuation sub-functions. Note: the LVDC

system was not developed, but assumed to have the same level of redundancy as the other quadrotor configurations, as discussed in Section 11.1.

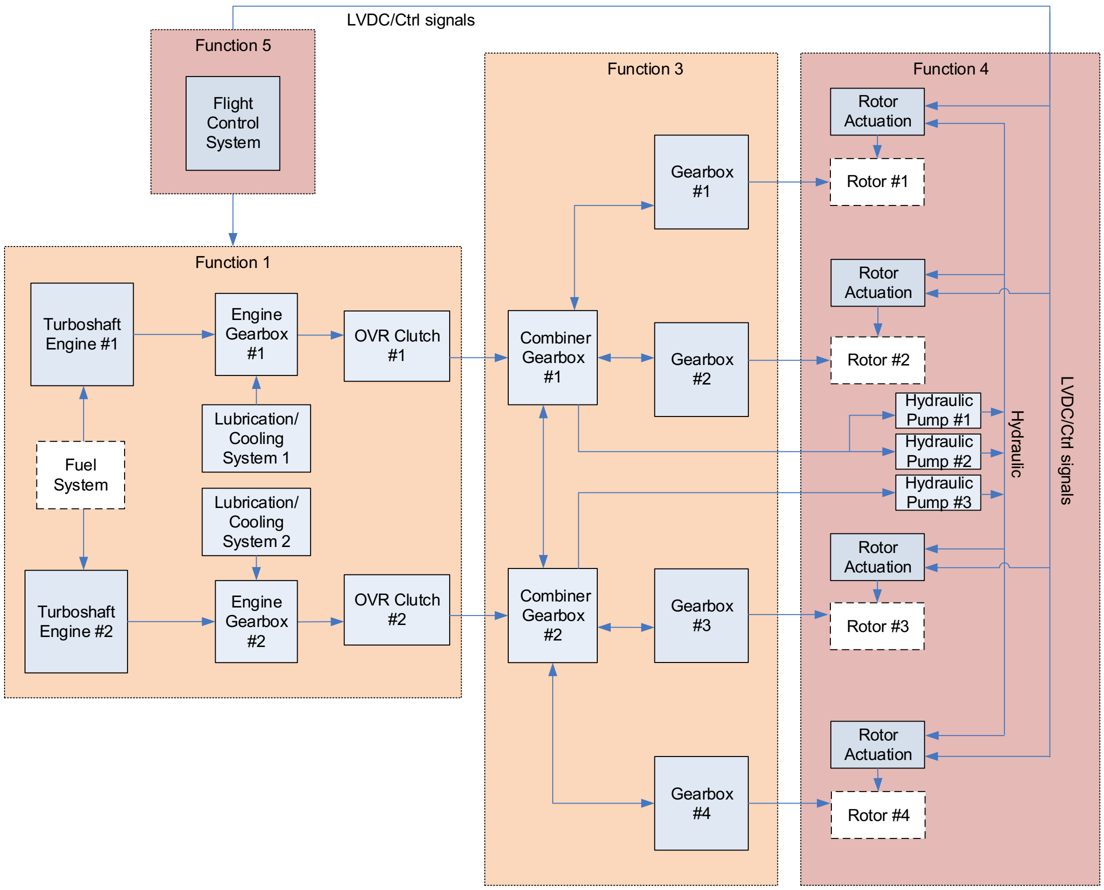


Figure 29: tQuad FBD

Relative to the hQuad, there are two main functional changes: the power plant and the torque generation to transmission. The tQuad power plant consists of two fully redundant turboshaft systems and no electric propulsion sources (e.g., no battery or generator), represented by the “fuel system”, “engine”, and “gearbox”. The fuel system was included in tQuad analysis as an underdeveloped functional block; many existing, fielded designs exist including certification specifications to aid in the safe design of conventional fuel systems. The tQuad configuration does not have electric propulsion sources, therefore there are no ESCs/motors required for conversion of electricity to shaft torque. This is represented in the FBD by removal of the Function 2 block and incorporating the overrunning clutches into Function 1.

The FBD shows the tQuad Function 1 and Function 3 layouts and interconnectivity. Another nuance to the tQuad is the interconnection between the overrunning clutches to the combiner gearboxes. In the eQuad and hQuad, there are four overrunning clutch functions (one per rotor) which drive the rotors which drive each of four rotors during normal operation. Interconnecting shafts and combiner gearboxes are used in OMI operation to transfer power to the affected rotor. Whereas the tQuad has two overrunning clutch functions, driving two combiner gearboxes and the rotor gearboxes via interconnecting shafts during normal operation. During OEI operation the power flows in a similar manner, with the exception that one engine is delivering power to the combiner and rotor gearboxes.

Note: Functions 4 and 5 layouts are identical to the eQuad and hQuad, but the FCS interconnectivity with ESCs/motors is no longer present.

6.4 Electric Quadrotor (without cross shafting)

The eQuad without cross-shafting FBD is shown in Figure 30. There are all five essential functions, described below.

- Function 1: Power plant. This function is identical to the eQuad, consisting of the high voltage battery network, HVDC power management and distribution, LVDC power, and Battery cooling.
- Function 2: Convert electrical energy to shaft torque. This function is identical to the eQuad, consisting of the ESCs, electric motors, associated motor/ESC cooling, FCS control signals and feedback sensors, and overrunning clutches.
- Function 3: Transmit torque to rotors. This function consists of the rotor gearboxes.
- Function 4: Actuation. This function consists of hydraulic actuators at each rotor and three hydraulic pumps driven by the independent rotor gearboxes.
- Function 5: Flight Control System. This function is identical to the eQuad, consisting of the flight control computing, inertial data, air data, rotor sensing, and collective pitch actuation sub-functions.

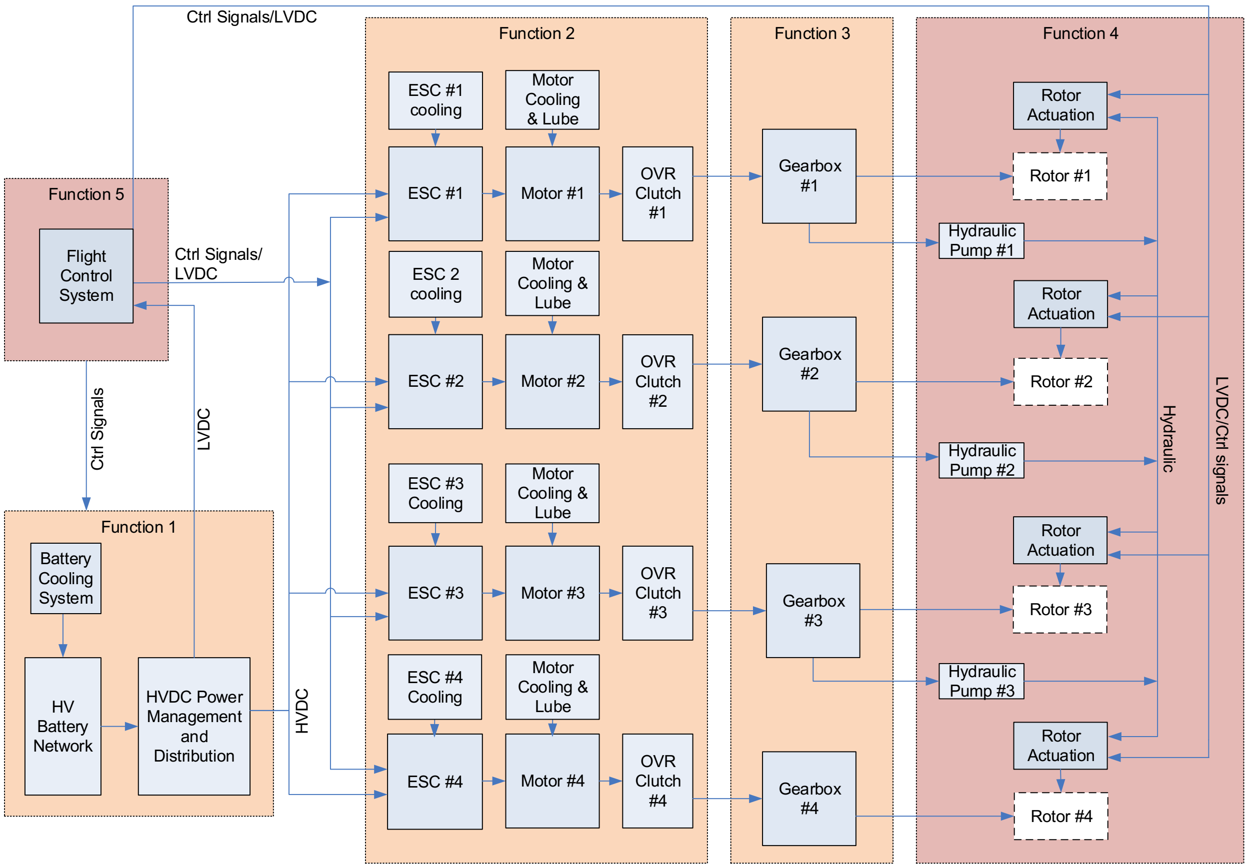


Figure 30: eQuad without Cross Shafting FBD

Relative to the other three quadrotor configuration, this variant without cross-shafting does not have combining gearboxes or interconnecting cross shafts. As such, the hydraulic pumps are driven by independent rotor gearboxes.

6.5 Collective-Controlled Hexarotor

The collective-controlled eHex, consists of all five essential functions, described below.

- Function 1: Power plant. This function is identical to the non-cross shafted eQuad, consisting of the HVDC battery network, HVDC Power Management and Distribution, LVDC power, and battery cooling.
- Function 2: Convert electrical energy to shaft torque. This function consists of the ESCs, electric motors, associated motor/ESC cooling, and FCS control signals and feedback sensors.
- Function 3: Transmit torque to rotors. This function is identical to the non-cross shafted eQuad, consisting of the rotor gearboxes.
- Function 4: Actuation. This function is identical to the non-cross shafted eQuad, consisting of hydraulic actuators at each rotor and three hydraulic pumps driven by the independent rotor gearboxes.

- Function 5: Flight Control System. This function is identical to the non-cross shafted eQuad, consisting of the flight control computing, inertial data, air data, rotor sensing, and collective pitch actuation sub-functions.

As detailed above, this configuration is nearly identical to eQuad without cross shafting, however, the overrunning clutch function was removed, as shown in Figure 31. The eQuad is assumed to need to power all four rotors for continued safe flight, while the guiding assumption for the eHex is that continued safe flight may be possible if one rotor is unpowered.

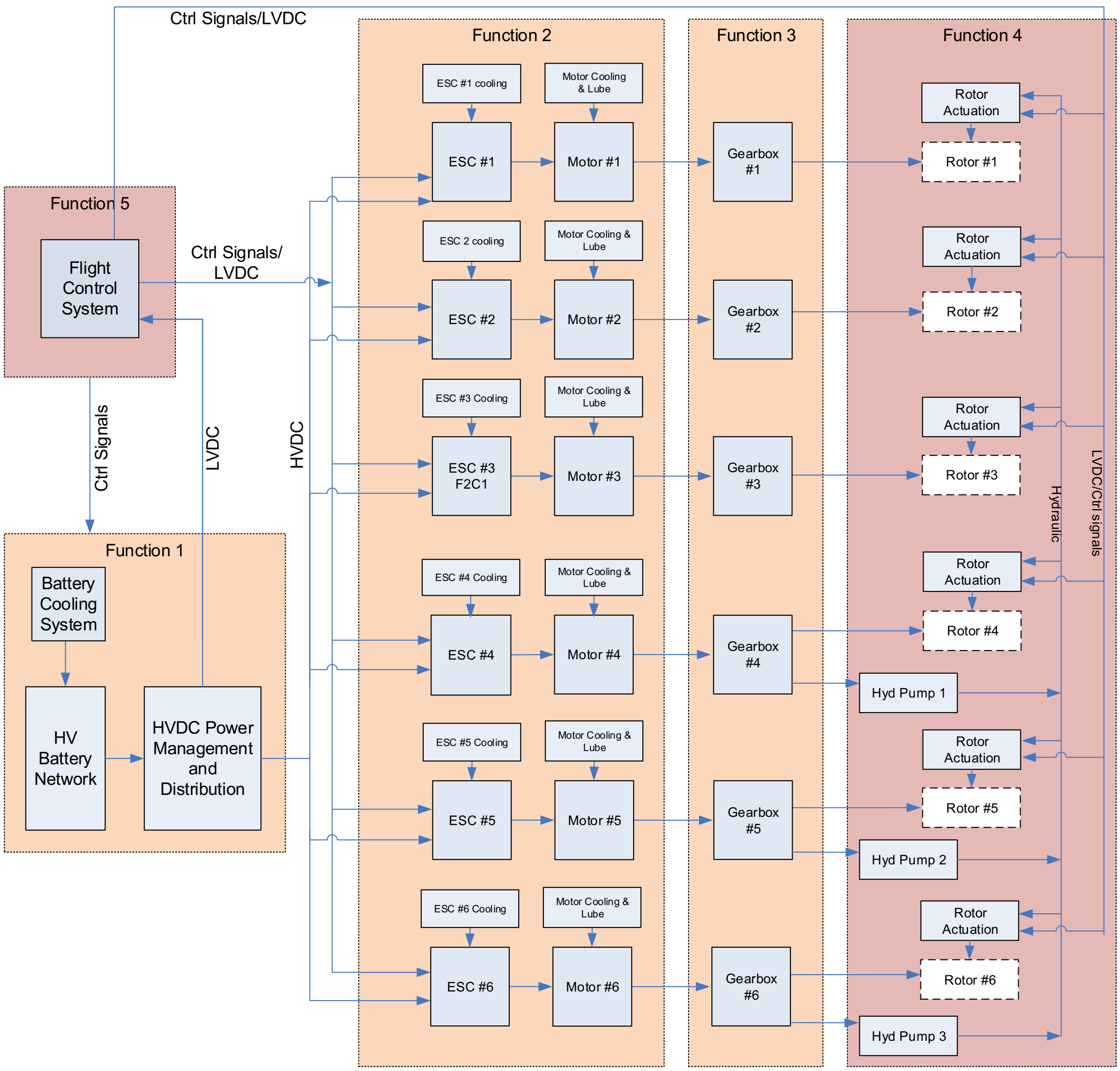


Figure 31: Pitch-Hex FBD

6.6 RPM-Controlled Hexarotor

The RPM-controlled eHex consists of four essential functions, described below.

- Function 1: Power plant. This function is identical to the collective-controlled eHex, consisting of the high voltage battery network, HVDC power management and distribution, LVDC power, and battery cooling.
- Function 2: Convert electrical energy to shaft torque. This function consists of the ESCs, electric motors, associated motor/ESC cooling, and FCS control signals and feedback sensors.
- Function 3: Transmit torque to rotors. This function is identical to the non-cross shafted eQuad, consisting of the rotor gearboxes.
- Function 5: Flight Control System (for RPM-Control). This function consists of the flight control computing, inertial data, air data, and rotor sensing sub-functions.

There is no actuation or actuator feedback position functions associated with the variable-speed, fixed-pitch configuration as shown in the FBD, see Figure 32.

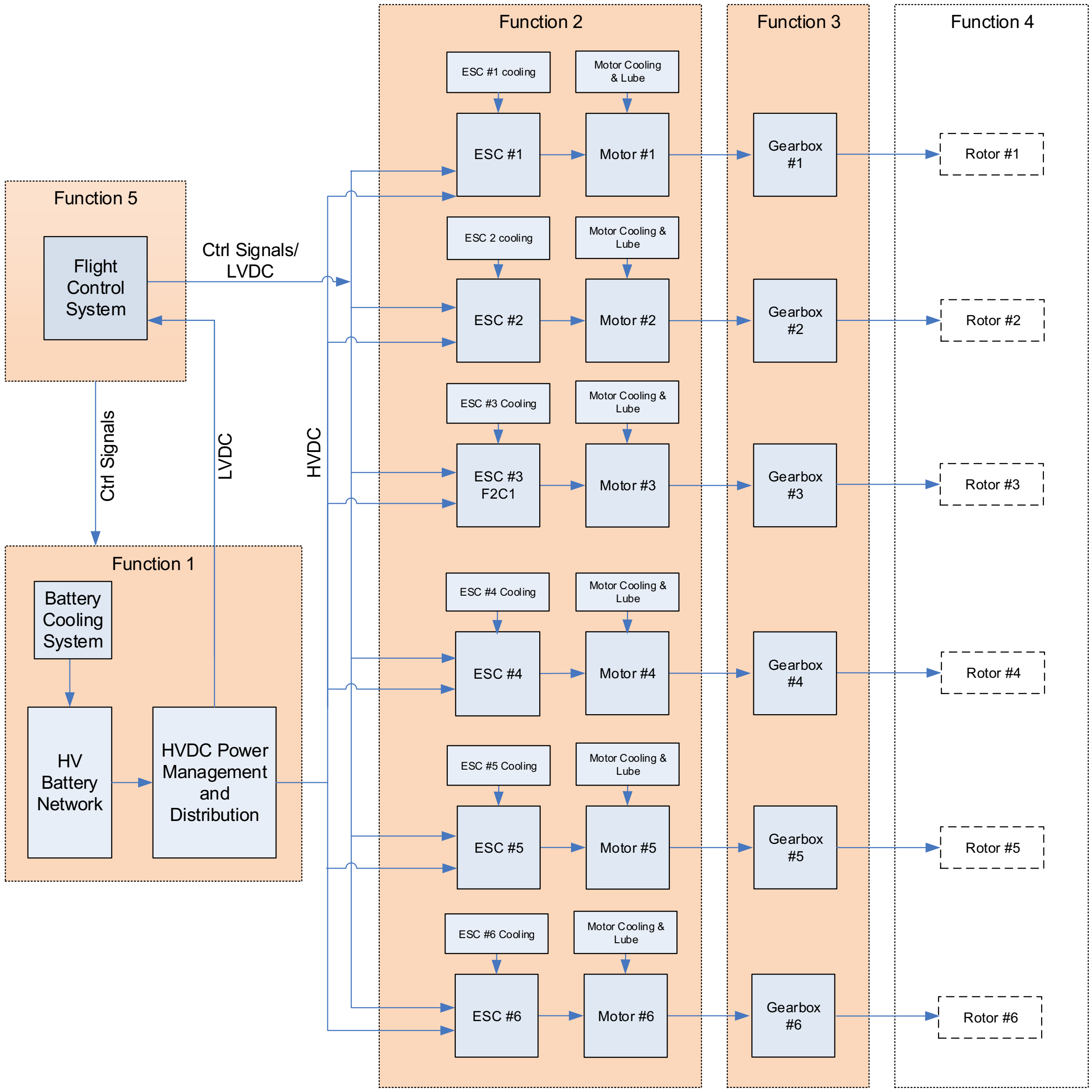


Figure 32: RPM-Hex FBD

6.7 RPM-Controlled Octorotor

The RPM-controlled eOct is identical to RPM-controlled hexarotor with two additional rotors. The design assumptions for the RPM-controlled eOct are assumed to be identical to the RPM-Controlled eHex.

7 FUNCTIONAL HAZARD ANALYSIS

The following sub-sections summarize the FHA main assumptions required for each configuration and key differences between designs. The safety analysis process for this study began with a FHA and lessons learned from prior work (ref. 5). The FHA is a structured analysis technique which systematically analyzes hazards arising from functional failures of a system. The FHA is used to evaluate the functions and corresponding failure conditions and severity classifications. A FHA considers the functions of the system under analysis, and identifies the failure conditions (hazards) by considering the effects of loss of the function, incorrect operation of the function, the ability to detect loss of function, or inadvertent occurrence of the function when not desired. The FHA typically considers all flight phases of the aircraft, as well as different operating environmental conditions, and how they affect functional failure severity. The phases of flight discussed in Section 5, were considered during the safety analysis. An associated FTA will be used in conjunction with the FHA in order to begin defining and allocating safety requirements to sub-systems in alignment with SC-VTOL-01 objectives. The FHA and the safety assessment will typically expand and evolve alongside the subsystem development. The FHAs for each of the aircraft vehicle configuration may be found in Appendix B.

The result of the FHA is a list of functional failures with an assigned severity, which depends on the possible outcome of the failure. A severity of catastrophic, severe, major, or minor is assigned to each functional failure, in accordance with ARP4761. Another result of the FHA is a list of derived safety requirements (DSR) needed to help mitigate and control the resulting hazard of the functional failure. The DSRs are provided to the systems engineering requirements management organization, and flowed down into appropriate design specifications in accordance with the requirements management process.

Following the FHA, the resulting functional failures are consolidated into a more concise list of hazards and assumptions, which become derived system requirements. Each hazard is assigned a severity, in accordance with the severity of the functional failure(s) encompassed by the hazard, and is also assigned a hazard probability. Different techniques have been employed at the discretion of safety analysts, to determine the hazard probabilities for each hazard.

7.1 Quadrotors

There are four quadrotor configurations analyzed in this study. The following Table 7 shows the key assumptions required for the configurations, highlighting common and unique assumptions.

Table 7: Design Features and Assumptions Summary for Quadrotor FHAs

	Applicability to Configuration			
Design feature or Assumption	**eQuad**	**hQuad**	**tQuad**	**eQuad (no X-Shaft)**
Variable pitch, fixed speed control scheme.	X	X	X	X
Cross-shafts between each rotor for emergency conditions.	X	X	X	
Loss of single rotor function must be considered a catastrophic hazard in all flight modes.	X	X	X	X

	Applicability to Configuration			
Design feature or Assumption	**eQuad**	**hQuad**	**tQuad**	**eQuad (no X-Shaft)**
Overrunning clutch is required for continued power application to each rotor in the event of loss of propulsion.	X	X	X	X
Loss of a single propulsor must be considered a catastrophic hazard in all flight modes.				X
Loss of dual propulsors must be considered a catastrophic hazard in all flight modes.	X	X	X	X
ESCs are assumed to have a pre-programmed reference speed such that if there is a loss in flight control computer signals, the ESCs will default to a reference speed that is able to maintain flight control path.	X	X		X
HVDC battery network provides fail-safe electrical power for emergency conditions and ground operations only.		X		
At least on motor/engine per rotor.	X	X		X
At least two centrally located turboshaft engines.			X	
Motors/engines are physically isolated such that failure in one motor/engine would not damage the other.	X	X	X	X
Motors/engines are sized such that single motor/engine loss is not catastrophic.	X	X	X	X
Rotors do not intermesh so that loss of cross shafts does not result in catastrophic dephasing.	X	X	X	X
HVDC batteries operate in depletion-only mode and no in-flight recharging occurs.	X	X		X
HVDC batteries are within the nominal voltage window to assumed nominal output power.	X	X		X

Appendix B includes the complete FHA for the eQuad, hQuad, tQuad, and eQuad without cross-shafting. The key FHA changes from the eQuad for the quadrotor configurations are summarized:

- hQuad: Functional hazards associated with the power plant are related to single or dual propulsion source failures. Hazard severity decreases for several power plant hazards due to the fail-safe HVDC network.
- tQuad: Functional hazards associated with the power plant are related to single or dual engine failure. All hazards associated with conversion of electrical energy to shaft torque are eliminated by design. Moreover, a failure in the combiner transmission or interconnecting cross shafts increases in hazard severity to catastrophic, due to the interconnectivity in the tQuad design described above.

- eQuad without cross-shafting: Without the interconnecting cross-shafts, a significant number of hazards increase in severity from minor to catastrophic relative to the cross-shafted variants.

7.2 Hexarotors and Octorotor

There are two hexarotor and one octorotor configurations analyzed in this study. Table 8 shows the key assumptions required for the configurations, highlighting common and unique assumptions.

Table 8: Design Features and Assumptions Summary for Hexarotor and Octorotor FHAs

	Applicability to Configuration		
Design feature or Assumption	**Pitch Hex**	**RPM Hex**	**RPM Oct**
Variable pitch, fixed speed control scheme.	X		
Variable speed, fixed pitch control scheme.		X	X
Loss of single rotor function is a considered minor hazard in all flight modes.	X	X	X
Overrunning clutch is required for continued power application to each rotor in the event of loss of propulsion.			
Loss of single propulsor is considered minor, but loss of dual propulsors is catastrophic.	X	X	X
ESCs are assumed to have a pre-programmed reference speed such that if there is a loss in flight control computer signals, the ESCs will default to a reference speed that is able to maintain flight control path.	X		
Rotor system control capability manages rotor system speed to prevent overspeed and undesirable thrust in the event of loss of propulsion.	X		
At least on motor/engine per rotor.	X	X	X
At least two centrally located turboshaft engines.			
Motors/engines are physically isolated such that failure in one engine would not damage the other.	X	X	X
Motors/engines are sized such that single motor/engine loss is not catastrophic.	X	X	X
Rotors do not intermesh.	X	X	X
HVDC batteries operate in depletion-only mode and no in-flight recharging occurs.	X	X	X
HVDC batteries are within the nominal voltage window to assumed nominal output power.	X	X	X

Appendix B includes the complete FHA for the hexarotors. The FHAs for the hexarotor and octorotor configurations are summarized:

- Collective-controlled eHex: A significant number of functional hazards decrease in severity from catastrophic to minor from the eQuad without cross shafting.
- RPM-controlled eHex: Without the assumed control system that can manage rotor system speed to prevent overspeed and undesirable thrust associated with the collective-controlled eHex, several failure conditions were determined to have a potential overspeed outcome, resulting in catastrophic outcome rather than minor.
- RPM-controlled eOct: The design assumptions for the RPM-octorotor are identical to the RPM-controlled eHex; therefore, the RPM-controlled eHex FHA applies to the RPM-controlled eOct.

7.3 FHA Considerations in Future Work

7.3.1 Loss of Single Propulsor: Minor vs Major

The loss of a single propulsor was included in the FHA for each of the vehicles considered. In the case of the quadrotors, the propulsors will fail, but the rotors will need to continue to function. In the case of the hexarotors and octorotor, the propulsors will fail and the appropriate rotors are able to be shut down.

With the current level of analysis, loss of a single propulsor for the study aircraft is deemed “minor" severity rather than “major”. This failure condition is “minor” under the assumption that there is low impact to handling qualities, with the ability to fly to the point of intended destination. This failure condition would increase to “major” severity if deemed to have significant impact to handling qualities, or if there is a need to land as soon as possible at the nearest suitable spot.

While a “major” severity would be conservative, most propulsor failures in the Fault Tree models meet the criteria associated with “major” severity hazards (10^{-5}). Moreover, increasing the severity of the Loss of Single Propulsor from “minor” to “major” would have no impact on the current work, as the fault tree model captures catastrophic and sever hazards, excluding major and minor hazards. Therefore, changes in the subject hazards from “minor” to “major” does not change the outcome in the PSSA, Section 13. Additional analysis would be required if the loss of a single propulsor is determined to result in a “severe” outcome, as the fault tree diagrams must model all “catastrophic” and “severe” hazards. Future work should evaluate pilot reactions and workload after a single propulsor fail to further develop and refine the severity classification for loss of a single propulsor.

7.3.2 Loss of Dual Propulsors Hazard Effect and Severity

Reduction in risk for the eOct configuration may be possible if combinations of dual propulsor failures can occur without catastrophic effect. This must be demonstrated via simulations and modeling to accommodate changes to the assumptions, the FHA, and FTA. Future work is required to reduce the severity of dual propulsor failures from “catastrophic” to “severe,” “major,” or “minor.” However, it is unclear if the additional weight of the added propulsors and relative increase of component power ratings will be an effective means to obtain adequate levels of safety.

7.3.3 Fire Hazards

Fire hazards were not developed under the scope of this effort as the focus was to develop DPFC architectures to meet the probabilistic failure criteria of VTOL.2510(a). Future work should further investigate the potential for fire hazard associated with a motor jam. This is expected to be more pronounced on the RPM-controlled aircraft because initial DPFC architectures directly connect the rotation of the rotor to the motor and the rotor is not controlled after the PMSM system is deenergized. The collective-controlled eHex maintains the ability to feather and manage rotor, and therefore motor, speed after the PMSM has been deenergized. The quadrotor configurations also separate the motion of the rotor and motor through collective-control and, additionally, through overrunning clutches. The overrunning clutches providing another layer of separation, allowing the affected rotor to continue to rotate separate from the deenergized motor.

8 FLIGHT CONTROL SYSTEM

High-level FCS schematics are developed for the quadrotor, hexarotor, and octorotor configurations in order to identify critical FCS sub-functions and interfaces for each vehicle. System redundancy in these schematics is intended to achieve reliability requirements allocated to the flight control system. A fly-by-wire FCS is common to all vehicle configurations, as is the assumption of on-board human piloted operation.

The FCS schematics share a common set of critical sub-functions, and the all-electric propulsion quadrotor serves as the baseline configuration given these common design features and assumptions. Cockpit controls and displays are excluded from the quantitative analysis in Section 13, as are network buses, rotor flapping sensors and radar/laser altimeters. These elements are common to conventional aircraft and therefore are not strictly necessary to include in the safety analysis of these novel urban air mobility vehicle configurations.

Collective-controlled vehicle configurations implement a triple hydraulic collective pitch actuation system, consistent with today's industry standard, due to the high reliability in comparison to electromechanical actuators (EMAs). The jamming probability of EMAs is much higher than that of hydraulic actuators. However, the feasibility of using EMAs is investigated particularly in the context of the collective-controlled hexarotor given the assumption that loss of actuator control of a single rotor is not catastrophic.

8.1 System Description and Analysis Approach

8.1.1 Common Critical FCS Sub-functions

The FCS layouts for all vehicle configurations share a common set of critical sub-functions. The air data function encompasses air data units and associated pitot-static instruments to provide critical air data such as airspeed, barometric altitude, and angle of attack to the flight control computers. Inertial navigation units provide translational and rotational positions, velocities, and accelerations to the flight control computers. The computing sub-function is comprised of the flight control computers themselves. The inertial data, air data, and computing sub-functions are powered via the low voltage DC bus as described in Section 11. Each of the rotor shafts is integral to the drive and power system as described in Section 9.

The remaining critical sub-functions of collective pitch actuation, hydraulic power, and rotor motor control are described and tailored for each vehicle configuration in subsequent subsections.

A triple channel FCS is used to achieve aircraft-level reliability requirements. Given this multi-channel architecture, one flight control computer communicates directly with one inertial navigation unit and one air data unit. The redundant flight control computers (FCCs) communicate via cross channel data links to support signal selection and actuator force equalization. The triplex FCCs, inertial navigation units, air data units, and pitot-static probes are implemented in each vehicle configuration.

8.1.2 Collective Pitch Actuation Hydraulic Power and Actuation System

A hydraulic actuation scheme is common to all collective-controlled vehicle configurations. Figure 33 shows the hydraulic power and actuation system for the baseline all-electric quadrotor. The three FCS channels and three corresponding hydraulic power systems are represented by red, blue, and green in the figure.

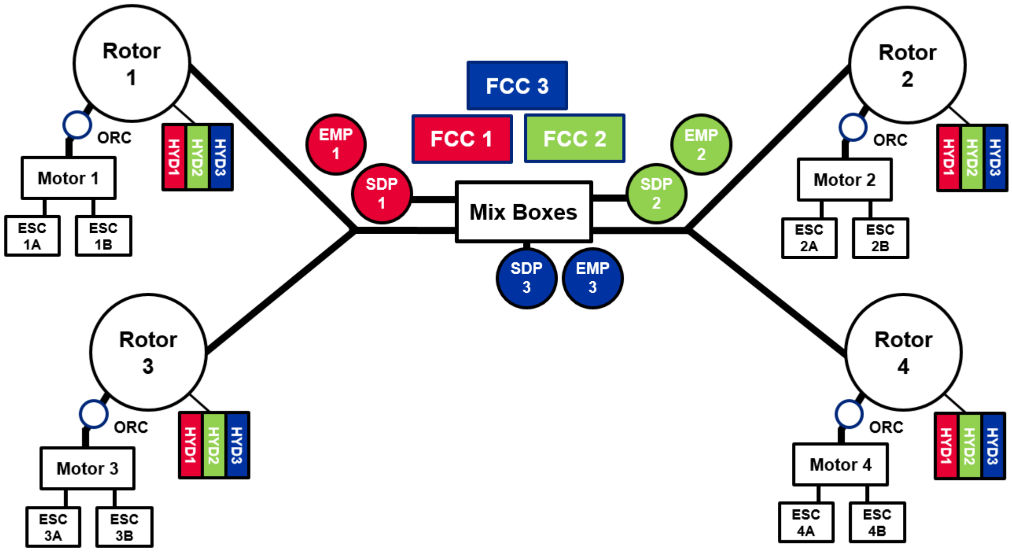


Figure 33: Baseline hydraulic power and actuation system for the all-electric quadrotor.

The drive system mix boxes described in Section 9 drive three shaft-driven hydraulic pumps (SDPs) which in turn power the three swashplate hydraulic actuators at each rotor. Three small electric motor pumps (EMPs) drive the actuators for ground checkout prior to rotors turning. Each of the three FCCs provides control of one hydraulic power stage (cylinder) per pitch control actuator in this triple-channel architecture.

The triplex FCCs and hydraulic system result in a Fail Operative, Fail Safe (FOFS) architecture for the hydraulic control and power functions. The hydraulic actuators remain jam critical. However, based on the current industry standard of single point jam hydraulic actuators used for helicopter main rotor and tail rotor control, hydraulic actuator jams have been considered extremely improbable and therefore this hydraulic actuation system serves as the baseline actuation configuration for the quadrotor and collective-controlled hexarotor configurations.

Figure 34 shows the hydraulic system and interfaces in more detail.

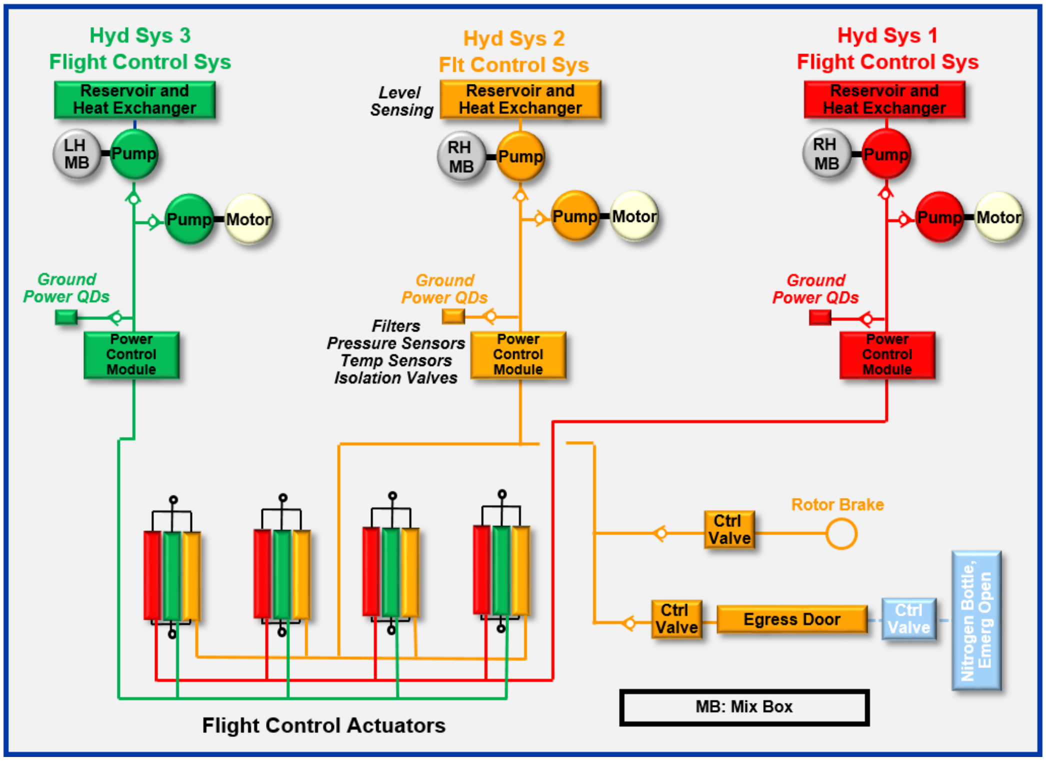


Figure 34: eQuad hydraulic power system and interfaces.

Each system contains a hydraulic reservoir and heat exchanger as well as a shaft-driven and electric motor pump described above. Ground power quick disconnects are provided for aircraft maintenance. The power control modules include filters, pressure and temperature sensors, and isolation valves for each system. Each hydraulic system provides power for one hydraulic power stage (cylinder) per rotor. Hydraulic power is also used for rotor braking and egress door functionalities via hydraulic control valves. Finally, the egress door is connected to a pneumatic backup for emergency operation.

8.1.3 All-Electric and Hybrid Quadrotor

The FCS schematic for the baseline all-electric and hybrid propulsion quadrotors is shown in Figure 35. Each single electric motor is controlled by dual Electronic Speed Controllers (ESCs) which accept both high voltage (to power the motor) and low voltage (to power the controller itself) sources. The rotor sensing sub-function encompasses RPM and torque feedback capability to the FCCs and ESCs. The RPM feedback from each rotor is provided to the two associated ESCs for RPM servo loop closure (speed governing) as described in Section 11. Independent RPM feedback from each rotor is also provided to the three FCCs to provide independent computational hardware and software monitoring of the ESC control functions. In the event the ESCs do not detect a failure and properly isolate themselves and/or the affiliated rotor motor, two out of the three FCCs can remove all power from the failed rotor. In addition, the rotor sensing sub-function encompasses torque feedback to the FCCs in order to perform non-critical loads limiting and other flight control functions. Collective blade pitch of the rotors is controlled using collective swashplates actuated by triple stage hydraulic actuators described in the previous section. Control signals from the FCCs are routed to the hydraulic actuators and the LVDC bus powers the FCCs and the hydraulic servo actuators.

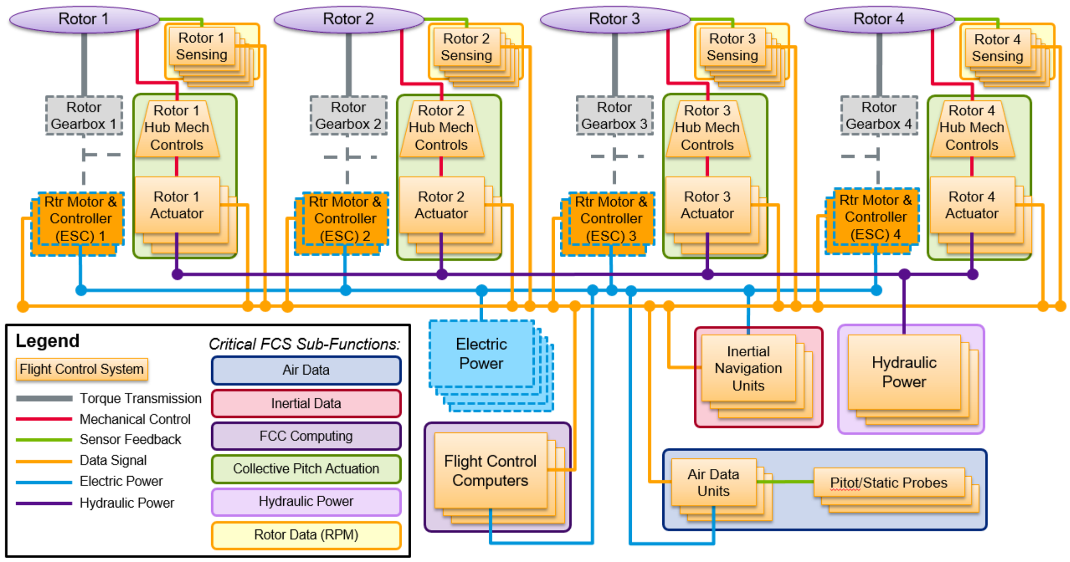


Figure 35: All-electric and hybrid quadrotor flight control system schematic. Dashed-line boxes indicate placeholders for non-FCS elements addressed in appropriate subsystem sections.

8.1.4 Turboshaft Quadrotor

The turboshaft quadrotor FCS schematic is shown in Figure 36. The primary difference with respect to the all-electric quadrotor baseline is the replacement of four electric rotor motors and eight ESCs with two turboshaft engines and associated full authority digital engine controllers (FADECs). This configuration is similar to how engine power is typically transmitted to conventional helicopter rotors. The turboshaft engines are controlled by FADECs, which are commanded by the FCCs using an integrated torque and thrust management approach in alignment with conventional flight control strategies. Subsequent design efforts would be required to determine detailed rotor sensing redundancy to regulate rotor speed and engine torques in both nominal and failure conditions.

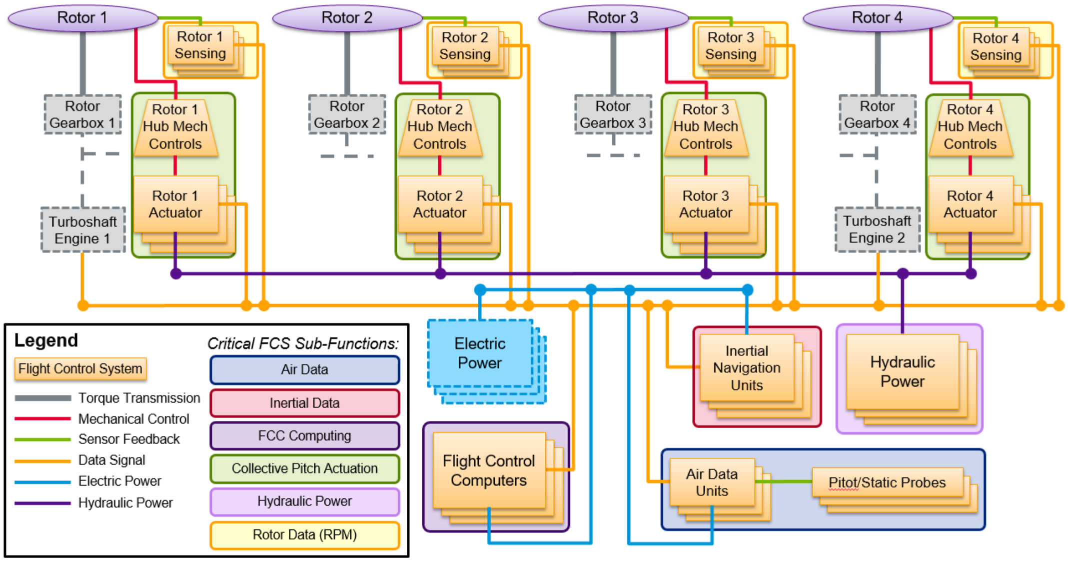


Figure 36: Turboshaft quadrotor flight control system schematic. Dashed-line boxes indicate placeholders for non-FCS elements addressed in appropriate subsystem sections.

8.1.5 *Collective-Controlled Hexarotor*

The FCS schematic for the collective-controlled hexarotor is shown in Figure 37, and differs from the all-electric quadrotor in the additional motor controllers, actuators, and rotor sensors associated with the two additional rotors. The critical FCS sub-functions of air data, inertial data, hydraulic power, pitch actuation and rotor sensing remain the same.

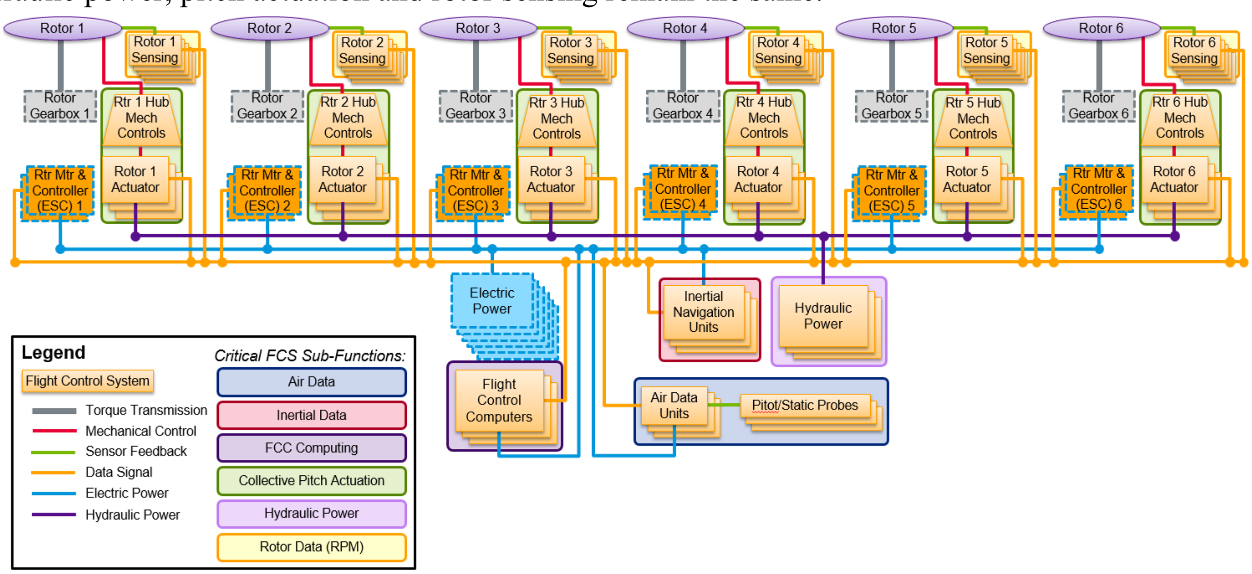


Figure 37: Collective-controlled hexarotor flight control system schematic. Dashed-line boxes indicate placeholders for non-FCS elements addressed in appropriate subsystem sections.

8.1.6 *RPM-Controlled Hexarotor*

The RPM-controlled hexarotor FCS schematic is shown in Figure 38. With respect to the collective-controlled hexarotor, the primary difference is related to the removal of mechanical hub pitch controls, pitch control servo actuators, and hydraulic power system associated with the collective pitch actuation sub-function. The ESCs are controlled by the FCCs in order to provide thrust control via RPM. The result is an overall system part count simplification from an FCS standpoint, though as discussed in Section 5, the need for dynamic rotor/motor control and response is increased to overcome rotor inertia.

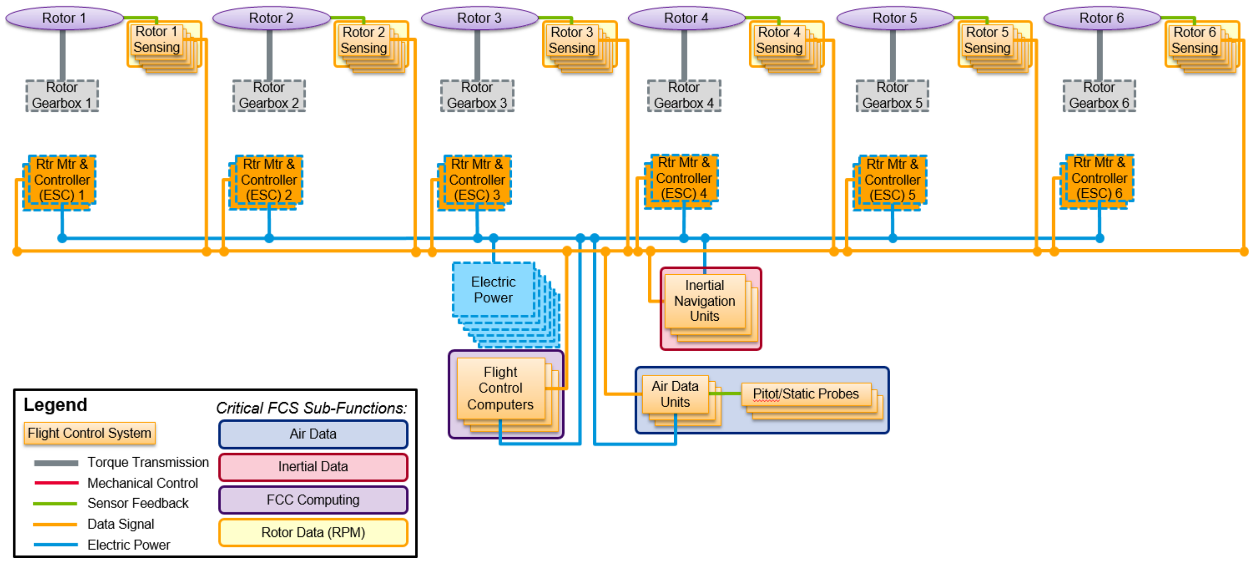


Figure 38: RPM-Controlled hexarotor flight control system schematic. Dashed-line boxes indicate placeholders for non-FCS elements addressed in appropriate subsystem sections.

8.1.7 RPM-Controlled Octorotor

The RPM-controlled octorotor FCS schematic is shown in Figure 39. With respect to the RPM-controlled hexarotor, the primary difference is the motor controllers, and rotor sensors associated with the two additional rotors.

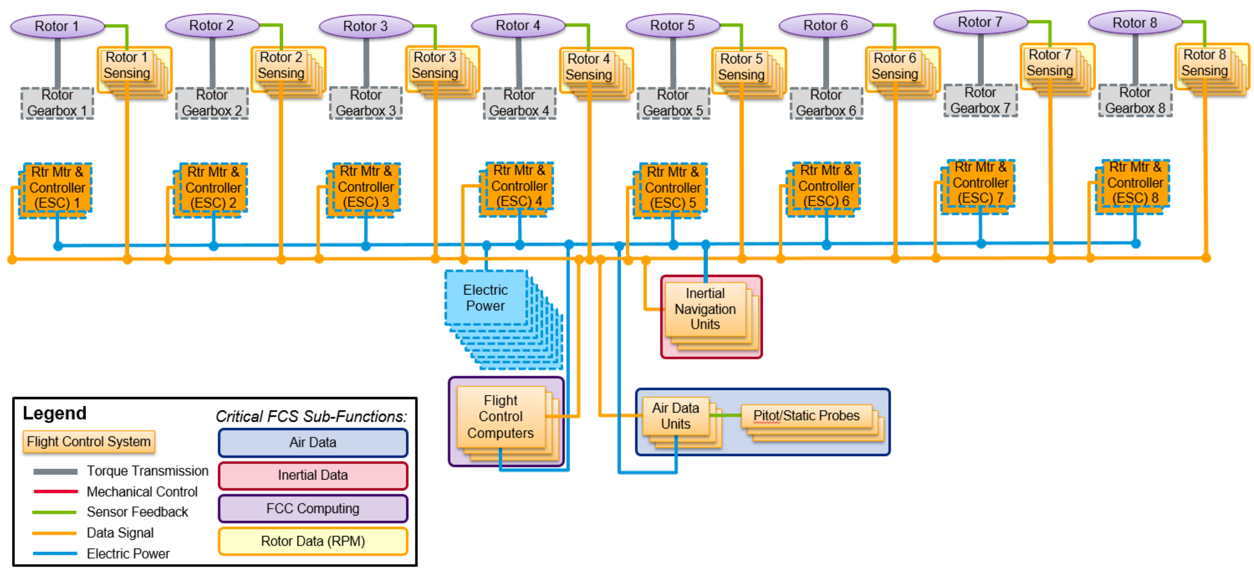


Figure 39: Octorotor flight control system schematic. Dashed-line boxes indicate placeholders for non-FCS elements addressed in appropriate subsystem sections.

8.2 Trade Studies

8.2.1 Electromechanical Actuation Architectures

The use of EMAs provides an opportunity to eliminate the hydraulic power system and affiliated logistics and maintenance issues. Due to the high mechanical jam and open failure rates of EMAs in comparison to hydraulic actuators, there is difficulty in achieving reliability requirements with EMA architectures. These architectures become more appealing as rotor redundancy increases, for example, if the collective-controlled hexarotor can be safely operated on less than six operating rotors.

Several single-screw EMA concepts are presented in order to investigate the effects of redundant control and drive path designs on actuator reliability. Figure 40(a) shows a schematic of a single channel configuration in which one FCC controls a motor, connected to the screw via a gearbox, and a single suite of sensors provides actuator position and motor speed feedback. Figure 40(b) shows a dual channel configuration with dual FCCs, controllers, and motors torque summed at the gearbox, as well as dual sensors for actuator position and motor speed feedback. Figure 40(c) shows a triple channel configuration with dual motors but triplex FCCs, controllers, resolvers, and position sensors. Finally, Figure 40(d) shows a triple channel configuration with three motors torque summed at the gearbox.

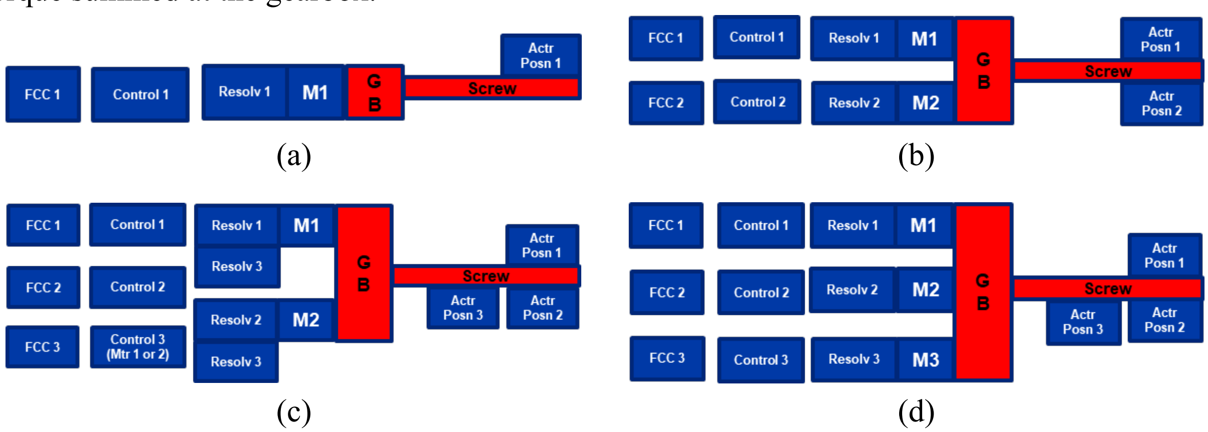


Figure 40: Single-screw EMA configurations.

The actuator reliability values for each configuration are summarized in Table 9.

Table 9: Actuator reliability comparison for EMA configurations shown in Figure 40.

Configuration	(a)	(b)	(c)	(d)
	Single FCC Single Controller Single Motor	Dual FCC Dual Controller Dual Motor	Triple FCC Triple Controller Dual Motor	Triple FCC Triple Controller Triple Motor
Actuator Reliability	5.98×10^{-4}	3.38×10^{-6}	3.02×10^{-6}	4.02×10^{-6}

Moving from a single to dual channel configuration increases reliability substantially, but the addition of a third FCC and controller has little improvement over the dual channel configuration due to the relatively high reliability of FCCs. The addition of a third motor reduces reliability as compared to the dual motor configurations, as the reliability of the actuator becomes primarily driven by the probability of mechanical jams and opens.

Future design work is still required to improve the reliability of EMA configurations sufficiently for use in the UAM concept vehicle configurations. However, when loss of control of a single rotor is not catastrophic (i.e. for the collective-controlled hexarotor), EMA configurations may be sufficient with additional design work.

8.3 Summary and Conclusions

Fly-by-wire FCS schematics are developed for all vehicle configurations with the assumption of on-board human piloted operation. The schematics include the required system redundancy incorporated in attempt to achieve reliability requirements, resulting in triplex FCCs, air data units, and inertial navigation units in all configurations, as well as redundant rotor sensing functions for RPM and torque feedback. From an FCS standpoint, moving from collective to RPM-controlled propulsion architectures results in an overall system part count simplification, but the need for dynamic motor/rotor control is significantly increased as shown in Section 5. Collective pitch control is critical in the quadrotor and collective-controlled hexarotor configurations, and therefore hydraulic pitch control is used due to low jam probability, consistent with today's industry standard. The high complexity of EMA configurations required to meet the reliability allocation for the quadrotor ultimately led to the triple hydraulic baseline. Less complex EMA configurations are more appealing as rotor count increases, particularly for the collective-controlled hexarotor configuration, where loss of control of a single rotor is not catastrophic. However, there is a need to further explore redundant EMA control and mechanical drive path designs in order to increase reliability, which comes at the expense of high complexity and increased weight.

9 DRIVE AND POWER SYSTEM

The following sections cover work related to the design and development for the drive and power system, component usage spectrum trends, and a discussion and potential design changes associated with VTOL.2250(c) single failure criteria. Derived requirements from S&C simulation activities, Section 5, the FHA, Section 7, initial FTA failure rate budgets, Section 6, and the statement of work were used to develop drive and power systems for each vehicle under study. Power spectrums, Section 4, and S&C simulations were evaluated for their impact on component usage, in order to characterize component usage trends as they pertain to vehicle parameters. VTOL.2250(c) single failure criteria is specifically discussed because of potential underlying design changes that have large impacts to the drive and power system design.

The drive and power system includes the necessary elements to generate and transmit torque to the rotors. Rotor shafts, applicable gearboxes, motors, inverters, generators, rectifiers, and gas turbines are included in the drive and power system. The rotor system and blades were not developed for this effort, as there are many fielded examples of fully articulated rotor systems that could be leveraged to support the PSSA, which was the primary focus of this effort. The ESC was split into two primary elements the inverter and the controller. The design authority for the inverter was kept within the drive and power system and the design authority for the motor controller was placed in the flight control system. The motor control was placed in the flight control system because of the direct coupling between the RPM-control schemes and the flight control system, allowing for fair comparison across study aircraft. The collective-control systems have a light coupling between motor control and the flight control system, so future work may find that the design authority for the motor controller best resides under the same authority as the inverter and motor, similar to today's engine and control interface, such as FADEC units. In the case of the hybrid-electric and turboshaft quadrotors, the associated electronic control of the turbomachinery resides in the drive and power system.

Drive and power systems were developed for the six vehicles under evaluation, reusing and adapting system architectures to minimize variables in the reliability and safety analysis. It is possible that systems could be further optimized for each configuration, but optimizing each system would create undesirable changes to the safety analysis results that would have to be further explained through subjective discussions. Concept sketches were refined from prior work (ref. 5) to show interconnections and general arrangement information and computer aided design (CAD) models were created for spatial integration and in support of initial component sizing.

9.1 Drive and Power System Description

The drive and power system was based on the single passenger eQuad from prior work (ref. 5). The eQuad drive and power system from prior work had to be modified to accommodate the larger, six passenger variant (ref. 4) and to create a modular architecture to reduce rework and minimize variables between study configurations. Architectures were developed for the (i) eQuad, (ii) hQuad, (iii) tQuad, (iv) eQuad without interconnecting shafts, (v) eHex, and (vi) eOct. The baseline eQuad architecture was modified to accommodate the various propulsion and control schemes. The hQuad is similar to the eQuad, except for the necessary turbogenerator to facilitate hybrid-electric operation. The tQuad is similar to the eQuad, as well, except that the four motors and motor gearboxes were replaced with two engines and engine nose gearboxes. The aircraft that do not utilize cross-shafts, namely the eQuad without cross-shafts, eHex, and eOct use a variant of the eQuad motors and motor and rotor gearboxes to facilitate power delivery. Control scheme variations were assumed to not affect the drive and power system architecture, therefore, the same drive and power system architectures are used regardless of control scheme changes.

The eQuad drive and power system is shown in Figure 41 and is comprised of permanent magnet synchronous motors (and associated inverters), rotor gearboxes with integral motor gearboxes and rotor shafts, pylon shafts, mix boxes, and intermediate shafts. A 15 degree flapping clearance was allocated for each rotor, requiring that the forward rotors were tilted forward nine degrees to allow >12 inches clearance between the pylon shaft centerline and the bottom of the rotor flapping envelope. A minimum of six inches was reserved for the pylon shafts and surrounding structures, plus an additional six inches, of clearance between the rotor flapping envelope and static structure. The aft rotors did not need adjustments to accommodate rotor flapping, but the aft rotors were tilted four degrees forward which is common tandem rotor design practice. Figure 42 illustrates the proposed shaft inclination angles for the quadrotor. The hexarotor and octorotor disc planes were not addressed under this effort, but future work should consider rotor boom design and flapping clearance for these configurations.

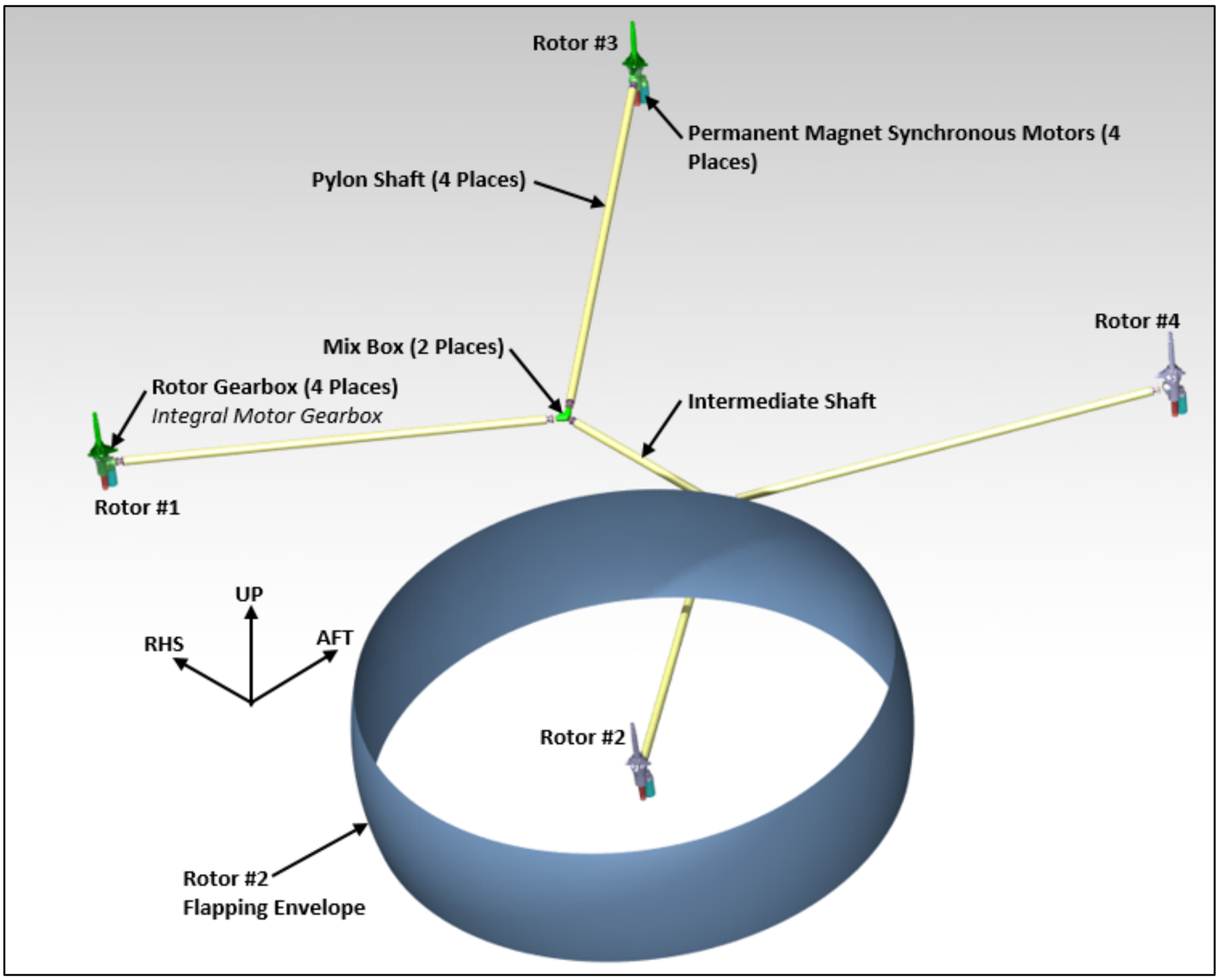


Figure 41: eQuad Drive and Power System

Table 10 shows the pertinent NDARC parameters and initial design parameters related to the drive and power system for the eQuad, hQuad, and tQuad. The power requirements for the eHex and eOct vary per rotor and are discussed in Section 9.1.4. The NDARC parameters come from the vehicle sizing routines, while the initial design parameters are the values used in the drive and power system layouts. ADS-50-PRF (ref. 25) was used to guide initial sizing criteria for the rotor gearboxes although future work should further reevaluate and refine proposed power ratings.

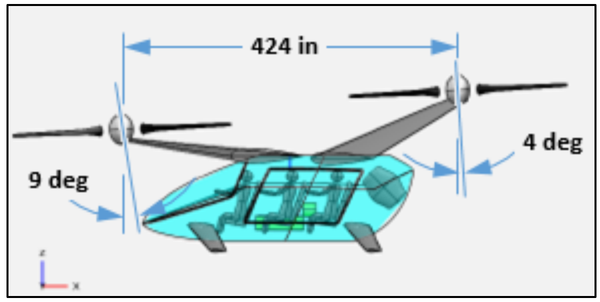


Figure 42: Illustration of Proposed Shaft Inclination Angles.

Table 10: NDARC and Initial Design Parameters

	NDARC Parameters				Initial Design Parameters			
Aircraft	Rotor Speed (RPM)	Motor Speed (RPM)	Motor Power (hp)	Gen. Power (hp)	Rotor Speed (RPM)	Motor Speed (RPM)	Motor Power (hp)	Gen. Power (hp)
eQuad	401	8,000	168	N/A	401	14,825	173	N/A
hQuad	487	8,000	160	709	487	17,924	173	704
tQuad	570	50,440	240	N/A	570	50,440	240	N/A

A 1.2 maneuver factor was applied to the rotor gearbox primary load path to account for maneuver loads similar to that of a tandem helicopter. By rearranging the tandem rotor gearbox rating method from ADS-50-PRF, the gearbox rating of each tandem rotor gearbox may be found to be equal to 120% of the engines uninstalled, SLS maximum continuous power (MCP) rating, times the number of engines, and divided by the number of lifting rotors. Similarly, the rating method for the quadrotor rotor gearbox is equal to 120% of the motors MCP rating, multiplied by the number of motors, and divided by the number of lifting rotors. Accordingly, the rotor gearboxes continuous rated power is 134 hp.

Emergency power requirements were also considered for the quadrotors using S&C models in absence of multirotor design guidance. Future work is required to understand emergency power conditions for the eHex and eOct because simulation results showed greater than two times MRP for those vehicle configurations, which needs to be reduced to minimize associated weight penalties. The worst case one motor inoperative (OMI) discrete gust case for the cross-shafted quadrotor was used to estimate emergency power requirements at each rotor. Figure 43 shows the power flow assumptions for the worst case OMI discrete gust case for Motor #2 failure. Simulation results show that 231 hp is required by the three, remaining Permanent Magnet Synchronous Motor (PMSM) systems, during a worst case, limit load condition. This represents the motor contingency rated power (CRP) requirement for a short duration burst. The motor gearboxes continuous power rating, commensurate with ADS-50-PRF, is 165 hp, or motor CRP divided by a 1.4 limit margin.

The power shown in Figure 43 transmitted through the gearbox and shaft systems represents the continuous power rating of the components sized for emergency use only, namely the spiral bevel gear sets, pylon shafts, and intermediate shaft. The continuous power rating for the spiral bevel gears and pylon shafts is 124 hp and was calculated by multiplying motor CRP by the number

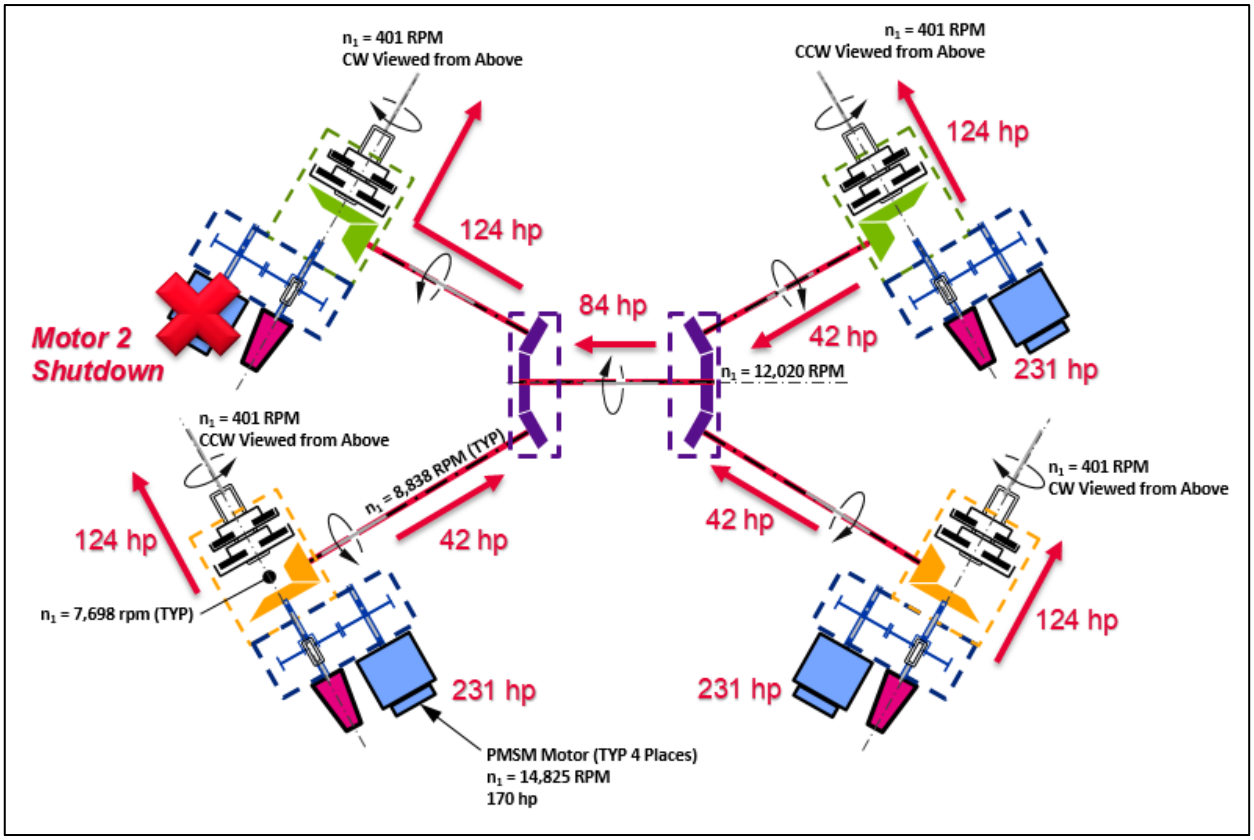


Figure 43: OMI Power Flow

of operating motors, dividing by the number of operating rotors, and dividing by a 1.4 limit margin. The continuous power rating for the intermediate shaft is 84 hp and was calculated by dividing the continuous power rating of the spiral bevel gears and pylon shafts by the number of operating motors and multiplying by two, to account for two pylon shafts transmitting power through the single, intermediate shaft.

9.1.1 eQuad Drive and Power System

The eQuad drive and power system, Figure 44, consists of four motors and associated inverters, four rotor gearboxes, two mix boxes, four pylon shaft assemblies with associated adapters, and one intermediate shaft assembly with associated adapters. Each motor and inverter is controlled by a motor controller; the design authority of which resides in the flight control system, see Section 8. The motors, inverters, and gearboxes require lubrication and cooling systems; the design authority of which resides in the thermal management system, see Section 10. Additionally, the drive and power system contains provisions to mount the actuators necessary for blade pitch control, see Section 8.

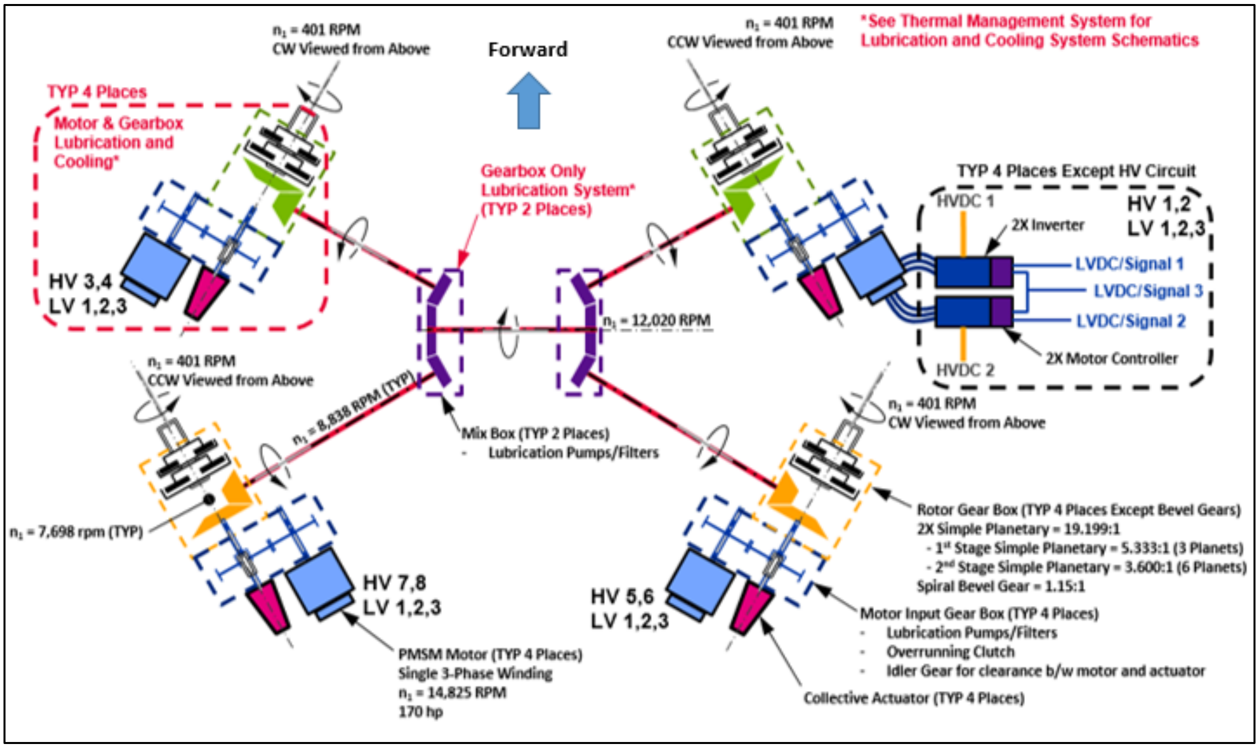


Figure 44: eQuad Drive and Power System Schematic

9.1.1.1 Permanent Magnet Synchronous Motor (PMSM) System

Each PMSM is a single winding motor, combined with a dual channel inverter and associated controllers. This motor is based on Safran's off-the-shelf ENGINeUS 126A1_455 Smart Motor, Figure 45. The 126A1_455 is designed to generate 169 hp continuously at 2,645-3,175 RPM and operate off a 600-850 volt direct current (VDC) source. The 126A1_455 is 14.76 inches diameter by 11.02 inches long. The concept motor designed for the eQuad is not commercially available and varies from the 126A1_455 in that it generates 173 hp continuously at 14,825 RPM. It operates using the same 600-850 VDC source. The stator is estimated to be 5.88 inches diameter by 5.29 inches long and the power electronics are estimated to require another 6.52 inches in length, making the overall motor package 5.88 inches diameter by 11.81 inches long, although detailed design and packaging studies are yet to be performed.

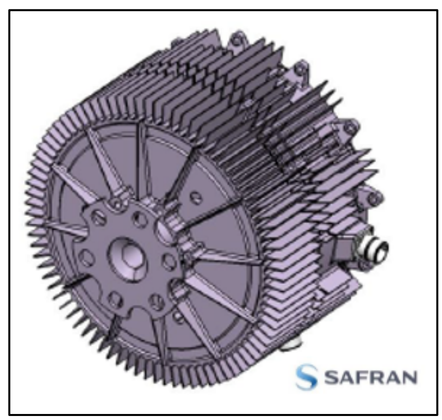


Figure 45: Safran ENGINeUS 100 Smart Motor

9.1.1.2 Motor Gearbox System

One motor mounts to the bottom of each of the four rotor gearboxes. The motors mounted to rotor gearbox #1 and #4 rotate counterclockwise when looking at the face of the motor and the motors mounted to rotor gearbox #2 and #3 rotate clockwise when looking at the face of the motor. Power travels from the motor into the rotor gearbox through an integral motor gearbox.

A spline connection takes power from the motor output shaft to the motor gearbox input pinion. The input pinion is a spur gear mounted on one ball and one cylindrical roller bearing. Power travels from the input pinion into an idler gear which is mounted on two internally mounted, cylindrical roller bearings. The idler gear being required to create the necessary radial distance between the motor and flight control actuator. The idler gear transfers power into the motor gearbox output gear. The output gear is mounted on two cylindrical roller bearings and houses two overrunning clutches and the associated clutch shaft and supporting bearings.

9.1.1.3 Overrunning Clutch System

The overrunning clutch system is located inside the output gear of the motor gearbox and transmits power from the output gear to the rotor gearbox separable gear shaft during normal operations. Two off-the-shelf Formsprag DC4127(3C)-N sprag overrunning clutches receive power from the output gear's integral outer race and transmit power to the clutch shaft's integral inner race. The clutch and clutch shaft are positioned within the output gear by an upper cylindrical roller bearing and lower Conrad ball bearing, both mounted between the output gear and clutch shaft. The clutch shaft also incorporates provisions for an upper cylindrical roller bearing and lower Conrad ball bearing to position the clutch shaft within the housing. The clutch shaft transmits power from the overrunning clutches to the rotor gearbox separable gear shaft via a working spline interface.

During emergency, OMI conditions the overrunning clutches will not allow torque in the main power loop to back-drive the affected motor. Instead, the overrunning clutch begins to freewheel, or overrun, and allow the clutch shaft to spin freely, while the output gear remains stationary, allowing the affected motor to be deenergenized without further degradation occurring.

9.1.1.4 Rotor Gearbox System

The rotor gearbox system, Figure 46, is comprised of a separable gear shaft, spiral bevel gear mesh, a two stage planetary system, and a rotor shaft. The spiral bevel pinion and gear are supported by bearing arrangements that allow for axial load in either direction. The bevel gear head is mounted to the separable gear shaft by a fixed spline and locknut and is supported by a duplex ball bearing and cylindrical roller bearing. The spiral bevel pinion is supported by two, preloaded angular contact bearings. The two stage planetary system consists of two sun in, ring fixed, carrier out simple planetary stages, with the planet gears supported by internally mounted spherical roller bearings. Under normal operating conditions the separable gear shaft receives power from the motor gearbox system via spline interface and sends power to the two stage planetary system.

During OMI conditions one of two different torque paths may occur. In one scenario, the separable gear shaft receives power from the motor gearbox, similar to normal operating conditions, but splits the power between the bevel gear head and the two stage planetary system. The spiral bevel gear then sends power through the spiral bevel pinion to the interconnecting shaft system to support the inoperative motor's rotor. In the other scenario, power is received from the interconnecting shaft system from the spiral bevel pinion and sent through the spiral bevel gear, into the separable gear shaft, and then into the two stage planetary system.

The rotor gearboxes are similar at all four rotors with the exception of the spiral bevel gear mesh to facilitate varying directions of rotation of each rotor. The spiral bevel gear mesh at rotors #1 and #2 are arranged so that the spiral bevel gear head is positioned above the spiral bevel pinion, or closer to the two stage planetary system. Rotors #3 and #4 are arranged so that the spiral bevel gear head is positioned below the spiral bevel pinion, or closer to the motor gearbox. Additionally, the baseline design includes spiral bevel gear meshes with opposing spiral angles at Rotor #1 and

#2 and Rotors #3 and #4, so that mesh forces push each gearhead out of mesh for the highest power, emergency condition.

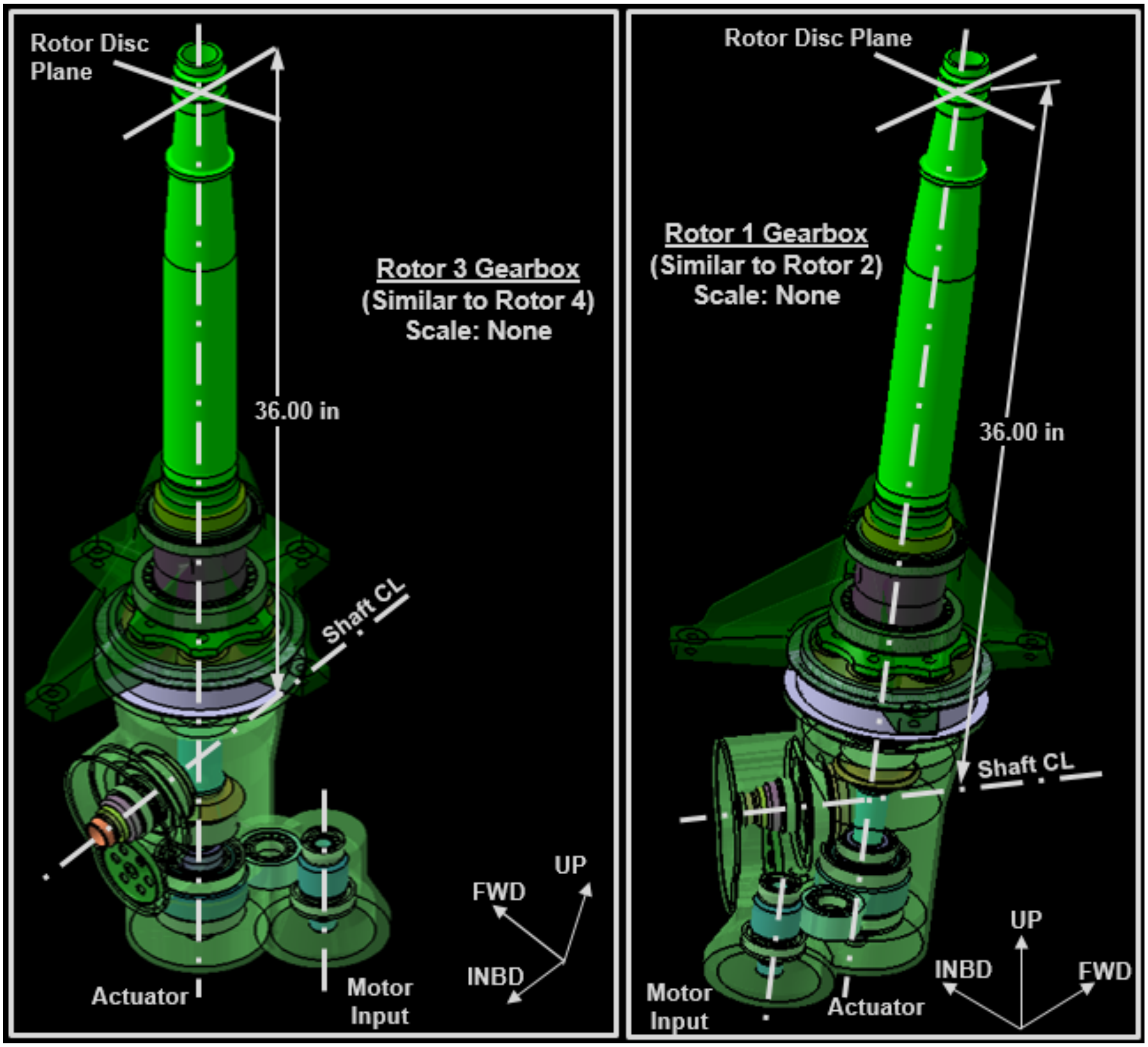


Figure 46: eQuad Rotor Gearbox System

9.1.1.5 Mix Box System

The mix box system, shown in Figure 47, is comprised of one spiral bevel pinion which mixes the power between two spiral bevel gears. The mix box is lightly loaded during normal operation, only supporting hydraulic system pump loads. The primary function of the mix box is to send power to the affected rotor during OMI conditions. To do so, the mix box is designed to (1) combine power from the two spiral bevel gear heads into the spiral bevel pinion and (2) transmit power from one spiral bevel gear and one spiral bevel pinion to the spiral bevel gear closest to the affected rotor.

The rotor centers and mix box were positioned relative to one-another so that the same spiral bevel gear can be utilized on either side of the pinion, reducing weight impacts of requiring two different pitch cones. Although the mix box was designed for two distinctly different power flows, large radial loads due to low shaft angles required two cylindrical roller bearings and one Gothic arch ball bearing for each of the spiral bevel gears and pinion.

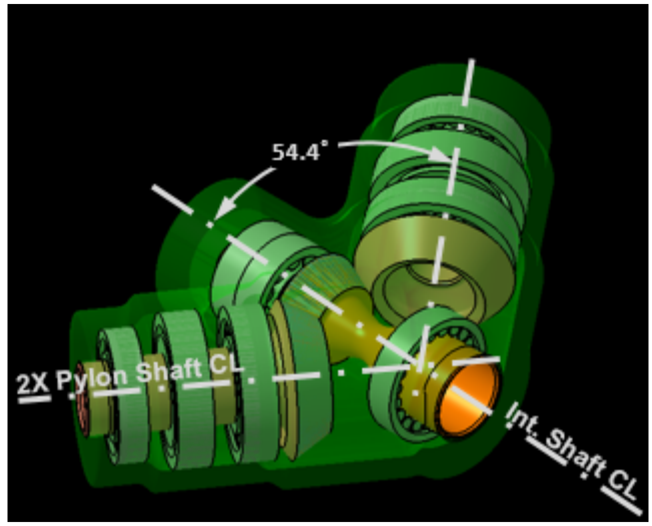


Figure 47: eQuad Mix Box System

9.1.1.6 Interconnecting Shaft System

The interconnecting shaft system is comprised of four super-critical pylon shafts and one super-critical intermediate shaft. The pylon shafts and intermediate shafts are similar to one-another, except that the pylon shafts are 248 inches long and the intermediate shaft is 112 inches long. The interconnecting shaft system was designed by Collins Aerospace and the main features of which are the bolt-on titanium diaphragm couplings with anti-flail mechanisms and the 5.00 inch outside diameter thermoplastic tubes. The thermoplastic tubes are permanently, mechanically fastened to titanium adapters, which allow for the diaphragm couplings to be bolted in place during installation on the aircraft. The intermediate shaft operates between the 1st and 2nd critical bending modes, and the pylon shafts operate between the 3rd and 4th critical bending modes.

9.1.1.7 Initial Gear Sizing

Initial gear sizing evaluated pitch line velocity, tooth bending stress, tooth contact stress, and flash temperature. More in depth analysis is required as designs mature, including mesh kinematics, profile, lubricant film thickness, and gear tooth profile geometry. Figure 48 shows the gear mesh location for the rotor and motor gearboxes. A summary of rough gear sizes for the motor gearbox, rotor gearbox, and mix box are presented in Table 11. Rough sizing was performed using methods that originate from gearing formula developed by Gleason Works and the American Gear Manufacturer's Association (AGMA).

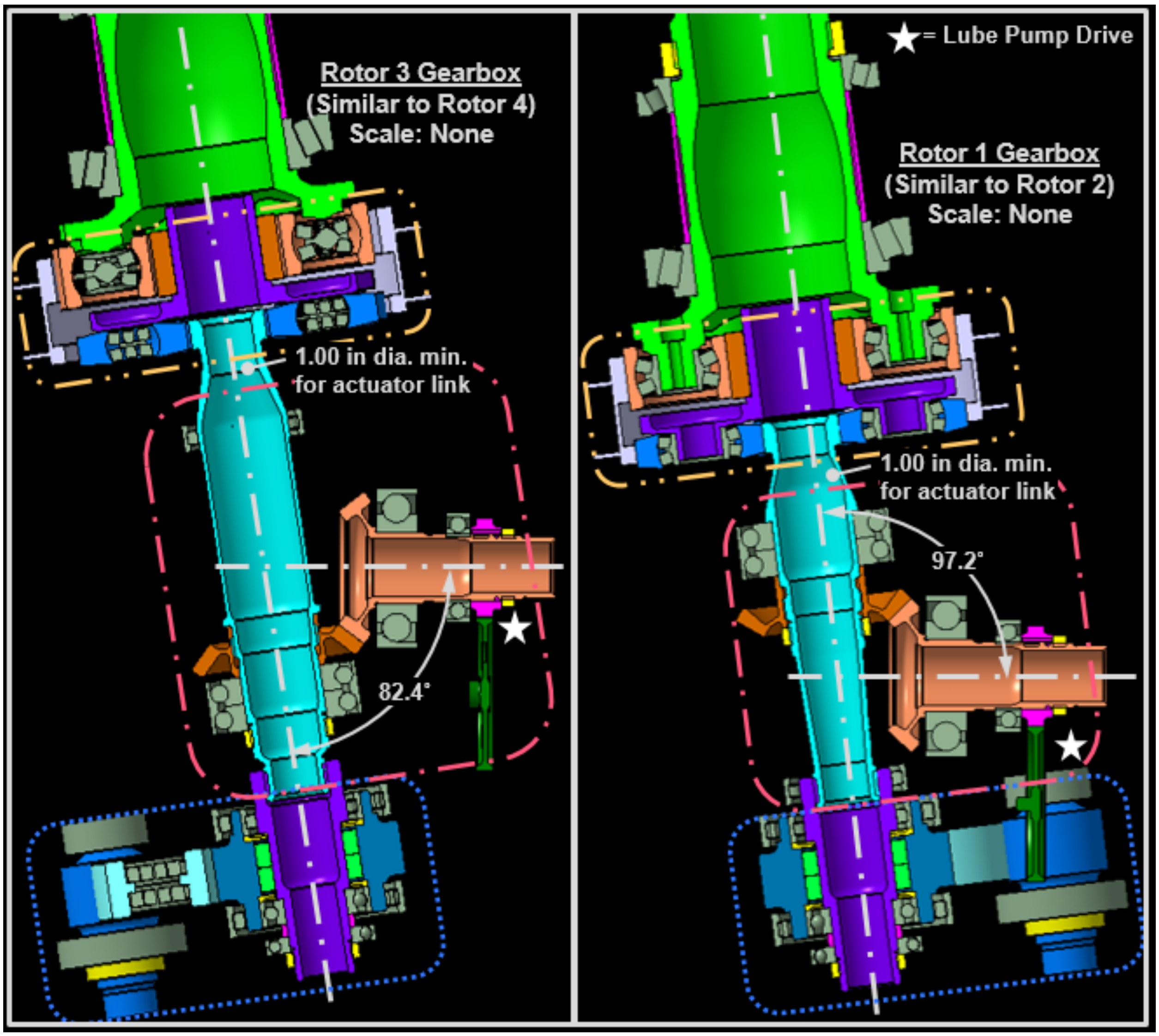


Figure 48: eQuad Gear Mesh Locations. Blue dashed line indicates motor gearbox mesh. Red dot-dashed line indicates rotor gearbox spiral bevel gear mesh. Yellow double-dot-dashed line indicates rotor gearbox two stage planetary system.

Table 11: eQuad Initial Gear Size Summary

Description	**Gear**	**# of Teeth**	**Pitch Dia. (in.)**	**Face Width (in)**	**Speed (RPM)**
Spur Gear Mesh Parallel Axis Gear Train Motor Gearbox	Pinion	27	2.1675	1.32	14,825
	Idler	35	2.8097	1.26	11,436
	Gear	52	4.2278	1.20	7,697.6
Spur Gear Mesh 1^{st} Stage Planetary System Rotor Gearbox	Sun	21	1.6776	0.63 (Sun/Planet)	6,254.3 (rel)
	Planet	35	2.7966		3,752.6 (rel)
	Ring	91	7.2708		1,443.3
Spur Gear Mesh 2^{nd} Stage Planetary System Rotor Gearbox	Sun	35	2.7960	1.37 (Sun/Planet)	1,303.0 (rel)
	Planet	28	2.2368		1,042.4 (rel)
	Ring	91	7.2695		401.00
Spiral Bevel Gear Mesh Shaft Takeoff Rotor Gearbox	Pinion	27	3.3691	0.73	7,697.6
	Gear	31	3.8682		8,838.0
Spiral Bevel Gear Mesh Mix Box	Pinion	25	2.1583	0.83	12,020
	Gear (2X)	34	2.9353		8,383.0

9.1.1.8 Initial Bearing Life Estimates

Romax Aero Nexus DT was used to assess bearing life. Each gear mesh was analyzed independently. The independent meshes evaluated were the motor gearbox spur gear mesh, the rotor gearbox spiral bevel gear mesh, first stage simple planetary system, and second stage simple planetary system, and the mix box spiral bevel gear mesh. The motor gearbox spur gear mesh includes the input pinion, idler, and output gear. The rotor gearbox spiral bevel gear mesh included independent analysis for each rotor, including variations in direction of spiral angle and shaft angles. The rotor gearbox planetary systems included the applicable sun, planet, and ring gears and planet posts and carriers. The mix box spiral bevel mesh was modeled as a single pinion and gear pair, omitting the second spiral bevel gear in mesh with the spiral bevel pinion.

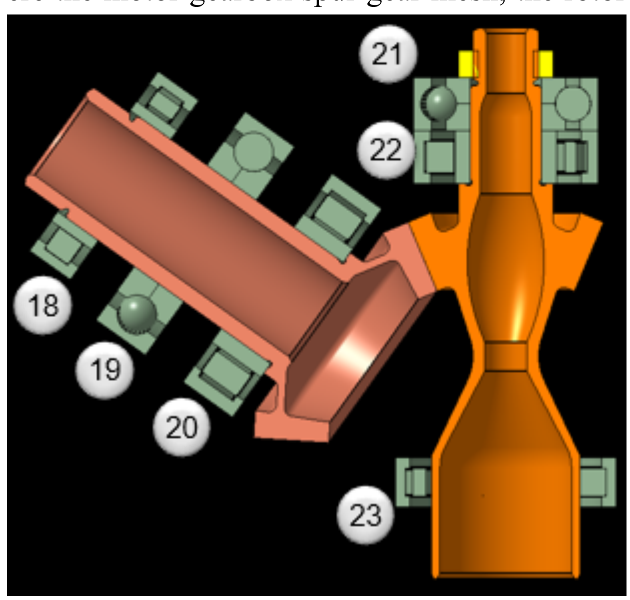


Figure 49: eQuad Mix Box Bearing Locations

Off-the-shelf bearings were used wherever practical. Figure 49 and Figure 50 show the bearing locations for the mix box and rotor gearbox systems, respectively. 100% bearing reuse within each rotor gearbox was designed into the system by rearranging the relative locations of each bearing and gear depending on spiral bevel gear head location. Romax's "Advanced" minimum B_{10} life for each bearing is shown in Table 12.

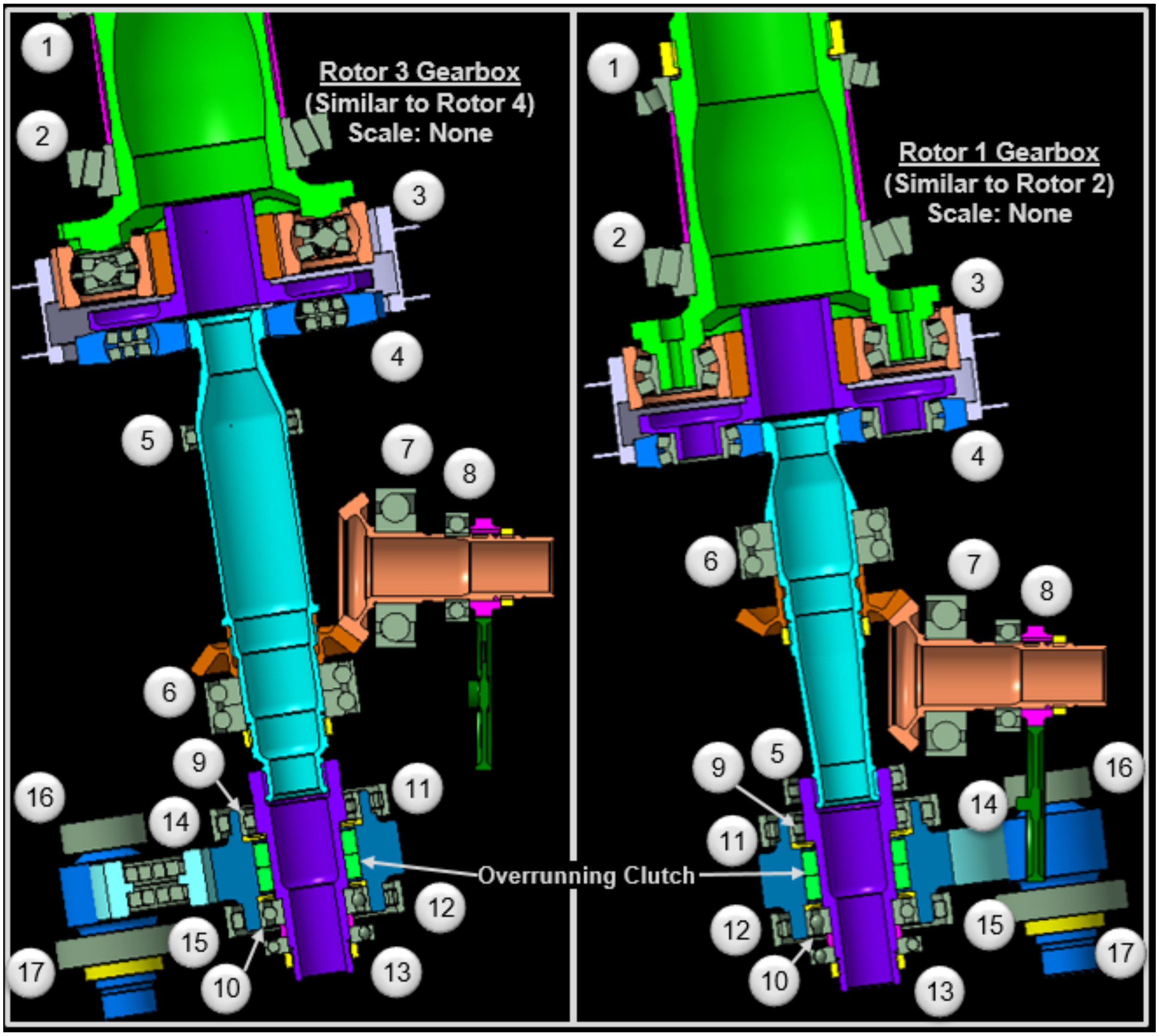


Figure 50: eQuad Rotor Gearbox Bearing Locations

Table 12: eQuad Bearing Summary

Bearing ID	Type	Shaft	Catalogue Part Number	Cubic Mean Power (hp)	B_{10} Life (hrs.)
1	Tapered Roller	Rotor	JL819349-JL819310	95	TBD
2	Tapered Roller	Rotor	42375A-42584	95	TBD
3	Spherical Roller	Planet 2	Custom	95	$>10^6$
4	Spherical Roller	Planet 1	22205E	95	11,010
5	Cylindrical Roller	SB Gear	NU1008	89	$>10^6$
6	Duplex Ball	SB Gear	3209E	89	14,971
7	Angular Contact	SB Pinion	7308BE	89	9,604
8	Angular Contact	SB Pinion	Custom	89	6,005
9	Cylindrical Roller	Clutch	NU1008	117	$>10^6$
10	Conrad Ball	Clutch	6007	117	$>10^6$
11	Cylindrical Roller	Spur Gear	NU1013	117	104,660
12	Cylindrical Roller	Spur Gear	NU1013	117	74,484
13	Conrad Ball	Spur Gear	16007	117	$>10^6$
14	Cylindrical Roller	Idler Gear	N1007EC	117	19,190
15	Cylindrical Roller	Idler Gear	N1007EC	117	19,620
16	Cylindrical Roller	Spur Pinion	NU2204EC	117	29,157
17	Conrad Ball	Spur Pinion	6207	117	17,070
18	Cylindrical Roller	Gear	NU1007EC	89	121,630
19	Split Inner Ring Ball	Gear	QJ207	89	$>10^6$
20	Cylindrical Roller	Gear	NU207EC	89	6,946
21	Split Inner Ring Ball	Pinion	QJ304	89	6,427
22	Cylindrical Roller	Pinion	NU304EC	89	6,926
23	Cylindrical Roller	Pinion	NU1007EC	89	$>10^6$
24	Cylindrical Roller	Gear	NU1007EC	89	121,630

9.1.2 hQuad Drive and Power System

The hQuad drive and power system, Figure 51, is similar to the eQuad drive and power system except for a turbogenerator which provides primary electrical power to the motors with a smaller battery network providing the power for emergency, turbogenerator failure conditions. Similar to the eQuad, the hQuad also includes four motors and associated inverters, four rotor gearboxes, two mix boxes, four pylon shaft assemblies with associated adapters, and one intermediate shaft assembly with associated adapters. Each motor and inverter is controlled by a motor controller; the design authority of which resides in the flight control system, see Section 8. The motors, inverters, and gearboxes require lubrication and cooling systems; the design authority of which resides in the thermal management system, see Section 10. Additionally, the drive and power system contains provisions to mount the actuators necessary for blade pitch control, see Section 8.

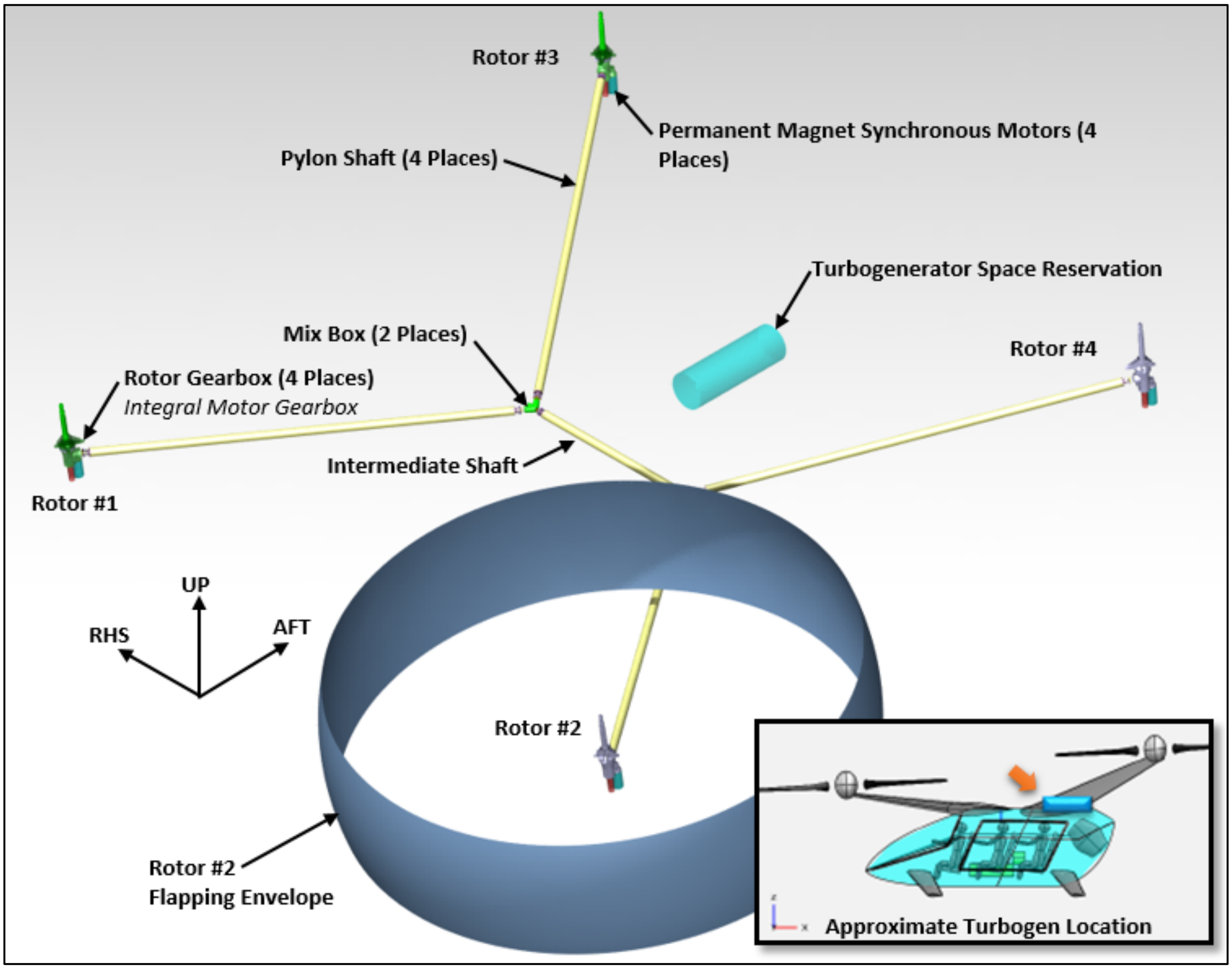


Figure 51: hQuad Drive and Power System

GE Aviation developed a concept turbogenerator system for the hQuad, based on their eFlex turbogenerator technology, shown in Figure 52. The turbogenerator is comprised of a turbine engine core mated to an alternating current (AC) induction generator. The AC generator is mounted to the engine exhaust and receives power directly from the power turbine shaft, at approximately 31,000 RPM, replacing the reduction gearbox of an off-the-shelf H85 turboprop engine. Mechanical power is converted to AC electrical power in the armature/stator and the integrated controller converts it to +/-270 volts direct current (DC) electrical power. The DC electrical power is sent to the electrical power and distribution system, see Section 11, where it will be sent to the four, remotely located motors.

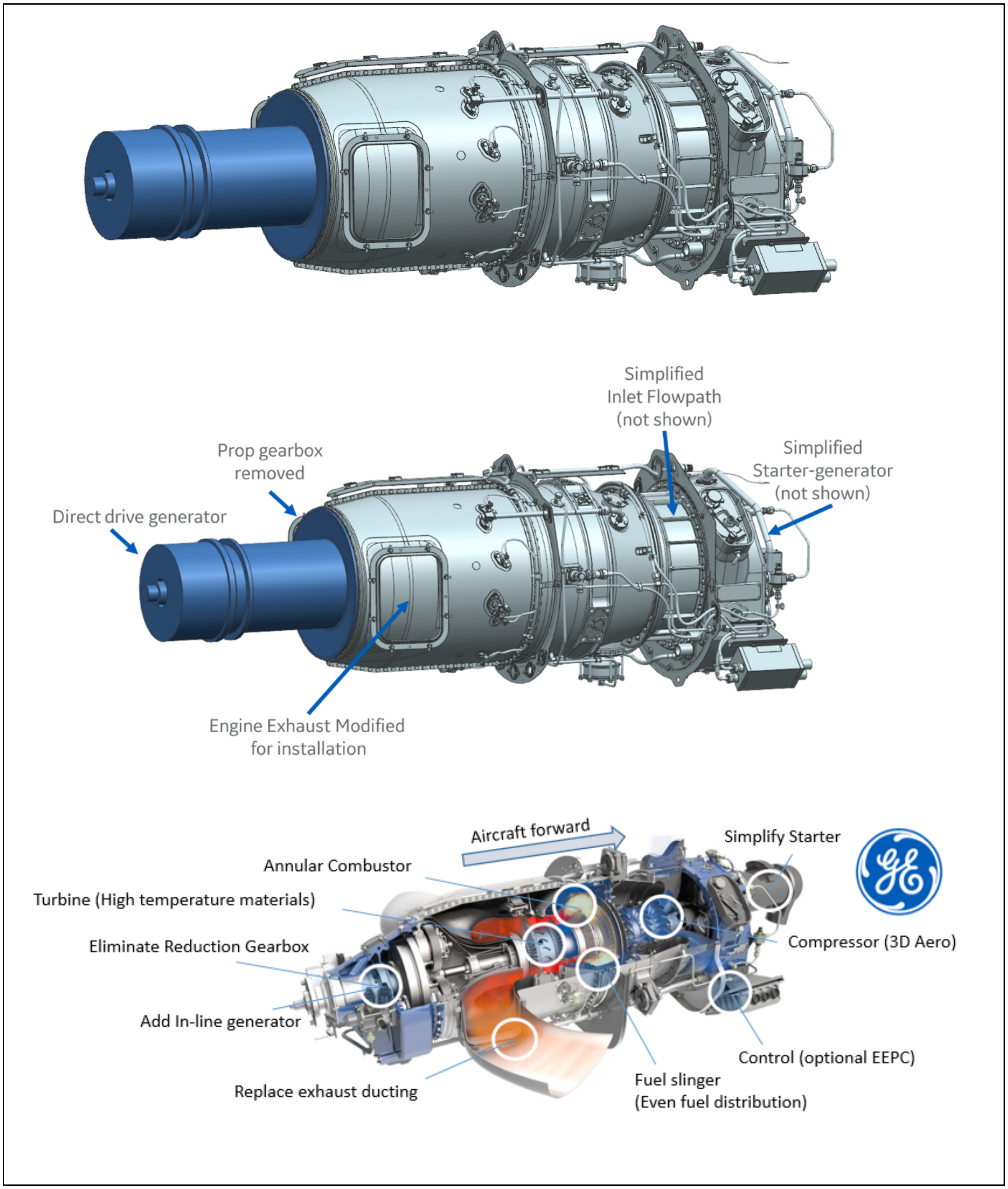


Figure 52: GE H85 eFlex Turbogenerator

9.1.3 tQuad Drive and Power System

The tQuad drive and power system, Figure 53, is similar to the eQuad drive and power system except that the four motor gearboxes are removed and two engine nose gearboxes receive power from two turboshaft engines. Similar to the eQuad and hQuad, the tQuad also includes four rotor gearboxes, two mix boxes, four pylon shaft assemblies with associated adapters, and one intermediate shaft assembly with associated adapters. The gearboxes require lubrication and cooling systems; the design authority of which resides in the thermal management system, see Section 10. Additionally, the drive and power system contains provisions to necessary to mount the actuators necessary for blade pitch control, see Section 9.

The turboshaft engine is based GE Aviation's small and reliable turboshaft engine, initial conceived for the US Army's Reliable Advanced Small Propulsion System (RASPS) program. GE's engine includes an inlet, compressor, heat exchanger/recuperator, and turbine, delivering power to the engine nose gearbox. The single stage centrifugal compressor discharges into a counter-flow heat exchanger (or recuperator). The exhaust recuperator recovers waste energy to preheat the compressor discharge air, reducing the amount of fuel needed to operate the engine. Air preheated in the recuperator flows into a simple single can combustor. Combustion air then enters a single stage axial turbine before re-entering the recuperator hot side.

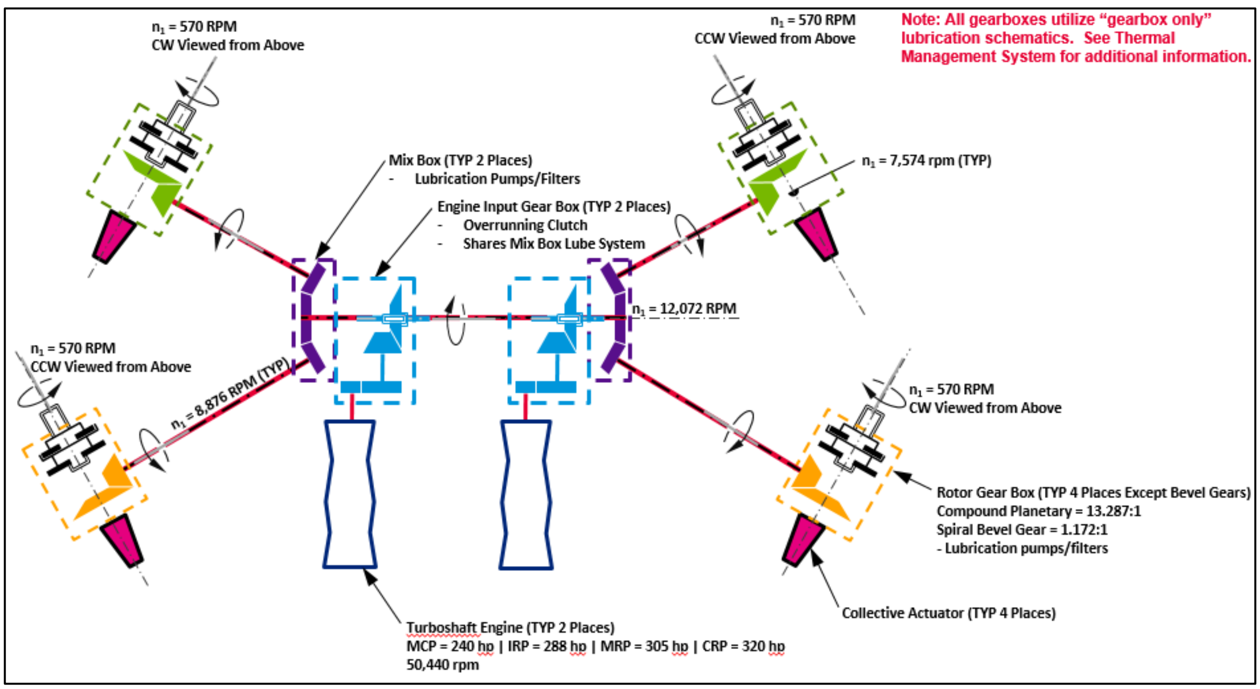


Figure 53: tQuad Drive and Power System Schematic

One engine nose gearbox, Figure 54, is mounted forward of each turboshaft engine and transmits power from the engine core to the intermediate shaft system and mix boxes. A spline connection takes power from the engine output shaft to the engine nose gearbox input pinion. The input pinion is a helical gear mounted on one ball and one cylindrical roller bearing. Power travels from the input pinion into a helical gear which transmits power into an integral lay shaft and spiral bevel pinion. The helical gear, lay shaft, and spiral bevel pinion are supported by two cylindrical roller bearings and one ball bearing. The spiral bevel pinion transmits power into a spiral bevel gear, mechanically fastened to the outer clutch shaft via fixed spline and locknut. The spiral bevel gear and outer clutch shaft are mounted on two cylindrical roller bearings and one ball bearing. The outer clutch shaft houses two overrunning clutches (Formsprag #DC4127(3C)-N) and the associated inner clutch shaft and supporting bearings. The inner clutch shaft sends power left and right into the intermediate shaft system and mix box.

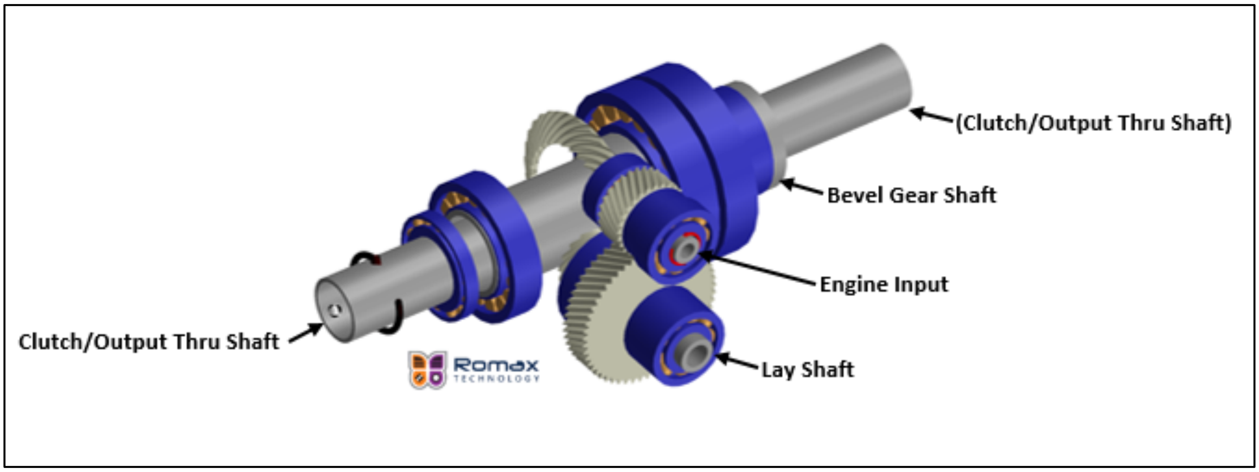


Figure 54: tQuad Engine Nose Gearbox

9.1.4 Drive and Power System – Quad, Hex, and Octorotors without Interconnecting Shafting

The drive and power system for the electric quadrotor, hexarotor, and octorotors without interconnecting shafting was largely simplified from the eQuad, hQuad, and tQuad configurations. Because the interconnecting shafting was not included in the vehicle configuration, the mix boxes, pylon shafts, and intermediate shafts were not required. Moreover, the spiral bevel gear mesh located inside each rotor gearbox was no longer required. Therefore, a modular rotor gearbox, Figure 55, could be developed that was largely similar to that of the eQuad, except that the spiral bevel gear mesh was removed.

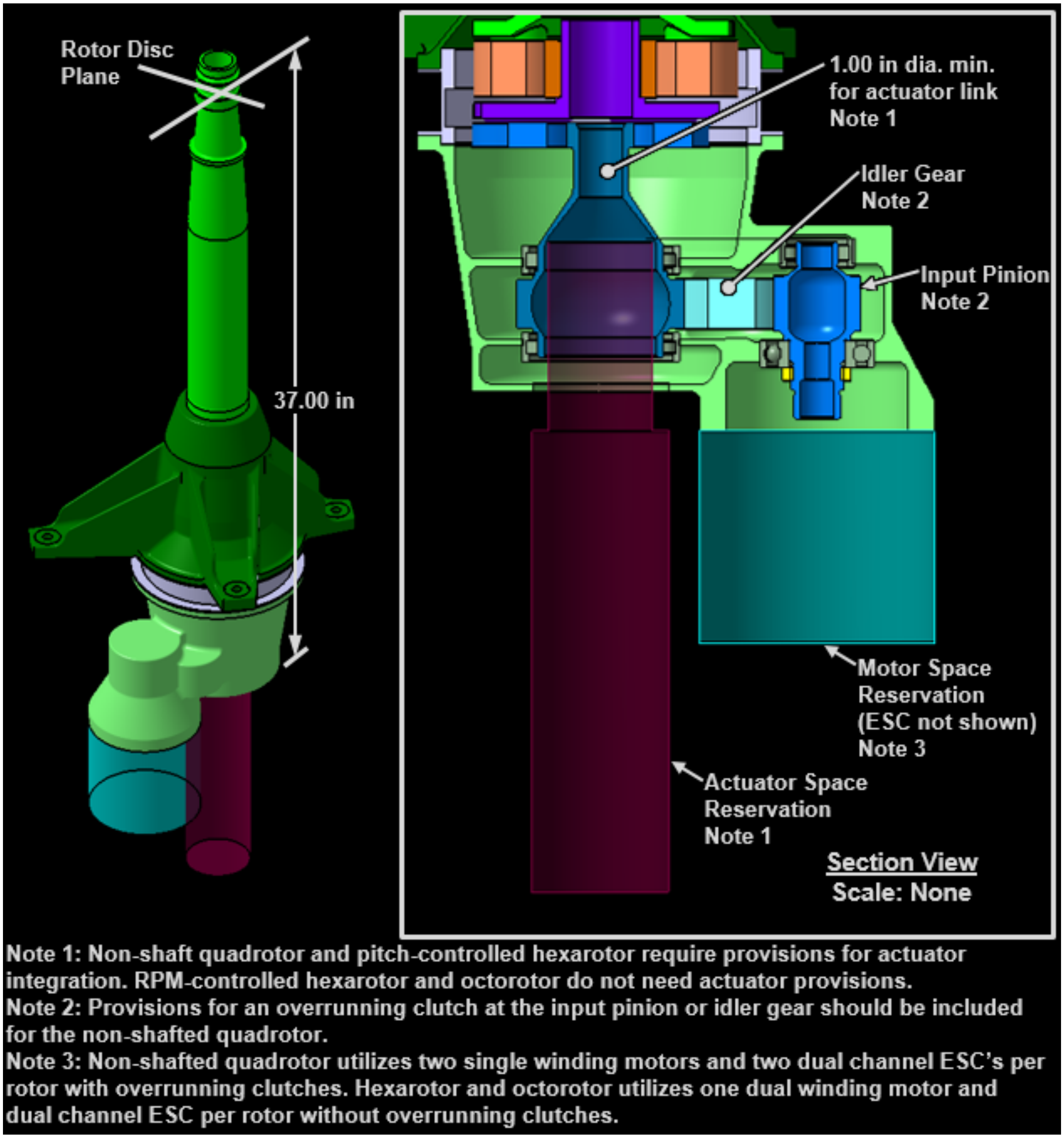


Figure 55: Rotor Gearbox Modules for Quad, Hex, and Octorotors without Interconnecting Shafting

The motor designs for the non-shafted eQuad, eHex, and eOct are also largely similar to the eQuad with interconnecting shafts; however, the redundancy and reliability requirements are increased for each of the non-shafted variants. Fundamentally, the eQuad is assumed to need to power all four rotors for continued safe flight; contrarily, the eHex and eOct are able to continue safe flight if one rotor is unpowered, Section 6 and 7. This drives to a different design scheme for the eQuad than it does for the eHex and eOct.

9.1.4.1 Rotor Gearbox Modules

Gear sizes and overall gearbox size will vary slightly between the quadrotor, hexarotor, and octorotor. For comparison purposes, the two stage planetary system was resized, as this is where the majority of the rotor gearbox weight will be located. Planetary gear mesh applied torque, angular velocity, pitch diameters, and face widths are shown in Figure 56 for the eQuad, eHex (both variable pitch, collective control and variable speed control), and eOct. Gear tooth counts are as shown in Table 11 for all three configurations. NDARC sizing models for the eHex and eOct show higher power demands for rotors located closer to the aft of the vehicle (i.e. motors for rotors one and two are smaller (lower power) than motors for rotors three and four).

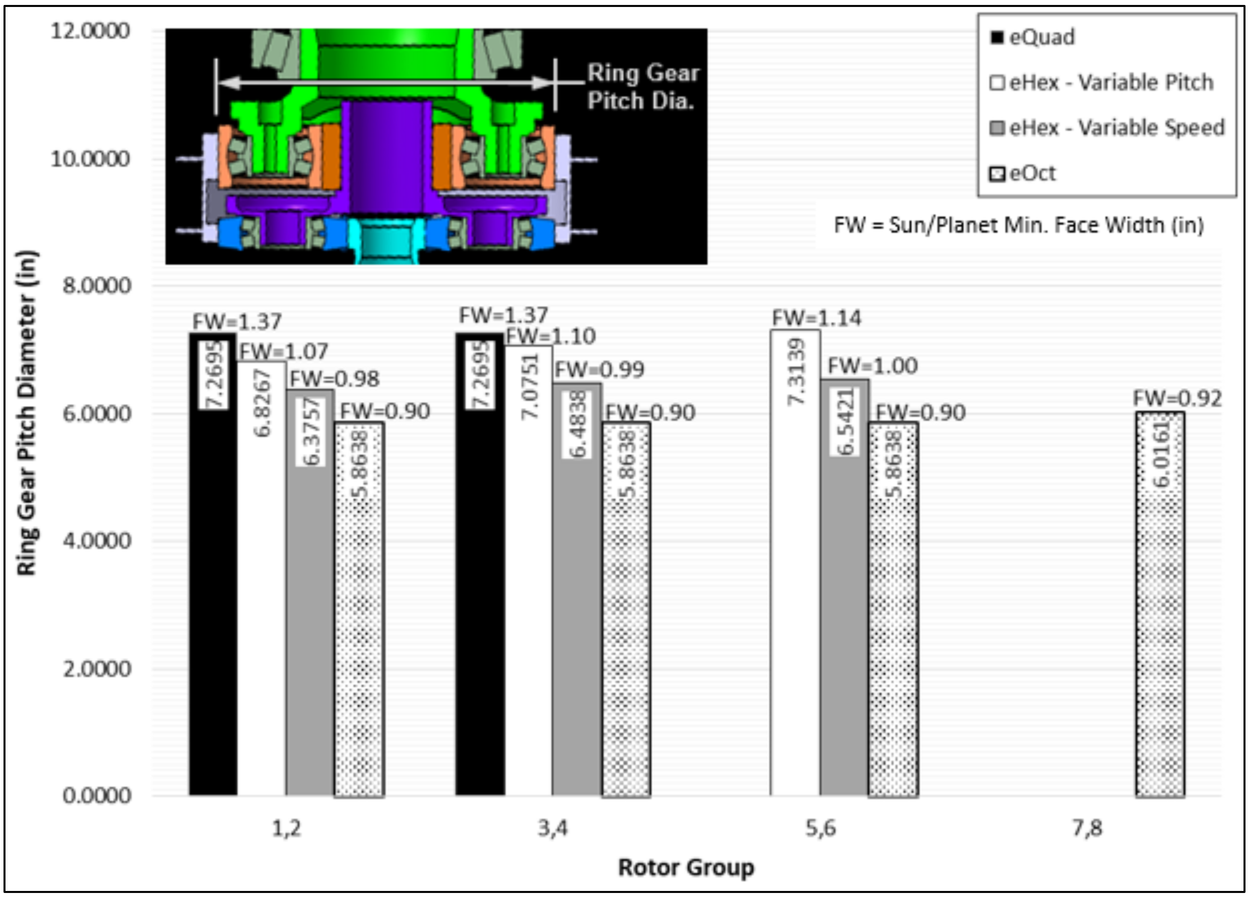


Figure 56: 2nd Stage Planetary System Gear Sizes for eQuad, eHex, and eOct

Life estimates for the bearings were not developed for the eHex nor eOct, but there is little risk of achieving >4,500 hours B10 for this transmission configuration with additional design and analysis for each, desired configuration. Table 12 shows bearing life estimates for the eQuad, as an example.

9.1.4.2 PMSM System for eQuad without Interconnecting Shafting

The variable pitch, non-shafted eQuad requires that if a fault is detected that a motor or inverter may be deenergized, but that the rotor must continue to be powered, leading to the need for a second motor/inverter system connected to the rotor gearbox through an overrunning clutch system, similar to the eQuad with interconnecting shafts, see Section 9.1.1. This results in two fully redundant, dual channel inverters transmitting power to two single channel motors. Each single channel motor transmits mechanical power to an overrunning clutch system in the motor gearbox. The motor gearbox then combines the power from each motor at the output gear and transmits the power to the rotors through a two stage planetary system, see Section 9.1.1 for additional description of the planetary system. Additional redundancy may be added to the system by adding a third single winding motor and third dual channel inverter (totaling 12 motors per aircraft); however, subjective reasoning dictates that more practical solutions exist if greater safety is desired for the eQuad, such as utilizing an interconnecting shaft system.

9.1.4.3 PMSM System for Variable Pitch eHex

The variable pitch eHex differs from the variable pitch, non-shafted eQuad in that continued safe flight may be achieved when one rotor is unpowered, but it is similar to the eQuad in that rotor speed can be controlled via the variable pitch system when the PMSM system is deenergized. This results in a PMSM system that does not need to transmit power through overrunning clutches and allows for a similar level of PMSM system reliability as the eQuad with interconnecting shafts. The PMSM system for the variable pitch eHex is similar to the eQuad, including a single winding motor and a dual channel inverter per rotor, see Section 9.1.1.1 for more detail. Each motor transmits mechanical power into the motor gearbox parallel axis gear system and directly into the two stage planetary system, without the need for an overrunning clutch. When a fault is detected, the affected rotor will need to be deenergized in order to prevent continued failure propagation, potentially leading to fires or other cascading events.

9.1.4.4 PMSM System for Variable Speed eHex and eOct

Similar to the variable pitch eHex, the variable speed eHex and eOct continue safe flight when one rotor is unpowered. However, the fixed pitch rotors associated with the variable speed control scheme result in a rotor overspeed hazard that must be considered further. In the variable pitch quadrotor and hexarotor systems, deenergizing the PMSM system would still permit control of rotor speed via the variable pitch control scheme. The flight control system and/or pilot inputs can be reasonably believed to keep rotor speed below accepted limits to prevent failure onset due to high rotor speeds, or rotor burst. This is analogous to autorotation in a conventional helicopter in which the engine is shut down and the pilot is able to retain control of the aircraft and keep the rotor systems below spec maximums.

In the case of the variable pitch, non-shafted eQuad and eHex, pilot inputs are not a very practical solution to control rotor speed of a single, unpowered rotor as this would require a notable increase in pilot workload. However, it is reasonable to require the flight control system to slow the rotor when the PMSM system is unpowered. Conversely, it is reasonable to assume that the motor controller will be able to limit rotor speeds to within acceptable margins of its internally stored reference speed. Therefore, the variable pitch, non-shaft eQuad and eHex would require both the PMSM system to fail and the flight control system to fail to create the potential for an overspeed hazard.

In the case of the fixed pitch, variable speed eHex and eOct, a failure of the PMSM system will result in loss of control of that rotor, creating a reasonable and conceivable overspeed hazard when the PMSM system is deenergized. This requires additional PMSM redundancy or positive means to slow and stop the rotor included in variable speed eHex and eOct to reduce the probability of an overspeed hazard. A dual winding, dual channel PMSM system is envisioned, however, and additional work is required to study this failure mode and the implications on the system. It is unclear whether the additional winding will effectively reduce the probability of the overspeed hazard to acceptable levels.

9.2 Discussion and Trade Studies

9.2.1 Comparing Component Usage Spectrum – Changing Control Scheme & Number of Rotors

The propulsion system of any vehicle is subject to varying loads. The loads can be broken into two primary categories, low cycle loads and high cycle loads. In vertical-takeoff and landing aircraft, low cycle loads are loads which may occur as little as once during the system life or as frequently as a few times per flight; examples include ground-air-ground cycles or power modulations associated with the transition from hover to cruise. Conversely, high cycle fatigue loads will accumulate rapidly (many times a minute or faster) and are generally seen in rotating components like rotor blades and shafts; examples include maneuver loads and design “imperfections” due to manufacturing tolerances, torque ripple, or conjugate action.

Each of the NASA concept vehicles were designed for the same mission profile, so low cycle load variations, such as ground-air-ground cycles will be similar between each vehicle. Additionally, it is fair to assume that high cycle loads due to torque ripple or conjugate action or similar, design-related phenomena will be similar.

However, the motors for the hexarotor and octorotor were sized for a specific rotor location. That is, motors #1 and #2 are smaller than motors #3 and #4, see Section 9.1.4, meaning that the average power demand per rotor may differ between vehicles for the design mission. In contrast, the motors for the quadrotor are equally sized, regardless of motor location. Due to variations in sizing assumptions, there are likely variations in drive and power system reliability for the hexarotor/octorotor and quadrotor.

Cubic mean power (CMP) was used to characterize how the relative changes in motor sizing may affect component reliability. Using the power spectrums for each vehicle in the design mission, Figure 6 and Figure 7, the cubic mean power was calculated for the pitch controlled eQuad with interconnecting shafts, the pitch controlled hexarotor, speed controlled hexarotor, and speed controlled octorotor. Table 13 summarizes the cubic mean power and compares it to the average motor power per vehicle; in this case MCP was used to calculate average motor power.

Table 13: Comparison of Cubic Mean Power to MCP

Motor Characteristics	**Pitch Cont. eQuad w/ Shafts**	**Pitch Controlled Hexarotor**	**Speed Controlled Hexarotor**	**Speed Controlled Octorotor**
Average MCP Rating [hp]	112	83	76	66

CMP per Motor [hp]	94	76	69	61
Percent of CMP to MCP	**84%**	**92%**	**91%**	**92%**

9.2.2 Effects on Reliability and Safety – PMSM Architecture

As seen in Table 13, the relative usage of the hexarotor and octorotor configurations is trending in an unfavorable direction due to architectural decisions to make the forward-most motors smaller than the aft-most motors. Additionally, simulations, Section 5, have shown that the vehicles without interconnecting shaft systems are at greater risk of frequent, large power transients and that variable speed, fixed pitch control schemes are more sensitive to turbulent conditions.

Architectural decisions may be modified, though, to create more reliable systems without changing control scheme or rotor quantity. In example, the PMSM system for the eQuad with interconnecting shafting can be modified, changing redundancy management philosophies, complexity, and weight. Table 14 summarizes four different PMSM system architectures for the eQuad with interconnecting shafts. Table 14 shows fail-safe (FS) and FOFS architectures for consideration. FS systems able to continue to operate safely after a single failure is detected and managed and FOFS are able to continue to operate safely after two independent failures have been detected and managed. The architectures shown in Table 14 vary the intermediate rated power (IRP) of the motor windings and power electronics to achieve varying levels of redundancy, each using the interconnecting shaft system to achieve power transfer between rotors. In example, a FOFS single winding, single channel system may be developed if each PMSM system is large enough to manage two complete PMSM failures. Also, a FOFS dual winding, dual channel PMSM system may be developed in which the individual winding and inverter size is reduced. Each of these systems must be developed further to draw fair comparisons between them and the baseline, FS single winding and FOFS dual channel inverter PMSM system.

Table 14: Potential PMSM System Architectures
Note: The shaded row is the baseline system described in Section 9.1.1.

Description	# of Windings per Vehicle	# of Inverters per Vehicle	# of Windings Required for Safe Flight	# of Inverters Required for Safe Flight	Approximate Winding IRP Rating [hp]	Approximate Inverter IRP Rating [hp]
FS Single Winding, FOFS Dual Channel	4	8	3	6	170	85
FOFS Single Winding, FOFS Single Channel	4	4	2	2	255	255
FOFS Single Winding, FOFS Dual Channel	4	8	2	6	255	85
FOFS Dual Winding, FOFS Dual Channel	8	8	6	6	85	85

9.2.3 Impacts of SC-VTOL-01 Single Failure Criteria on Drive and Power System

SC-VTOL-01 extends fail-safe and redundancy management design techniques to leverage the potential to segregate failures through unique, multi-rotor configurations. In doing so, SC-VTOL-01 has more stringent design requirements associated with redundancy management and single or multiple load path designs. While many differences exist between SC-VTOL-01 and other certification specifications, such as CS-23 or CS-27, the single failure criteria presented in SC-VTOL-01 is particularly challenging when considering the drive and power system for heavier-than-air VTOL aircraft.

SC-VTOL-01, VTOL.2250(c) requires, "…For Category Enhanced, a single failure must not have a catastrophic effect upon the aircraft." CS-27 has similar single failure criteria for Category A certification basis, except that some subparts allow exceptions. For instance, 29.901(c) states:

> *For each power plant and auxiliary power unit installation, it must be established that no single failure or malfunction or probable combination of failures will jeopardise the safe operation of the rotorcraft except that the failure of structural elements need not be considered if the probability of any such failure is extremely remote.*

As can be seen, 29.901(c), a requirement for CS-27 Category A certification basis, allows for single failure of structural elements if the failure is extremely remote. SC-VTOL-01, on the other hand, does not permit single failures without exception, regardless of the probability of failure.

In absence of direct compliance with VTOL.2250(c), SC-VTOL-01 will permit single failures which lead to catastrophic effects on a limited, case-by-case basis. In either case, direct compliance with VTOL.2250(c) or via specific exception to VTOL.2250(c), a rigorous safety assessment process is required to verify compliance to the stringent, single failure criteria (ref. 8). For direct compliance with VTOL.2250(c), the conclusion of the safety assessment process should demonstrate that all single failures are not catastrophic (ref. 7). This includes demonstration that the single failure is not catastrophic or could include fail-safe design practices which allow continued safe flight and landing for either (1) the remainder of the normal flight envelope, or (2) an acceptable emergency flight procedure, including adequate time to safely divert and land at a pre-approved landing site, such as a vertiport or helipad.

The single failure criteria has a potentially large impact on all aircraft intended to be type certificated under SC-VTOL-01. Single load path structures are of prime concern, being mechanical systems in which redundant load path structures can be difficult to integrate. Blade loss, rotor loss, or excessive vibrations leading to structural failure (and others) are reasonable and conceivable potentially catastrophic failure modes that must be addressed. Even the simplest power system architectures known to the UAM community, direct-drive motors, shown in Figure 57, must be evaluated because single load path structures are commonly found in direct drive systems. Direct drive sysetems generally include single load path rotor system interfaces, housings, and mounts plus blade, and rotor systems contain typically contain single load paths due to weight and volume constraints.

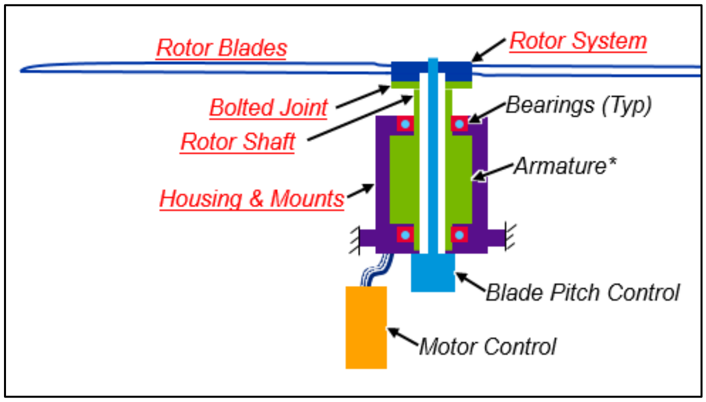


Figure 57: Direct Drive Motor Illustration with Potential Single Load Path Structures Shown in Red and Underlined.

9.2.3.1 Reasonable & Conceivable Catastrophic Failure Modes for Quad, Hex, and Octorotors

In the case of the NASA RVLT quadrotor, loss of function of a single rotor is assumed catastrophic. Reasonable and conceivable failure modes leading to catastrophic events exist in the drive and power system, primarily in the rotor blades, rotor system, and rotor gearbox system; see Section 12.2.3 for drive and power system failure modes. The catastrophic events may include rotor burst, blade loss, or other failures leading to excessive vibrations and structural failure. In order for direct compliance with VTOL.2250(c), multiple load paths must be integrated into the quadrotor design to prevent single failures resulting in rotor burst, blade loss, or excessive vibrations leading to structural failure.

In the case of the NASA RVLT hexarotor and octorotor, the number of rotors are assumed to allow for continued safe operation after loss of function of a single rotor. However, rotor burst, blade loss, and excessive vibrations leading to structural failures remain reasonable and conceivable events and require further consideration. See Section 7 for additional information.

9.2.3.2 Potential Design Changes for Compliance with VTOL.2250(c)

Fail-safe designs within the rotating frame exist in fielded helicopters today. Fail-safe, mult-load-path designs and in situ monitoring have been being incorporated developed and introduced to the VTOL fleet since the 1970s or earlier (ref. 26). Fielded designs include multiple load path structures and early, in situ fault detection. Extending existing, fielded designs and research into experimental designs, multiple load path structures were developed for the rotor shaft system and rotor gearbox planetary system, and prior designs and research in real-time diagnostics and prognostics in the rotating frame are discussed. Additional development is required to verify that these design features will (1) allow for continued safe operation after the first failure, (2) detect the first failure prior to inducing a secondary failure (through inspection, real-time monitoring, or other), and (3) not add non-inspectable, latent failures into the system.

The baseline rotor gearbox system includes a rotor shaft system with an integral planet carrier. The rotor shaft system is a single load path structure supported by two tapered roller bearings. It transmits torque from the planetary system, into the rotor system and transmits aerodynamic bending loads from the rotor system to the tapered bearing set. The tapered bearing set transmits the bending loads out to stationary structure. In an attempt to create a multiple load path structure to comply with VTOL.2250(c), an additional four-point contact ball bearing and torque tube were added to the rotor shaft system, see Figure 58. The four-point contact ball bearing is mounted to the rotor hub and transmits aerodynamic loads in the event that the rotor shaft structural integrity is compromised. The torque tube is mounted to the rotor hub through a torque plate and transmits torque from the integral planet carrier at the base of the rotor shaft, to the torque plate and into the rotor hub in the event that the rotor shaft structural integrity is compromised. The rotor shaft diameter was increased in order to make room for the torque tube.

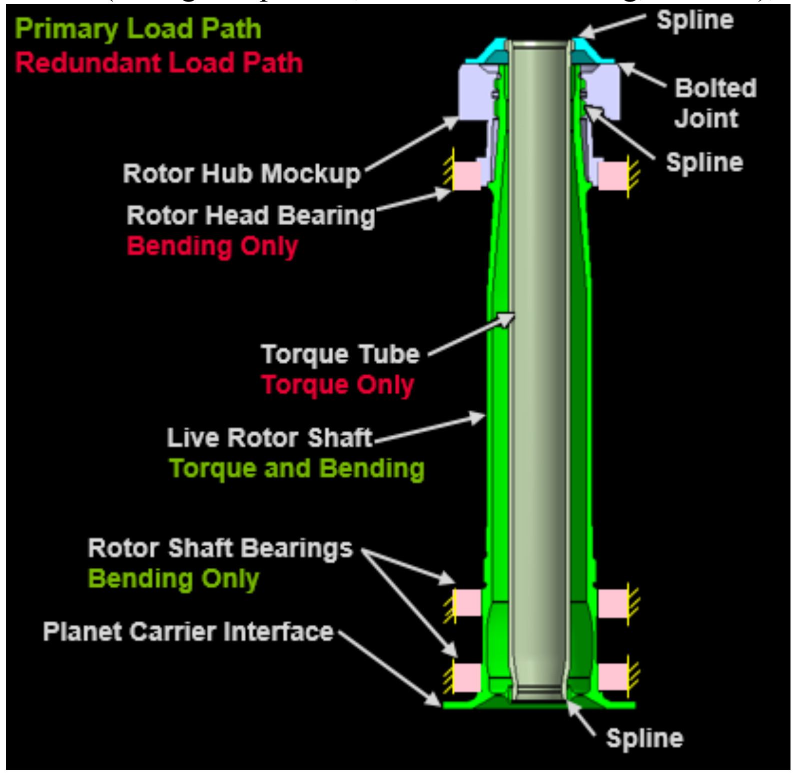


Figure 58: Cross Section of Proposed Dual Load Path Rotor Shaft System

The baseline rotor gearbox planetary system also includes single load path gear systems. The baseline final reduction state uses two simple planetary systems (each are sun-in, ring-fixed, carrier-out systems), see Section 9.1.1 for additional information. A dual load path compound planetary system, Figure 59, was developed to eliminate single failures from the final reduction stage based on the high contact ratio, staggered compound planetary system developed by Robuck, et al (ref. 27, 28, 29, 30). The compound planetary system transmits power from the input sun gear, into six planet gears; the planet gears react against the stationary ring gear and transit power

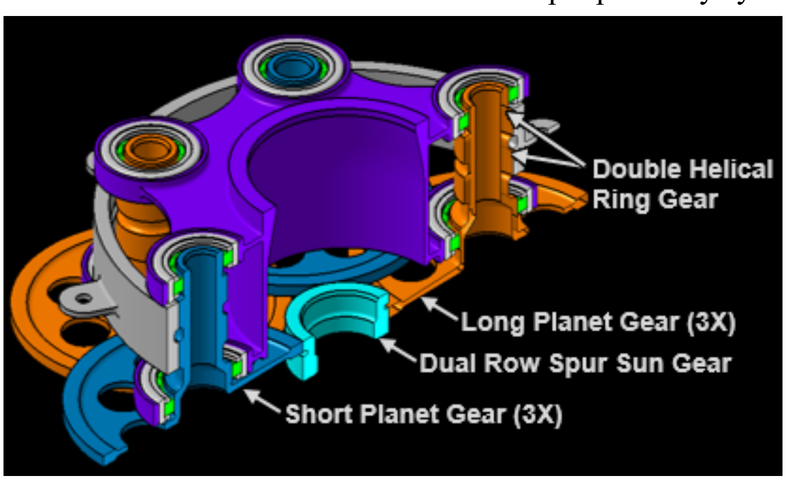


Figure 59: Section View of Proposed Dual Load Path Compound Planetary System

through the output carrier to the rotor shaft system. The dual load path system for application in the NASA RVLT concept vehicles utilizes staggered long and short planet gears to create two rows of spur gear teeth at the sun/planet mesh. The long/short planet gears then transmit load into a high contact ratio double helical planet/ring mesh. The redundant row of the planet/ring helical gear teeth and the high contact ratio allow for torque to continue to be transmitted in the event of a single gear tooth failure. Design, analysis, and testing over decades of seeded fault and overload testing have shown that in similar final reduction stages a propagating crack in the root of a gear tooth will locally fracture or sever one gear tooth and not propagate to neighboring teeth. Crack growth simulation can be used to model behavior of tooth fractures to control crack propagation direction through manipulation of tooth to rim thickness and stiffness variations from tooth to tooth. Additional design, analysis, and verification testing are required to substantiate the fail-safe features of this gear train in this application.

Fail-safety may also be achieved through damage tolerant designs and in situ monitoring of specific failure modes. In situ monitoring may include systems intended to indicate cracks during daily, visual inspections or may be more advanced, integrated systems which alert the pilot and crew in flight of an impending failure. An early in situ monitoring (in conjunction with multiple load path structures) system was developed for the YUH-61A, in which a pressurized rotor shaft and pressurized rotor system components incorporated pressure-differential loss-detection systems to indicate "safe" or "unsafe" conditions during visual inspections. The monitored components were also designed for damage tolerance, so that they could continue operation for a minimum of 30 flight hours after the crack was detected by the in situ monitoring system (ref. 26). More recent advances in in situ monitoring include vibration health monitoring. A comprehensive bench test program ran through the 1990s to characterize the ability to detect gear tooth bending fatigue failures using accelerometers and associated processing equipment. Specifically, seeded fault testing on the CH-47D Forward Transmission Spiral Bevel Pinion showed continued safe operation for over two hours in a "get-home" cruise power setting after the crack was detected (ref. 31). Currently, EASA and others are investigating the use of acoustic emissions and other sensor suites to detect failures in the rotating frame. A recent study on an EC225 showed that acoustic emission sensors showed improved detection of planetary gear failures over vibration, accelerometer-based systems (ref. 32).

9.3 Summary and Conclusions

Modular drive and power system architectures were developed for the NASA RVLT concept vehicles under consideration. The modular elements were used to facilitate a fair comparison of component reliability and system complexity in order to meet PSSA catastrophic failure probability budgets for each aircraft studied. PMSM rough sizing was performed in order to validate the feasibility of early architectural assumptions like high speed motors (~15,000 RPM), and steps were taken to fix the turbogenerator, also increasing fidelity of early direct drive generator assumptions (~30,000 RPM).

Architectural decisions like including interconnecting shafts, safety philosophy, and number of rotors will impact component reliability due to differences in duty cycle and more time spent at higher relative powers. S&C models show manageable power transients for the quadrotor with interconnecting shafts, but as interconnecting shafts were removed, control schemes changed, and number of rotors increased, the power transients increased to unmanageable levels and future design work must lower power transients. Usage spectrums show that resizing motors for each rotor, as NDARC models show for the eHex and eOct, may lead to more aggressive component usage, reducing reliability.

SC-VTOL-01, VTOL.2250(c) single failure criteria may create undesirable changes to mechanical systems if a fail-safety philosophy is not incorporated early in the design process. Fail-safe mechanical designs exist, but each design must be carefully evaluated to ensure that it is not impeding assemblability or inspectability, which could create catastrophic, latent failure modes. The intent of 2250(c) instead is to prevent to be solely relying on monitoring means to prevent catastrophic failures. More guidance material is being developed and will be proposed as MOC.

In future work, further evaluation of VTOL.2250(c) single failure criteria and its impact on dynamic system design choices is recommended. Analytical modeling and testing of fatigue crack growth and detection means (in situ monitoring, inspection intervals, etc.) may be used to characterize design criteria and methods to meet the single failure criteria in the dynamic system.

Each of these areas should be evaluated further in the future. Future work should investigate the probability of diagnostic and prognostic systems to detect crack growth in the rotating frame. Progressive bearing failures should be investigated to determine their impact on motor performance as the bearing failures progress, specifically effects of metallic bearing chips on winding and insulation integrity and chip migration through or around PMSM systems.

The intent of 2250(c) is to limit reliance on monitoring required to prevent catastrophic failures. Current work focused on rotor shaft and gear tooth bending fatigue cracks, bearing failures, PMSM failures, and potential detection methods. Future work should include additional design trade studies, analytical modeling, and testing to support development of future means of compliance.

10 THERMAL MANAGEMENT SYSTEM

This section includes the work performed by the thermal management team. Two separate tasks are addressed:

- Develop system architecture for cooling system concepts to support PSSA, vehicle reliability and safety assessment;
- Assess the reliability and safety impacts of liquid cooled vs. air cooled motors and motor controllers.

The first task involved developing conceptual designs for TMS for the batteries and electric propulsion and drive system components for the eQuad, hQuad, tQuad, eHex, and eOct aircraft. System architecture diagrams were provided to the reliability and safety teams for further assessment and use in the FMECA and PSSA. A representative thermal design mission was developed based on the NDARC vehicle sizing mission and was considered in the cooling system designs to ensure compatibility with hot day operation.

Following the conceptual design development, a more detailed trade study was performed to evaluate differences between air and liquid cooled motors and motor controllers. Boeing collaborated with Safran Electrical and Power, using their ENGINeUS 100kW Smart Motor as a basis for the analytical studies. The Safran motor was selected because of its nominal power output and adaptability to both air and liquid cooling. During the trade study, air, and liquid cooling systems were sized and analysis was performed to evaluate internal motor and power electronic component temperatures throughout the thermal design mission.

10.1 System Description and Analysis Approach

To support development of thermal management concepts, the team defined a thermal design mission based on the electric quadrotor NDARC vehicle sizing mission. The NDARC mission was adjusted for ambient conditions that would typically be used for cooling system sizing. The vehicles considered in this study are intended for operation in urban environments; therefore Phoenix, Arizona was selected as a representative hot urban environment.

Review of Phoenix climatic design information (ref. 33) identified three relevant conditions for further study.

- Phoenix 1% hot day
  - 108.3°F Dry Bulb
  - The 1% frequency of occurrence (based on warm conditions) is a typical cooling system sizing condition.
- Phoenix hot mean – mean of 10% monthly high temperatures
  - 88.2°F Dry Bulb
  - This condition is considered a representative temperature for assessing cooling system performance impacts on component reliability.
- Phoenix worst case hot day
  - 125°F Dry Bulb at SL
  - This condition is based on the maximum temperature under which the aircraft and system components must be able to operate. For electronic components, this means that component temperatures remain below manufacturer defined limits.

The pressure altitude and ambient temperature profiles from the vehicle sizing mission were adjusted for the Phoenix design conditions described above. Power requirements were assessed to determine whether those also needed adjustment from the NDARC mission to the Phoenix conditions. Air density comparison for the two profiles showed variation within 5%, so the NDARC power levels were considered representative.

The thermal management sizing mission is shown in Figure 60 through Figure 62.

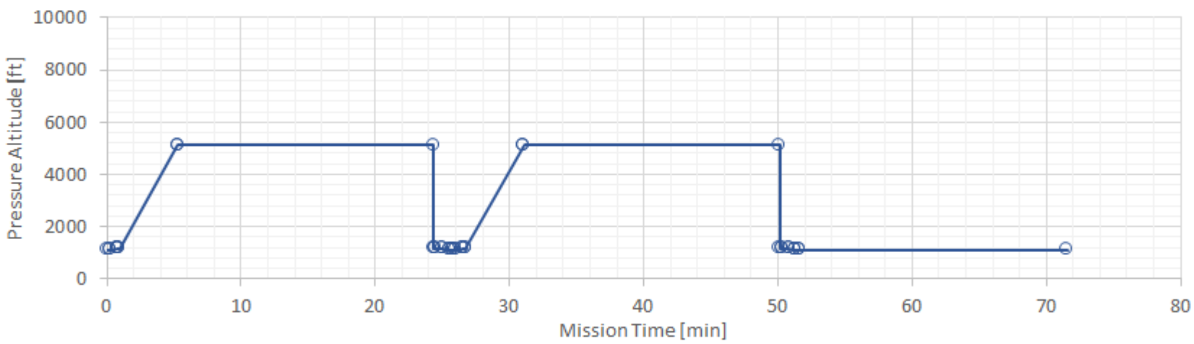


Figure 60: Thermal Sizing Mission, Pressure Altitude

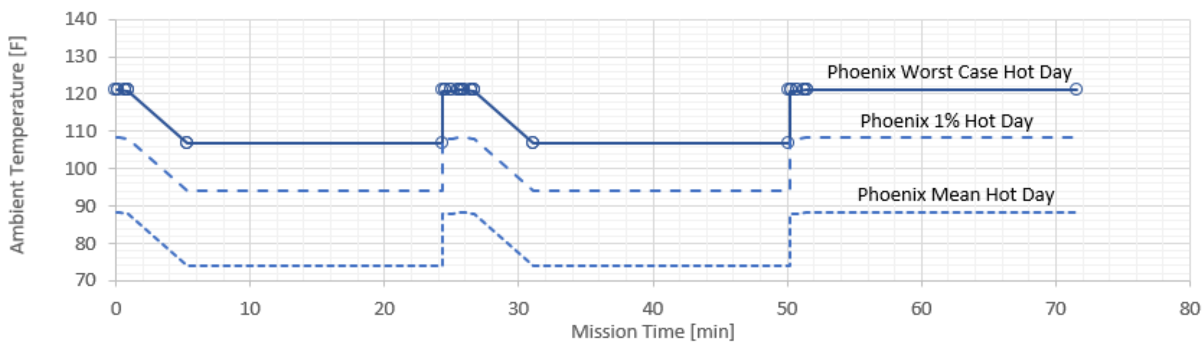


Figure 61: Thermal Sizing Mission, Ambient Temperature

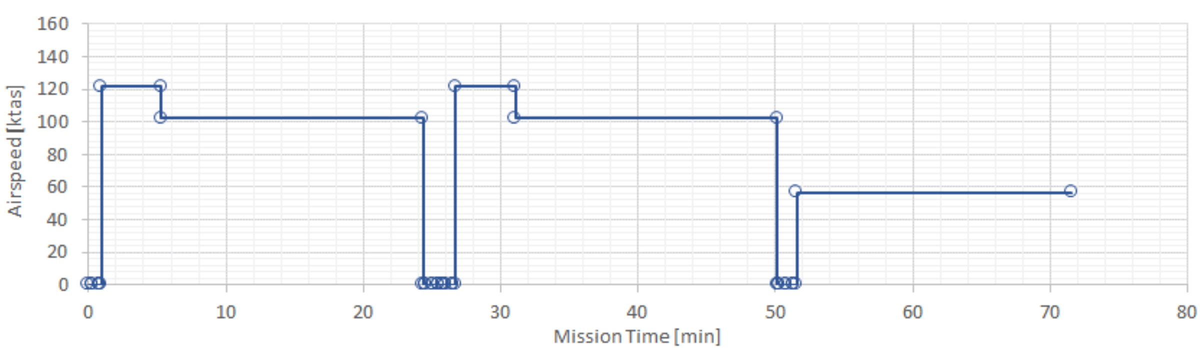


Figure 62: Thermal Sizing Mission, Airspeed

The vehicle power requirements and resulting component heat dissipation values are shown in Figure 63 through Figure 66. The heat dissipation values shown are based on constant efficiency throughout the mission. In reality, component efficiencies will vary during the mission as power demand, motor speed, torque and other parameters change (ref. 34) identified that battery efficiency can vary widely during different mission phases from 88.6% during Climb to 97% during

Taxi. The impact on battery heat dissipation is illustrated by the three graphs in Figure 66. Thermal losses from power distribution cables are not included but may be significant and should be included in the vehicle thermal management design.

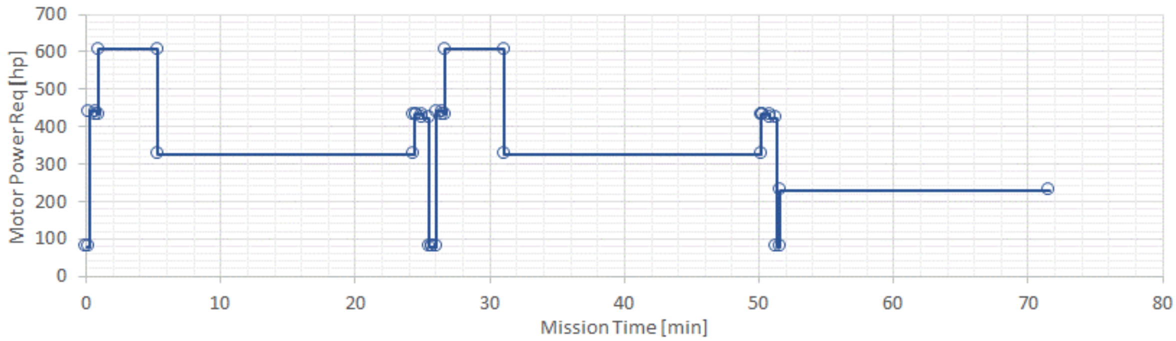


Figure 63: Thermal Sizing Mission, Motor Power Required

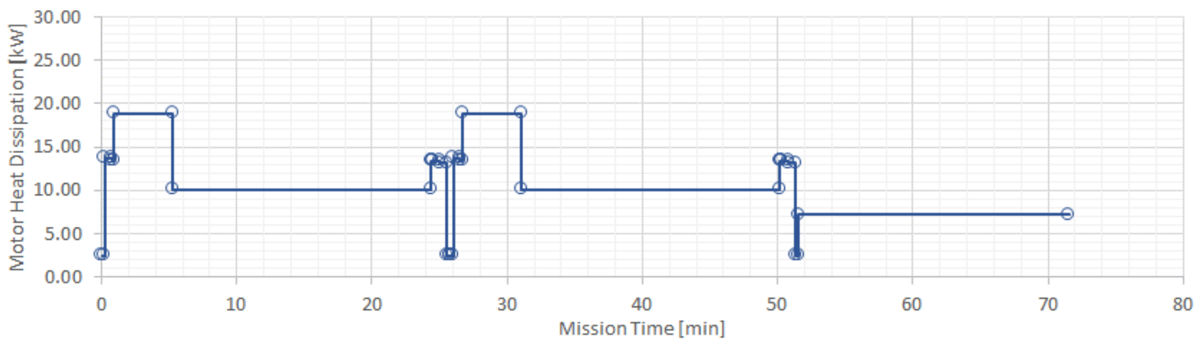


Figure 64: Thermal Sizing Mission, Motor Heat Dissipation

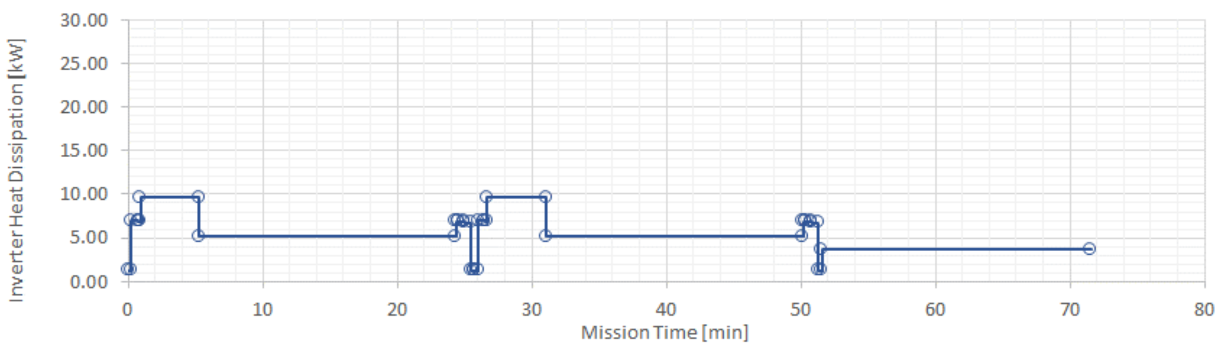


Figure 65: Thermal Sizing Mission, Inverter Heat Dissipation

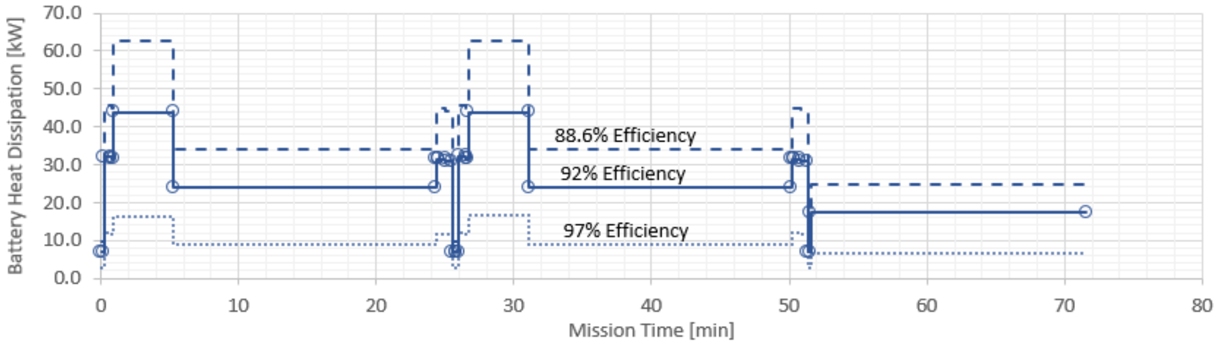


Figure 66: Thermal Sizing Mission, Battery Heat Dissipation

Conceptual cooling systems were developed to address specific cooling needs for the batteries and electric propulsion and drive system components.

Lithium-ion (Li-ion) batteries are the most temperature sensitive of the components considered, the cooling system should maintain battery temperatures as defined below:

- 77°F Optimal
- 140°F Maximum

Review of typical battery configurations determined that air cooling is feasible and that temperature variation within the battery pack can be minimized with good design. Temperature variation will be dependent on the cooling flow rate provided, as well as the battery arrangement and resulting cooling flow path. Coolant pressure drop through the battery is also an important consideration when defining the cooling flow path as it affects fan power.

The battery cooling system is shown in Figure 67. The system includes both primary and back-up cooling modes. The primary cooling system is a vapor refrigeration system that provides cooled air to the battery packs via refrigerant to air evaporators. The primary cooling paths are closed loop. The cooled air leaves the evaporators and flows through the battery packs where the heat from the batteries is transferred to the air. The warm air flows back to the evaporators where the heat is absorbed by the refrigerant. Dual fans are included for each battery pack, capable of providing cooling flow in the event of a single fan failure. If there is a failure in the vapor refrigeration system, the back-up cooling system will draw outside air via the back-up fan. In this mode of operation, the battery pack cooling fans will be used to provide the outside to the battery packs and the heated air will be dumped overboard through the back up valve.

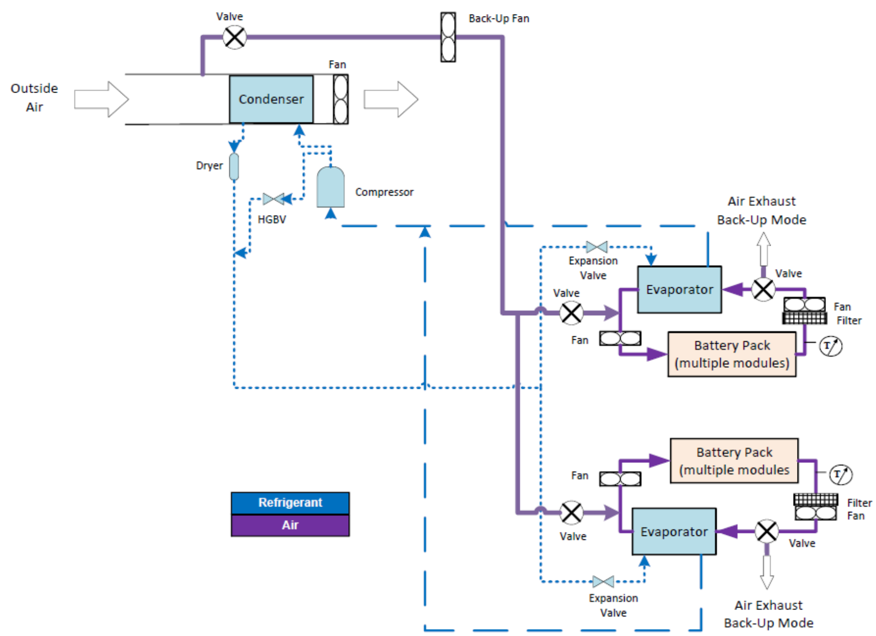


Figure 67: Battery Thermal Management System

A thermal management system (TMS) was developed for a gearbox-only configuration, applicable to eQuad, hQuad, and tQuad platforms. Since the gearbox itself generates a relatively low amount of heat, a simple closed-loop TMS with adequate oil volume provides enough cooling capacity to dissipate the frictional heat generation. This schematic is configured to deliver oil flow at the required temperature and pressure to the lube jets, providing cooling and lubrication to the gear mesh. Figure 68 below shows the gearbox cooling/lubrication schematic.

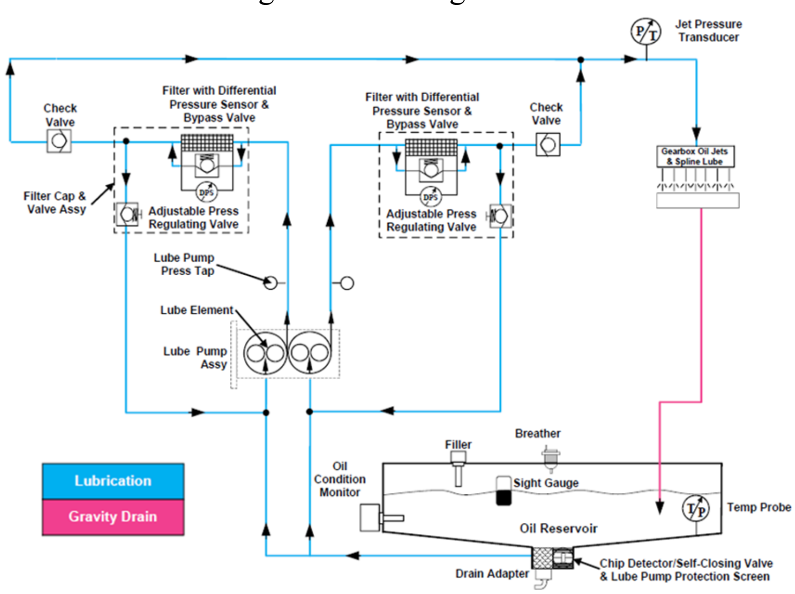


Figure 68: Thermal Management System Schematic, Gearbox-Only Configuration

The above configuration provides a steady flow of MIL-PRF-7808 oil to the gearbox oil jets. A fully-redundant lubrication pump assembly ensures that adequate flow and pressure are maintained at the lube jet inlets in the event of a single pump failure. Each pump element provides the full volumetric oil flow through independent filter assemblies with integrated pressure regulating valves (PRVs). The PRVs are calibrated to return 50% of each pump's outlet flow back to its inlet during normal operation. If flow from one pump drops, the opposite pump's PRV will close (partially or fully) as necessary to make up for the reduced flow from the malfunctioning pump and ensure full, continuous oil flow to the jet gallery. Check valves are included at the filter/PRV outlets to prevent backflow in the event of single pump failure or system shutdown.

This study also considered the use of an electric motor and its controller – the Power Electronics (PE) unit – to drive the propulsion system. The Thermal Management System (TMS) must be modified to accommodate the higher heat loads of the motor and PE. The additional system components are indicated in Figure 69. This system design applies to all rotor gearboxes except the tQuad configuration.

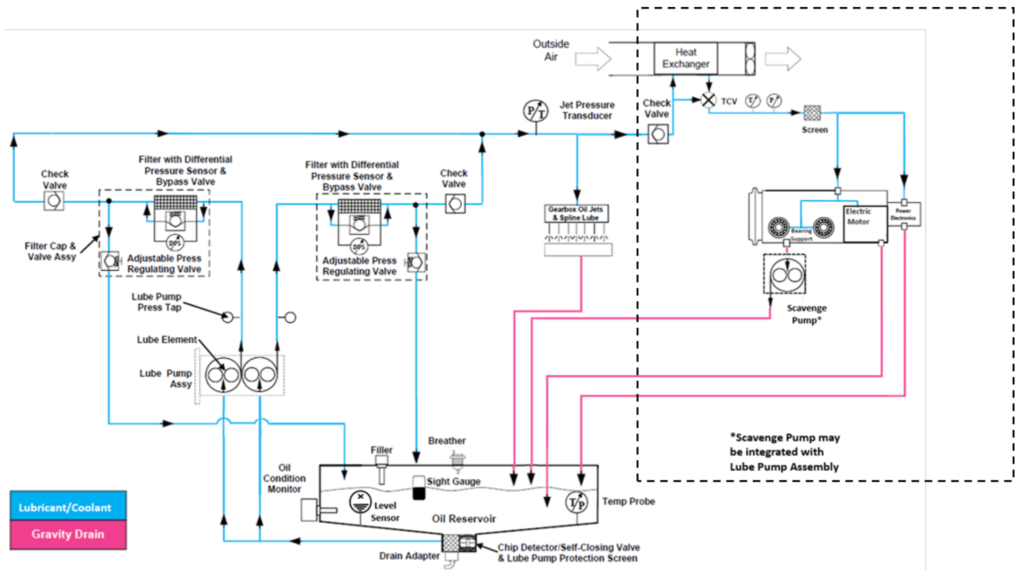


Figure 69: Thermal Management System Schematic, Integrated Motor, PE, Gearbox Configuration

The above schematic shows the additional equipment for this configuration. The baseline system is unchanged, but an electric motor, PE, heat exchanger, and scavenge pump are added in parallel with the gearbox cooling flow (see outlined portion of schematic). The heat loads of the motor and PE are significantly higher than the gearbox alone, and vary significantly throughout the mission profile depending on the power needs of the propulsion system. Therefore, heat management is critical to ensuring the life and operation of all components.

Since this configuration heavily integrates mechanical lubrication with electronics cooling into a single closed-loop system, MIL-PRF-7808 turbine oil was selected as a combined coolant/lubricant. MIL-PRF-7808 has a good balance of lubrication and heat transfer properties, while also acting as a dielectric material, which is critical when directly cooling electronic devices that could induce arcing in the presence of metallic particles from the gear/bearing lubrication elements of the system.

The motor and PE are shown in parallel, which ensures oil at the same temperature and pressure is provided to each inlet. To reduce overall system flow, pump, and reservoir size, the motor and PE can be configured in series. However, care must be taken to ensure that the inlet temperature and pressure requirements of both components are met throughout the mission profile. This study considered an integrated motor design, which contains both the motor and PE in a single unit.

Specifically, Safran identified the ENGINeUS 100 Smart Motor as a viable candidate to meet the power needs for this study. For the purposes of this study, it is treated as a motor and PE unit in series, and Boeing determined a single, upstream oil temperature, while Safran assessed the individual component temperatures. Further details are provided in the Trade Studies section below.

As the vehicle operates, the oil absorbs heat from the motor and PE, significantly increasing in temperature. A heat exchanger is included upstream of the motor and PE to dissipate the system heat to the ambient air. The system was analyzed assuming both ram-driven and fan-driven flow through the air side of the heat exchanger (discussed later in the analysis section of this report). A built-in temperature control valve (TCV) allows cold oil to bypass the heat exchanger during cold day operation or cold start, protecting it from pressure spikes induced by the oil's higher viscosity at low temperatures. During normal operation, the TCV modulates to control the heat exchanger outlet temperature within an allowable range suitable to the downstream components.

Lastly, a bearing scavenge pump is included to draw oil from the bearing sump, which is typically at ambient pressure and would otherwise rely on gravity to drain into the sump. The scavenge pump ensures steady oil flow back to the reservoir during vehicle accelerations and altitude changes. As noted in Figure 69, the scavenge pump is shown separately for clarity, but can be packaged in a single assembly with the lube pump elements.

10.2 Trade Studies

Following conceptual design definition, a more in-depth study was conducted to explore differences between liquid and air cooled motors and motor controllers. In order to get a useful comparison, the team felt it was important to select a motor capable of both liquid and air cooling. The Safran ENGINeUS 100 Smart Motor was selected as a candidate and Safran Electrical and Power agreed to collaborate on the study.

The ENGINeUS 100 fits within the power required for a single motor from the electric quadrotor vehicle sizing study, and was used to define the power profile for the trade study, shown in Figure 70. The ENGINeUS 100 heat dissipation was defined by Safran and considered the power required and component efficiencies over the mission profile.

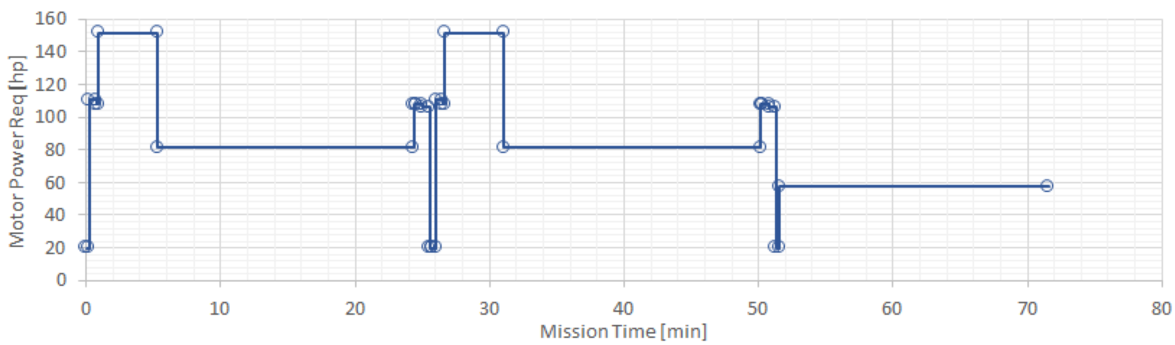


Figure 70: Motor Cooling Trade Study, Motor Power Required

The trade study mission profile and boundary conditions are the same as those shown previously in Figure 60 - Figure 62 for pressure altitude, ambient temperature, and airspeed. These profiles were provided to Safran for assessment. It was determined that acceptable cooling could be achieved on the worst case hot day so the other hot day conditions were not considered.

The cooling system shown previously in Figure 69 was used for the trade study cooling analysis. However, the focus was on Smart Motor cooling only, and did not address the drive system components. In addition, an alternate air cooled Smart Motor concept was defined for the trade study. Detailed analysis was performed by Safran to assess internal motor and power electronics component temperatures for each cooling method.

Over the course of the study, two sub-trades were identified. For both the liquid and air cooling systems, the use of fan-driven (constant flow) and ram-driven (vehicle speed dependent flow) cooling were considered. Note that this refers to how the heat is shed to the ambient air heat sink. For the liquid-cooled system, heat is shed indirectly to the ambient air through an oil to air heat exchanger. For the air-cooled system, heat is shed directly from the Smart Motor to the ambient air.

A simplified thermal model was developed to determine the oil inlet temperature to the Safran ENGINeUS 100 Smart Motor. The model accounts for steady heat loads generated by the motor bearings and gear box, and time-varying heat generated by the motor. Boundary conditions and motor heat dissipation are based on the Phoenix Worst Case mission profile, which defines the altitude, outside air temperature, and air flow through the heat exchanger. Pressure losses were not modeled and oil flow rates were balanced to ensure the required oil flow is delivered to each branch, driven by a generic constant-speed pump. A schematic of the simplified thermal model is shown in Figure 71 below.

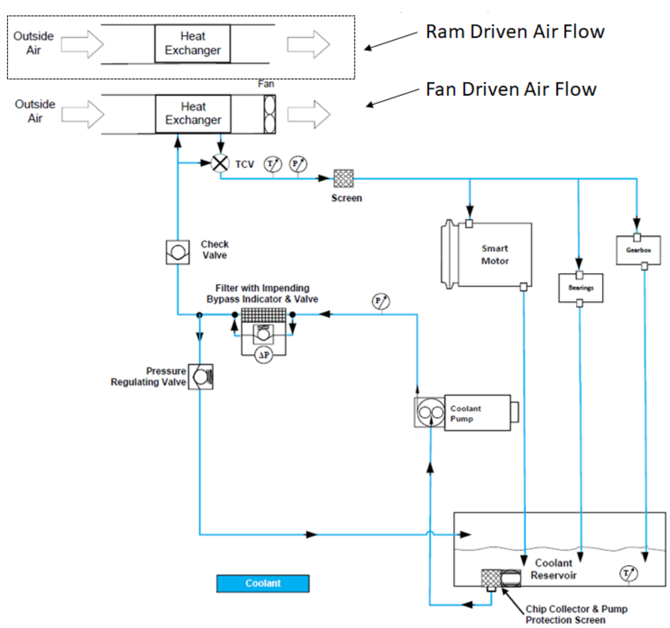


Figure 71: Oil Cooled Thermal Model Schematic

The model also accounts for the heat exchanger effectiveness as a function of the air-side and oil-side mass flow rates. The cooling air through the heat exchanger was modeled two ways, assuming either ram-driven or fan-driven flow, as shown in Figure 71. The ram-flow version of the analysis assumes the vehicle has a ram inlet scoop to supply air through the heat exchanger. The air side mass flow is a direct function of the mission profile (vehicle speed and altitude). The fan-flow version of the analysis assumes the vehicle has a built-in cooling fan to provide a constant volumetric flow of air to the heat exchanger, regardless of the speed and altitude.

The resulting oil temperature at the motor inlet was calculated for both cases and provided to Safran for their motor/PE analysis. Using the same mission profile, Safran calculated a maximum operating temperature profile for the motor and PE that comprise the ENGINeUS 100 Smart Motor, to verify that both components remained within their allowable operating temperature ranges throughout the mission. Boeing and Safran analysis results are discussed in the following section.

The oil temperatures at the motor inlet (determined by Boeing's thermal analysis) are shown below. Figure 72 shows the results of the ram-driven flow analysis, and Figure 73 shows the results of the fan-driven flow analysis.

For most of the mission, the ram flow system keeps the oil temperature 8-10°F cooler than the fan flow system. However, being reliant on forward velocity for air flow, the heat exchanger loses all cooling capacity when the vehicle enters hover mode. The resulting oil temperature quickly approaches 170°F at the motor inlet. Figure 72 shows the vehicle airspeed over time for reference. Note that the oil temperature peaks align with the sudden drops in velocity throughout the mission profile.

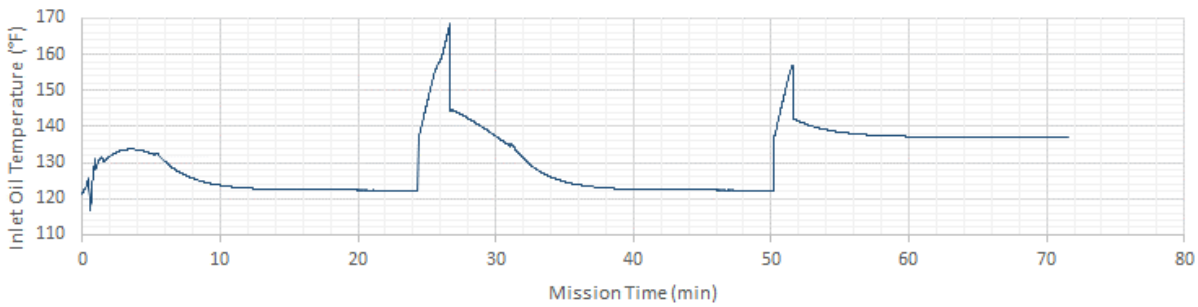


Figure 72: Inlet Oil Temperature (Ram-driven Flow)

To counter this effect, the model was modified to consider the use of a cooling fan, which would provide constant volumetric air flow through the heat exchanger, regardless of the vehicle operation. The heat exchanger always maintains some level of effectiveness in this configuration, since cooling flow is always available. The air side mass flow changes with altitude, but it never drops to zero entirely. Therefore, the high temperature spikes are eliminated, as shown in Figure 73. The oil runs slightly warmer throughout most of the mission, since a fan would generally provide less mass flow than a ram scoop in forward flight.

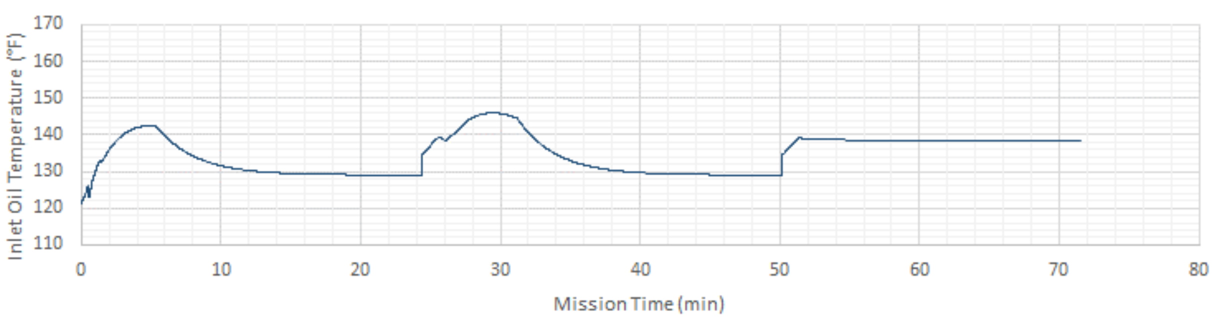


Figure 73: Inlet Oil Temperature (Fan-driven Flow)

In either configuration, the oil temperatures during level flight are well below the maximum allowable inlet temperature for the motor.

The oil temperatures shown in Figure 72 and Figure 73 were then provided to Safran to analyze with respect to the ENGINeUS 100 Smart Motor. The resulting maximum temperatures of the motor and PE for the ram and fan flow configurations are shown in Figure 74 and Figure 75, respectively. Note that Safran's CFD model results were not available at the time of publication, and a simplified, conservative model was used to generate the figures below. Peak temperatures are therefore shown as a step function and are not time-dependent.

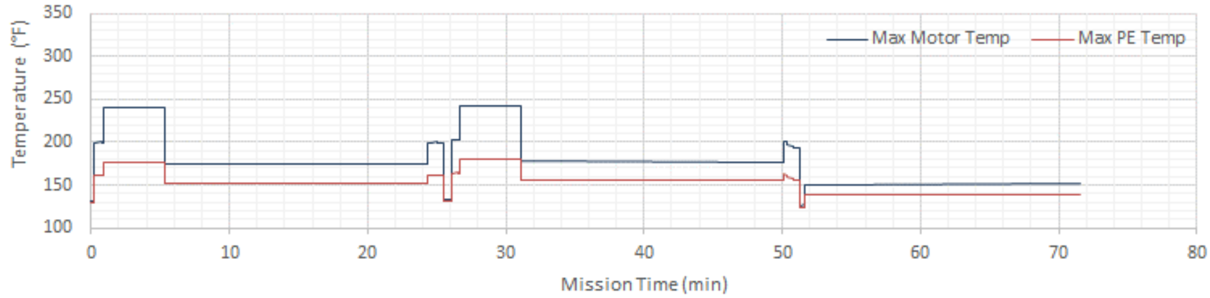


Figure 74: Motor/PE Maximum Temperature (Oil Cooled/Ram Driven)

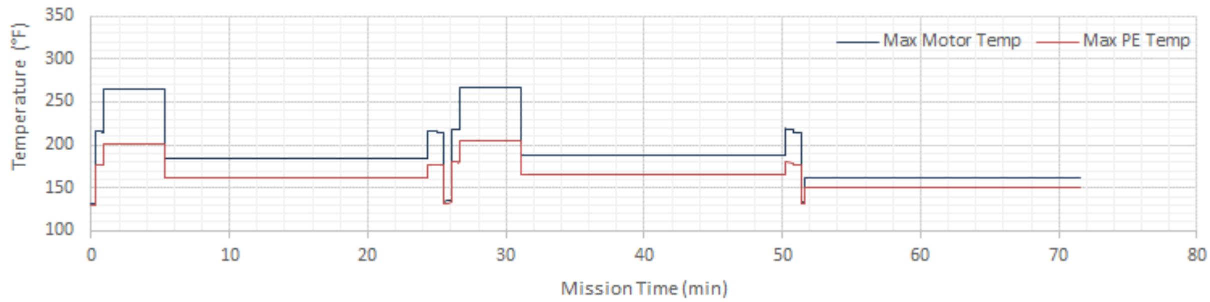


Figure 75: Motor/PE Maximum Temperature (Oil Cooled/Fan Driven)

The motor and PE maximum temperature profiles above show a similar response to the power profile of the vehicle, with the fan cooled configuration running slightly hotter than the ram cooled configuration. Due to thermal mass of the cooling system, oil temperature changes lag behind any sudden motor and PE temperature changes, which are more closely tied to the power demand. For example, sudden drops in the power demand are reflected very quickly as motor and PE temperature drops. However, the oil temperature may still be increasing from the power demand a few minutes prior, before rebounding due to the sudden drop in component temperatures. The cooling system thermal mass and heat exchanger cooling air both impact component temperatures. These effects should be explored further to better understand the contributions of each.

The air cooled options for the ENGINeUS 100 Smart Motor are shown schematically in Figure 76 and Figure 77 for the ram and fan cooled versions. Both options flow outside air over the outer surface of the Smart Motor, which utilizes cooling fins to improve cooling efficiency. The ram-driven option relies on the vehicle forward flight speed to push air through the annulus cooling passage described above. This option requires a forward facing air inlet. Cooling flow and velocity will vary with vehicle speed and will include periods when little to no flow is provided, as shown in Figure 78. The fan-driven option includes a fan that draws outside air through an annulus between the motor and a shroud or cover. The fan will provide a constant volumetric flow rate of cooling air, and thus a constant velocity through the cooling fins, as shown in Figure 79.

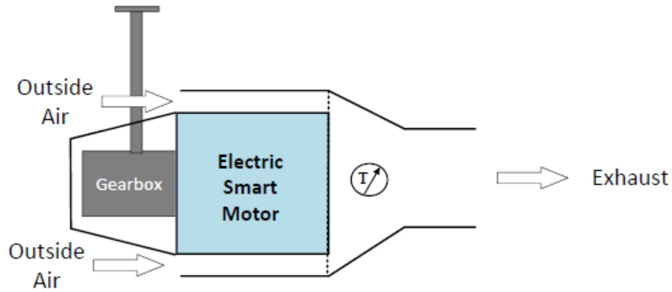


Figure 76: Air Cooled Motor, Ram Option

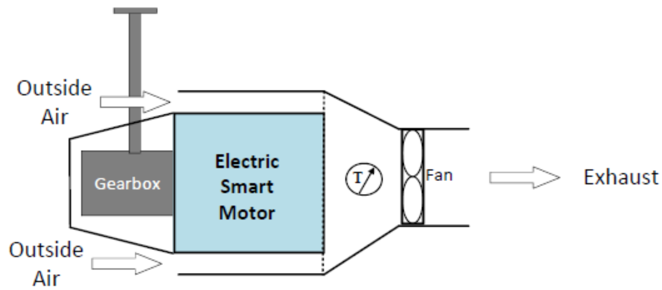


Figure 77: Air Cooled Motor, Fan Option

Safran performed the detailed cooling analysis to predict internal component temperatures, varying cooling flow with vehicle speed for the ram option and using a fixed cooling flow for the fan option. The resulting internal temperatures are shown in Figure 80 and Figure 81.

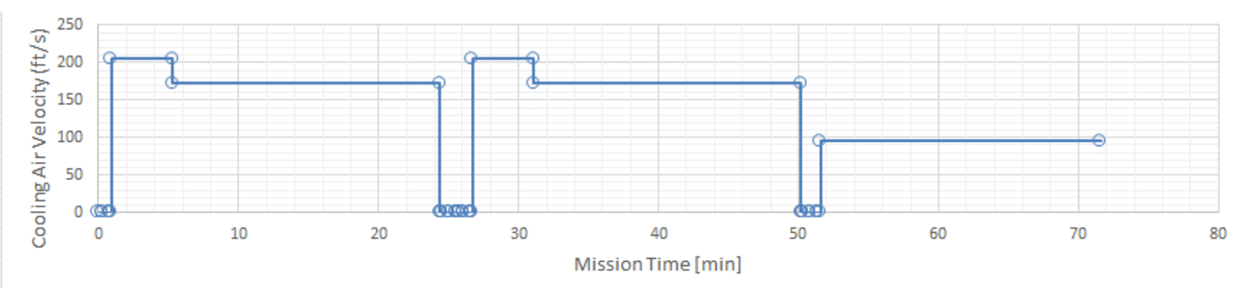


Figure 78: Cooling Air Velocity (Air Cooled/Ram-Driven)

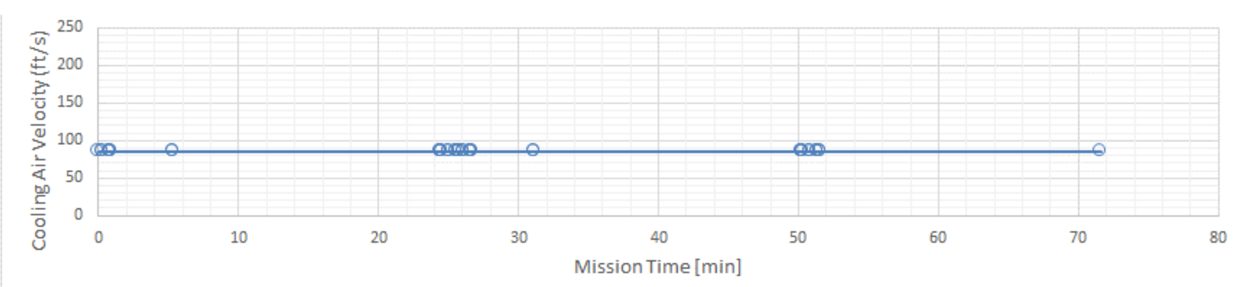


Figure 79: Cooling Air Velocity (Air Cooled/Fan-Driven)

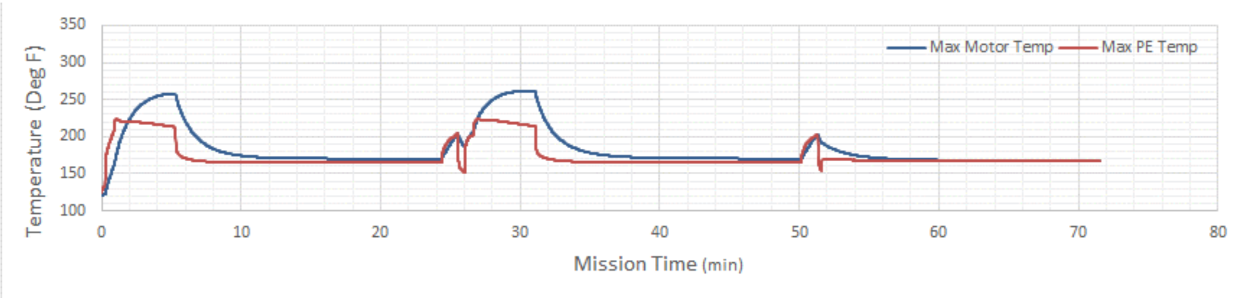


Figure 80: Motor/PE Maximum Temperature (Air Cooled/Ram-Driven)

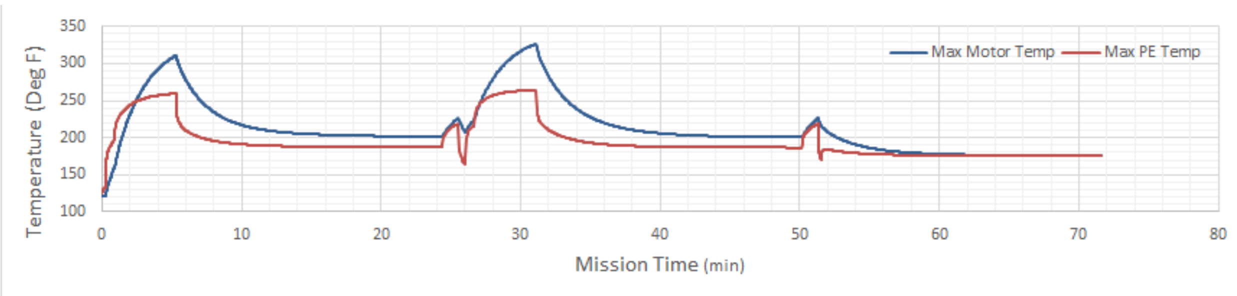


Figure 81: Motor/PE Maximum Temperature (Air Cooled/Fan-Driven)

Similar to the oil cooled analysis, the motor and PE maximum temperature profiles follow the power requirement profile of the vehicle. For both ram- and fan-driven cooling flow, the peak temperatures occur at the end of the power transients. During the high power climb segments of the mission, the temperature increase for the ram cooled components was much less than for the fan cooled option. The high power increase was balanced by the increase in cooling flow, which was especially noticeable for the PE temperatures. Throughout the mission, the maximum temperatures for the ram cooled option were considerably lower than those of the fan cooled option. The relatively long segments of low power cruise, with high velocity cooling flow, helped to offset the temperature increase experienced during the high power transients with no cooling flow. A larger fan may be able to replicate the effects of ram cooling flow, and should be explored further.

10.3 Summary and Conclusions

Table 15 summarizes the results of the thermal management system trade study. Overall recommendations are highlighted in green, and those with potentially greater design challenges are highlighted in yellow.

Table 15: Trade Study Comparison

Criteria	Oil Cooled/ Ram-Driven	Oil Cooled/ Fan-Driven	Air Cooled/ Ram-Driven	Air Cooled/ Fan-Driven
Thermal Results	High Power Climb • Max Motor 240°F • Max PE 175°F Low Power Cruise • Max Motor 175°F • Max PE 150°F	High Power Climb • Max Motor 265°F • Max PE 200°F Low Power Cruise • Max Motor 185°F • Max PE 160°F	High Power Climb • Max Motor 260°F • Max PE 225°F Low Power Cruise • Max Motor 170°F • Max PE 165°F	High Power Climb • Max Motor 325°F • Max PE 265°F Low Power Cruise • Max Motor 200°F • Max PE 185°F
Reliability	• Lowest component temperatures throughout mission will improve reliability • Fewer parts to factor into overall vehicle reliability	• Additional fan/wiring lowers overall system reliability	• Air cooled motor creates potential for hot spots in windings and reliability degradation	• Highest component temperatures throughout mission will degrade reliability • Air cooled motor creates potential for hot spots in windings and reliability degradation • Additional fan/wiring lowers overall system reliability
Weight Delta (relative to Air Cooled/Ram-Driven system)	• +8.1 lbm heat exchanger • +48.2 lbm MIL-PRF-7808 • Increased pump size • Additional weight for tubing	• + 8.1 lbm heat exchanger • + 48.2 lbm MIL-PRF-7808 • + 13.5 lbm electric fan • Additional weight for fan plenum • Increased pump size • Additional weight for wiring/tubing	• Weight for shroud, air inlet/exhaust ducting	• +7.7 lbm electric fan • Additional weight for fan plenum • Additional weight for wiring
Engineering Assessment				
Cooling	• Highly effective cooling in flight, but system needs to withstand intermittent high temperature excursions when velocity drops • Thermal mass of cooling system affects transients • Peak component temperatures are significantly lower than air-cooled systems; higher peak power ratings may be possible	• Sustained cooling; no sudden high oil temperature excursions due to low velocity flight/hover. • Thermal mass of cooling system affects transients • Peak component temperatures are significantly lower than air-cooled systems; higher peak power ratings may be possible	• Highly effective cooling in flight, but system needs to withstand intermittent high temperature excursions when velocity drops • High velocity cooling during cruise offsets temperature increase during high power transient with low cooling flow	• Similar behavior to ram -driven system, but peak temperatures during climb are higher (larger fan could compensate for this)
Limitations	• Forward facing inlet required (higher drag penalty) • No cooling in hover • Limited cooling at low speed	• Additional power needed to run fan	• Forward facing inlet required (higher drag penalty) • No cooling in hover • Limited cooling at low speed	• Additional power needed to run fan
Benefits	• Best cooling performance • Higher peak power ratings may be possible	• Stable cooling throughout mission • Higher peak power ratings may be possible	• Lowest weight impact	• Stable cooling throughout mission

11 ELECTRIC POWER AND DISTRIBUTION SYSTEM

Distributed Electric / Hybrid-Electric Propulsion of Urban Air Mobility (UAM) applications offers the benefits of low or zero carbon emissions, low noise, and reduced auto traffic congestion in urban areas, among other benefits. Safety and reliability of the HV battery and power distribution systems is critical and some of the electrical system and component areas require improvements to existing technology. Many variations exist in the public domain and other industries. DE/HEP typically includes an energy storage system, electrical inverters or rectifiers, electric machinery (motors and generators), and an affiliated power distribution system.

The electrical energy may be stored in the form of batteries, capacitors, liquid fuels, gaseous fuels, other future technologies, or a hybrid combination of these. The electrified aircraft configurations in this study leveraged Li-ion battery technology, which does not yet meet the performance requirements needed in the UAM market, but it is the best available technology as a baseline for our analysis and assumptions. Li-ion battery technology is estimated to need at least 2-3 times the energy density of current commercial technology to be feasible in the UAM market (ref. 4). In addition to new battery technology, there may be more efficient ways of storing energy than batteries. While battery technology offers zero operating emissions, battery weight does not decrease during flight like liquid or gaseous fuel consumption. Unlike a fuel and engine system, batteries are a closed loop operation and no mass is released during electrochemical conversion reactions that generate electricity. Other alternative technologies, such as hydrogen-based fuels, not only offer zero carbon operating emissions, but also provide the additional benefit of continuous vehicle weight-reduction during mission operations due to electrochemical conversion reactions that generate and emit water during that can be released overboard. Of course, the vehicle gross weight and energy storage capacity is very sensitive to mission radius, so for shorter missions the aircraft will be less sensitive to the decrease in weight.

This section shows the Li-ion battery based electrical designs completed to support other subsystem evaluation, highlight potential trade spaces for additional work, and provide guidance for future system design and technology requirements.

11.1 Conceptual Electrical System Layouts

Conceptual electrical system layouts were developed for each aircraft configuration (all except the tQuad) to show the HVDC Li-ion battery network interconnectivity with other systems. Battery "packs" are representative of several HVDC components required to deliver electrical energy, e.g., battery cells or modules, connectors, contactors, wiring, battery management system, thermal management system, etc.

The following layouts are the first step in subsystem design that will need to be refined as technologies improve (e.g., battery energy density) and reliability predictions are substantiated. Additional work is needed to develop HV battery systems and power distribution to a component level such that more quantitative and accurate reliability analysis may be applied to the system. The HVDC management and power distribution design is heavily dependent on the battery sizing. For example, cabling, contactors, active and passive safety hardware devices, and the battery management system require detailed battery design and requirements. In this current scope of work, the NDARC models rely on batteries with energy densities that do not exist in current commercial production or pre-production technology. Without development of the battery design itself, sizing and selection of power distribution components is not possible for quantitative analysis.

The main purpose of the layouts is to demonstrate redundancy management architectures for other systems (e.g., power to ESCs and FCS Channels) and associated redundancy in the fault tree (FT) models for the PSSA.

11.1.1 eQuad Electrical Layout

The eQuad electrical system is configured with eight total individual battery packs, where each pack is electrically isolated and encompasses all major components (e.g., cells, modules, connectors, contactors, diodes, monitoring and control systems, packaging, etc.). The packs are distributed such that two battery packs are located close to each motor to minimize wiring weight and physically separate the batteries for a redundant architecture, Figure 82. Two redundant packs per motor was selected for two reasons: (1) to provide a conservative assumption of improved reliability to match that of the motors and (2) to provide six completely isolated sources of LVDC power for critical control functions (i.e. FCCs and ESCs). At each motor, the dual redundant packs may power both ESCs in the event of loss of one pack. As shown in Figure 82, out of the eight packs, six deliver a step-down voltage via a DC/DC converter to provide redundant LVDC power to the FCS channels. The DC/DC converters are connected to two fwd packs (Packs 1A, 2A) and the four aft packs (Packs 3A, 3B, 4A, and 4B). This configuration shows physical separation of the LVDC power sources, in which each FCS channel is connected to one pack in the fwd and one aft: FCS Channel 1 has redundant power from battery Packs 1A and 4A, Channel 2 from Packs 2A and 3A, and Channel 3 from Packs 3B and 4B. Three hydraulic pumps are connected to each pitch control actuator, as described in Section 8. The eQuad *without* interconnecting cross shafts uses the same battery pack configuration.

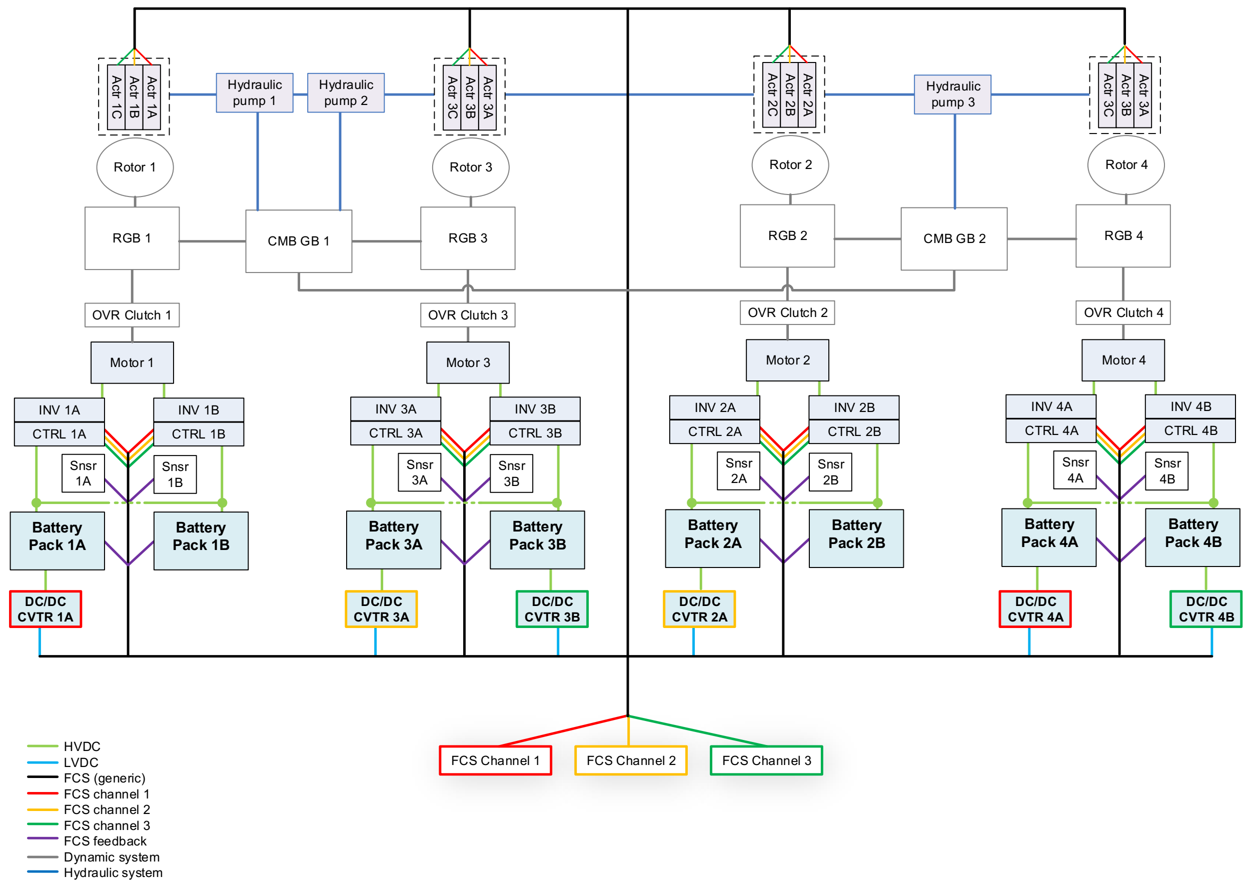


Figure 82: eQuad Electrical System Layout

11.1.2 hQuad Electrical Layout

The hQuad electrical system is configured with a turbogenerator (Fuel System, Engine, Gearbox, and Generator) as the primary source of HVDC power to the motors. The hQuad also has eight individual battery packs and six DC/DC converters distributed in the same fashion as the eQuad; however, the battery packs in the hQuad are only utilized for ground operations and in the event of a turbogenerator failure. The HVDC line shows interconnectivity of the turbogenerator power to the ESCs and nodes where each pack can connect to deliver emergency power, shown in Figure 83. In the hQuad, two redundant packs per motor was leveraged from the eQuad to provide six completely isolated sources of LVDC power. This configuration shows the same physical separation of the LVDC power sources as the eQuad, in which each FCS channel is connected to one pack in the fwd and one aft: FCS Channel 1 has redundant power from Battery Packs 1A and 4A, Channel 2 from Packs 2A and 3A, and Channel 3 from Packs 3B and 4B.

While the hQuad has a similar 8 battery and 6 converter layout as the eQuad, each pack is less than 1/10th the size of the eQuad batteries based on the NDARC models.

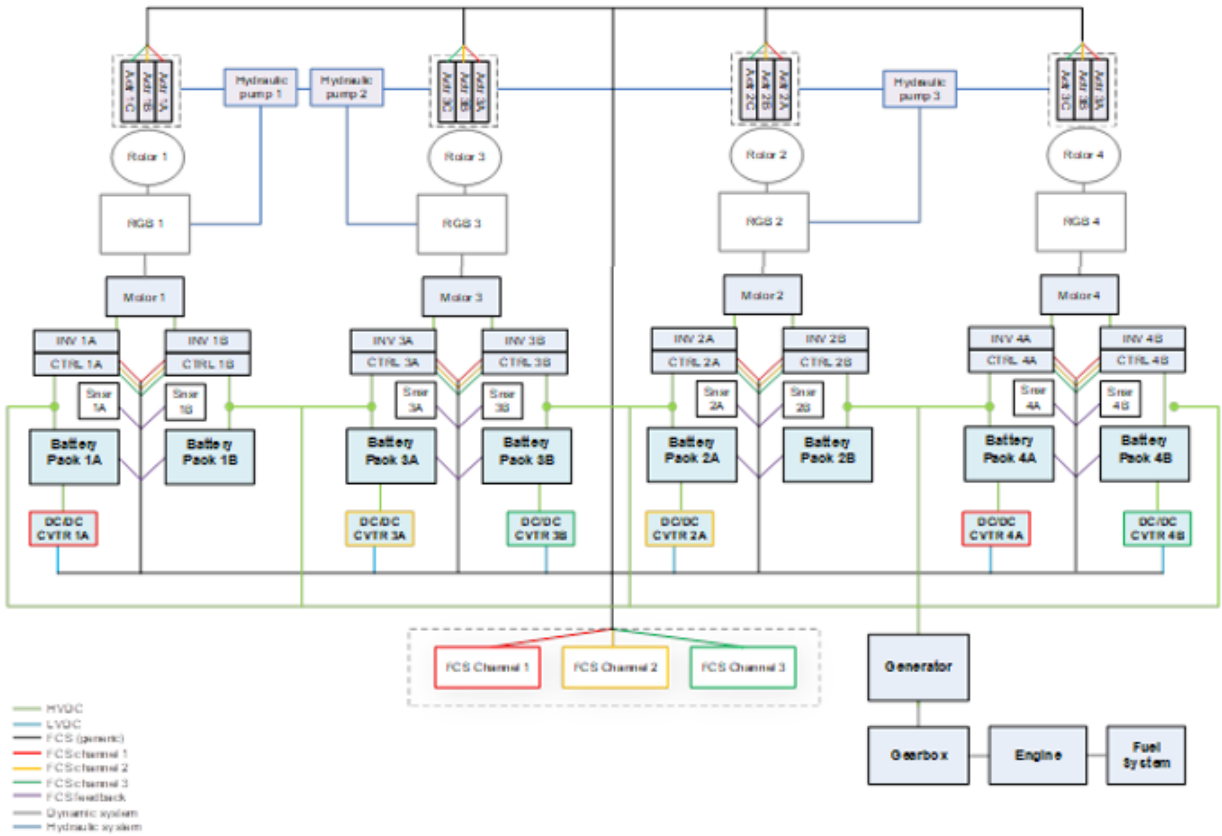


Figure 83: hQuad Electrical System Layout

11.1.3 Hexarotor Electrical Layouts

The electrical systems for the pitch controlled and RPM-controlled hexarotors are identical, see Figure 84 for the pitch-controlled layout. The hexarotor electrical system is configured with six total individual battery packs, where each pack is electrically isolated and encompasses all major components (e.g., cells, modules, connectors, contactors, diodes, monitoring and control systems, packaging, etc.). It was assumed that twelve packs may not be practical due to weight increase of battery packaging. The packs are distributed such that only one battery pack is located close to each motor, which minimizes wiring weight. A single pack per rotor requires additional development for safety and reliability budgets to be similar to that of the quadrotor configurations. Several design trades should be further evaluated for the reliability and safety implications of this layout. A few, non-exhaustive examples include:

- Dividing batteries into more packs which will increase weight due to battery packaging but could create local, fail-safe systems;
- Determine battery sizing needs associated with higher load motors (e.g., larger packs for aft motors may be required due to differences in load profile from middle or forward motors);
- Develop and compare interconnectivity between batteries to provide redundancy to motors via multiple packs already included in the layout (e.g., Battery Packs 2 through

6 can provide backup power to Motor 1 in the event of loss of Battery Pack 1 functionality)

This layout is used to demonstrate the same logic for the six redundant, isolated LVDC sources for the FCS channels. In the hexarotor configurations, each pack provides a completely isolated source of LVDC power. As shown in Figure 84, each pack delivers LVDC power to the FCS through a DC/DC converter. This configuration shows the similar physical separation of the LVDC power sources as the quadrotors, in which each FCS channel is connected to one pack in the fwd and one aft: FCS Channel 1 has redundant power from Battery Packs 1 and 4, Channel 2 from Packs 2 and 3, and Channel 3 from Packs 5 and 6. The RPM-controlled Hexarotor uses the same battery pack configuration and the pitch-controlled Hexarotor shown in Figure 84.

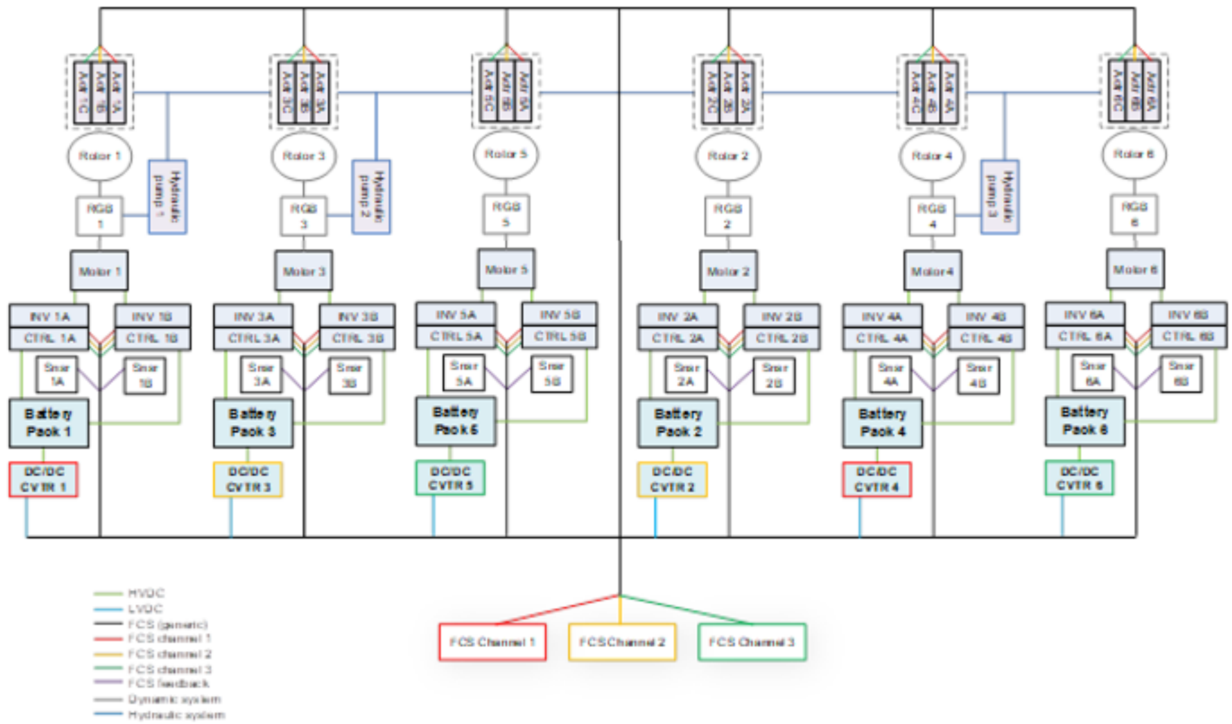


Figure 84: Pitch-Hex Electrical System Layout

11.1.4 Octorotor Electrical Layout

The octorotor electrical system is configured with eight total individual battery packs, where each pack is electrically isolated and encompasses all major components (e.g., cells, modules, connectors, contactors, diodes, monitoring and control systems, packaging, etc.). The octorotor electrical system is shown in Figure 85, in which ESCs, motors, gearboxes, and sensors are not shown. It is assumed that sixteen packs may not be practical due to weight increase of battery packaging. Like the hexarotor configuration, the packs are distributed such that only one battery pack is located close to each motor. A single pack per rotor requires additional development for safety and reliability budgets. The trade between dividing batteries into smaller packs for reliability and the weight increase due to battery packaging associated increasing pack count from eight to sixteen must be further evaluated.

This layout is used to demonstrate the same logic for the six redundant, isolated LVDC sources for the FCS channels. In the octorotor configuration, each pack provides a completely isolated source of LVDC power. As shown in Figure 85, each pack delivers LVDC power to the FCS through a DC/DC converter. This configuration shows the similar physical separation of the LVDC power sources as the quad and hexarotors, in which each FCS channel is connected to one pack in the fwd and one aft: FCS Channel 1 has redundant power from Battery Packs 1 and 6, Channel 2 from Packs 5 and 2, and Channel 3 from Packs 7 and 8. Other combinations of fwd/aft packs should be evaluated based on the environmental conditions to minimize impact to batteries, but this configuration represents LVDC redundancy management for the FCS.

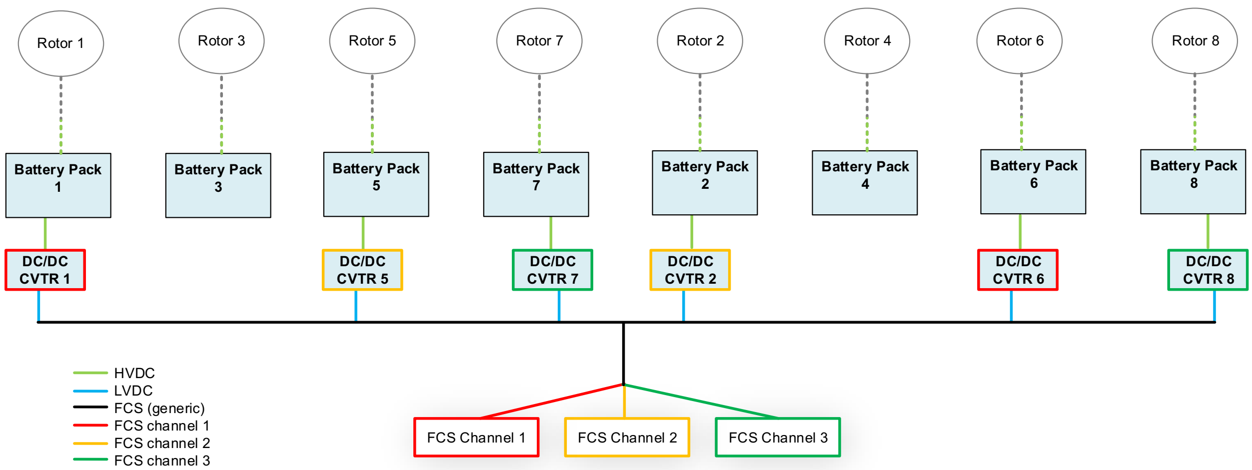


Figure 85: Octorotor Electrical System Layout*

11.2 Trade Studies

In developing the electrical system layouts, several trade studies were conducted to qualitatively understand safety and reliability relative to the conceptual system layouts. The electrical systems discussed in Section 11.1 are based on current technologies and the following subsections discuss trends in safety and reliability of these systems in the context of vehicle configurations that this study models (except the tQuad) and the UAM market as well as focus areas for future work.

11.2.1 High Voltage/Current Impacts to Hardware Reliability

There are several benefits to increasing voltage in the electrification of aircraft, but the optimal voltage for each configuration was not identified in this effort and should be evaluated in future work. Trends in current technology, agnostic to vehicles configuration, are discussed below to highlight the impacts of higher voltages and currents on hardware reliability.

The flow of electrical current generates heat due to Joule heating (I^2R), especially in the motor windings. Therefore, equipment and wiring will have electrical current limits and additional weight may be required, e.g., insulation or thermal management equipment.

Increasing the system voltage will reduce current. This offers several benefits to electrical components, for example: reduced distribution conductor size, reduced wire weight, reduced bus contactor and motor inverter switching device sizes (e.g., MOSFETs and IGBTs), reduced

Joule/resistive heating losses (I^2R) therefore improving efficiency and reducing the need for cooling equipment. In addition, higher voltages will impact the detailed design of the battery and reduce the current draw on the batteries. Reducing battery heating from high currents reduces the need for thermal management equipment as well as maintaining or preserving battery life. An increase in voltage requires more cells or modules to be connected in series (see the Section 11.2.2). When designing the battery system, appropriate allocations for safety and reliability targets are necessary at the cell, string, and pack level.

On the other hand, higher voltage can have potential negative impacts to power distribution components. Higher line voltage increases motor voltage spikes that can breakdown motor winding insulation (epoxy, enamel, polymers/polyamides etc.) causing unwanted discharge, including corona. Use of higher voltage drives the need for improved insulation technology (i.e., high grade corona resistant materials) or thicker insulation which could increase weight. Higher voltages will break down or consume wire insulation over time. For example, a 270 VDC line voltage may cause 350-420V at motor windings; whereas a 700 VDC line voltage may cause 1.2-1.4kV at motor windings. Since the ESCs create voltage transients above the line voltage, good insulation, filtering, shielding, and high reliability drive components are required. In addition, prognostics and health management for insulation breakdown is highly desirable.

In general, the design should evaluate the breakdown voltage margins electrodes/wires that is also the product of the pressure (altitude) and distance between electrodes/wires, referred to as Paschen's Law. At standard atmospheric conditions, this breakdown voltage is approximately 327 volts with bare electrodes. The use of higher voltage systems in smaller UAM aircraft also presents a challenge to maintaining adequate wire separation. Wiring between motor controllers and motors requires EMI shielding, which adds weight, and should require EMI emissions testing (e.g., MIL-STD-461, DO-160) and susceptibilities adjustments.

Moreover, the weight savings associated with compact vehicle HV systems may not be as significant as found in larger vehicles (e.g., 787). For larger aircraft, higher voltages and the benefit of wire weight savings can be significant; however, 270 VDC is considered a reasonable starting point for UAM vehicle distribution voltage. For example, the reliability of wiring, switching and terminal design for 270V systems is well established with the evolution of the 787, F-35 and Air Force INVENT programs. 270 VDC can also provide acceptable motor winding reliability.

The main factors decreasing equipment reliability include:

- Thermal cycling with large temperature extremes
    - Differential mechanical thermal expansion/contraction fatigue of components
    - Electronic device overheating
    - Transistor breakdown as temperature goes higher (thresholds lower as temp increases)
- Transistor overvoltage and spikes (shoot through)
- Vibration

11.2.2 Battery Feasibility and Future Considerations

State-of-the-art Li-ion battery packs pose engineering challenges before being a widely available, viable energy storage system for UAM. Previous work highlights the minimum energy density requirements for RVLT concept vehicles under study at the cell and pack level (600 and 450 Wh/kg, respectively) (ref. 4) which is far in excess of that of commercial technologies (less than

300 Wh/kg at the cell level). The NDARC models assume the minimum energy density required, therefore current technology does not exist to design the actual batteries (e.g., series and parallel cell configurations, packaging, battery management systems, active and passive safety hardware and software, etc.) for this study. A few major considerations which must be accounted for in future work.

In this study, it was assumed that batteries would only be discharging in flight within the nominal voltage window to deliver nominal output power. Two future considerations would be (i) the possibility for in-flight re-charging (e.g., in the hybrid-electric) and (ii) more accurate evaluation of battery discharge cycle output in modeling and analysis. For example, the all-electric configurations shown several flight profiles in the NDARC models for a “single” discharge of the batteries. Current battery technologies drop in voltage over a discharge cycle, therefore there may need to be additional safety margin to ensure the voltage dropping during the end of discharge (e.g., the last flight before the batteries require charging) must provide sufficient voltage and reliability power to the ESCs/motors. Additional work is needed to account for the battery output at the end of discharge in which voltage continues to decrease, causing current to increase to match the power output, and more heat is generated.

The battery pack arrangements (e.g., number of cells, series and parallel strings, etc.) were assessed for each air vehicle to demonstrate feasibility and highlight focus areas for future work, shown in Table 16. Ignoring the weight penalty of commercial off the shelf (COTS) lower energy density, cylindrical COTS cells (size 18650) in electric vehicles were used in this evaluation because there is publicly available information; however, other higher pre-production Li-ion chemistries and other form factors (e.g., Pouch cells or larger format cylindrical cells) may be beneficial to design and reducing number of cells. Table 16 contains the total energy storage capacity dictated by the NDARC models, the number of packs based on the conceptual layouts from Section 1.2, and the resulting C-rate calculated from the peak loading from the NDARC models. The peak C-rates do not account for emergency conditions in which loss of one pack would increase the load on another pack or packs; however, the design assumptions were that the ESCs were fully redundant and each pack would be sized to provide adequate power and energy after the loss of one ESC.

Table 16: Battery Energy Capacity and Peak C-rates (in accordance with ref. 4)

	eQuad	**hQuad**	**Pitch-Hex**	**RPM-Hex**	**RPM-Oct**
Battery System Energy	1325 MJ (368 kWh)	100 MJ (28 kWh)	1487 MJ (413 kWh)	1390 MJ (386 kWh)	1636 MJ (455 kWh)
No. of Packs & Pack Energy	8 @ 46 kWh	8 @ 3 kWh	6 @ 69 kWh	6 @ 64 kWh	8 @ 57 kWh
Peak C-rate	1.5	11.6	1.5	1.4	1.4

The C-rate is a metric that represents the ratio of discharge current to nominal rated discharge current that would discharge the battery in 1 hour. For example, a 1C discharge rate would discharge the rated battery capacity in 1 hour, 2C would be discharge the rated battery capacity in 0.5 hours. Regardless of pack design or cell chemistry, this value shows that the fully electric aircraft configurations would have less than a 2C rate even at peak loading. This is within anticipated discharge rates for eVTOL batteries of ~2-4C (ref. 4).

While the hQuad would only use the battery system in the event of turbogenerator system failure, the peak load is greater than 10C rate (which would fully discharge the battery within 6 minutes) which is not a feasible discharge rate for Li-ion. This would significantly heat the battery quickly, inducing failure modes described in Section 11.2.4, or be inoperable due to the electro-chemistry limits and internal resistance. Moreover, the mission profile would need to be evaluated to understand how much energy capacity the battery system would have based on the LVDC power draw for the FCS, Figure 83, and passenger amenities like heating, air conditioning, and charging of personal electronics. Additional work is needed to understand the reliability for hQuad battery designs for optimizing the benefit of the fully redundant back up system, recommended focus areas for future research on hybrid-electric aircraft systems:

- Review energy demands during and after a turbogenerator failure, potentially increasing energy capacity to ensure safe flight and landing and to power LVDC systems. Considerations for LVDC systems powered off of a common bus architecture with the turbogenerator system.
- Evaluate alternative chemistries for a single use, high energy/high power non rechargeable batteries (e.g., lithium-primary cells); while this would be added weight, it would be less than Li-ion weight (or increased energy stored at the same weight) with higher reliability chemistries that are not as susceptible to the Li-ion high-discharge failure modes.

In addition to the C-rate, cell configurations of series/parallel strings of COTS cells were analyzed to compare battery systems at two voltages shown in Table 17: 800 V which is beneficial for larger aircraft and 270 V which would provide sufficient benefits to UAM air vehicles, refer to Section 11.2.1. The number of cells in series defines system voltage (V) and the number of cells in parallel defines current capacity (Ah). Therefore, each air vehicle configuration will have the same number of cells in series at each system voltage (211 series for 800V systems and 71 in series for 270V systems), but the number of parallel strings with change depending on the NDARC energy storage capacity and pack sizes listed in Table 16. The total number of cells is also shown to highlight potential oversizing. For example, in the hQuad ends up with 4 strings of 71 series cells in the 270V to meet the energy storage requirement, but only 1 string of 211 cells for an 800V system meets the requirement. Lower voltage systems, thus shorter strings, may offer additional benefits regarding pack reliability, but must be investigated further along with detailed battery and electrical system design that was not conducted in this study. For example, at high voltages, there are significantly fewer strings in the hQuad such that loss of a string may render battery inoperable in an emergency scenarios.

Table 17: Example COTS (ref. 35) Cell Configurations for 800V and 270V Systems

	eQuad	hQuad	Pitch-Hex	RPM-Hex	RPM-Oct
800 V Systems					
Series/Parallel Configuration	17p211s	1p211s	25p211s	24p211s	21p211s
No. of Cells/Pack	3587	211	5275	5064	4431
270V Systems					
Series/Parallel Configuration	50p71s	4p71s	75p71s	70p71s	62p71s
No. of Cells/Pack	3550	284	5325	4970	4402
NCA cell chemistry: 3.8 V nominal and 3.4 Ah (used in Tesla EV battery modules) (ref. 35)					

In addition to detailed design of the battery, key battery focus areas were identified during this study that should be considered in future research:

- Reliability predictions and failure rate are continually updated and refined throughout an aircraft's life, typically substantiated by millions of flight hours for a mature platform or system. Failure rate predictions are based on the detailed arrangement of cells and the control and monitoring hardware and software of each individual battery design. Li-ion batteries for aircraft primary electric propulsion have very few flight hours amongst many unique battery designs. While the electric vehicle market has substantial driving hours, the battery designs are not directly translatable for use in reliability studies of aircraft propulsion batteries. This is a significant obstacle in substantiating safety and reliability claimed for electrified UAM vehicles, which may also hinder regulatory compliance (e.g., EASA SC-VTOL-01). This is a critical focus area that industry should emphasize in the future of eVTOL and electrified aircraft.
- In the conceptual electrical layouts, battery packaging was assumed to have some optimal balance between dividing the batteries into smaller modules or packs for reliability and the impact to weight increase associated with packaging. Typically ~20-30% of an individual battery pack is packaging to contain the cells, safety devices and hardware, and passive thermal management materials. Therefore, moving between the four rotor vehicles to six or eight is assumed to limit how packs could be divided and distributed. Moreover, dividing batteries into smaller packs could decrease reliability by introducing more and more components for packaging small packs. Redundancy management architectures and weight trade studies will be important to conduct with improvements in battery technology.
- Guidance from regulators is available to demonstrate safe designs through testing and installation. For example, RTCA/DO-311A provides a standardized method for verification and characterization of the safety and performance of battery systems. In addition, FAA AC 20-184 (ref. 37) offers a means of compliance, providing guidance for testing and installation of Li-ion battery systems. AC 20-184 states,

  > *The lithium battery system shall be designed to minimize the impact of self-sustained, uncontrolled increases in temperature or pressure, as a result of any failure within the battery. The probability of impact must be extremely improbable (1 event in 1 billion ($1x10^{-9}$) flight hours).*

  However, there is no publically available reference data or agreed upon value to substantiate the probability of failure. In the future, energy technologies will continue to increase in inherent risk due to higher energy/power density, therefore will require greater scrutiny in how they are managed, tested, stored, operated, etc. This includes high energy/power next generation Li-ion, beyond Li-ion (Li-S, Li-O2, etc.), or clean fuels (e.g., hydrogen).

11.2.3 Power Fluctuations as a System Design Problem

Another trade study area of interest is the potential increase in battery capacity to serve as a buffer against power demand fluctuations on the turbogenerator system. Detailed design and maturation of neighboring systems, including drive and power and thermal management, is needed to quantitatively determine the potential reliability impacts of increasing the battery capacity to buffer against power demand fluctuations. Industry trends and available electrical system design practices provide a means to guide UAM designs. The basic components of a hybrid-electric system include:

- Turbine engine driven AC generators
- AC-to-DC power converters
- DC power buses
- Brushless "DC" motor controllers (MOSFET or IGBT type inverters)

In this concept, there is an energy storage device, a super capacitor or battery, to serve as the buffer within the DC portion of the overall power system. There would be two basic system options to address power demand fluctuation, or "droop":

- Increase size of the generators and AC-DC converters to minimize DC power droops and buy reliability margin
- Add super capacitors with the affiliated weight and un-reliability to the power system

A key consideration for this approach would be if the system has large regeneration ("Regen") power potential, then super capacitor technologies may be well suited for this approach. This can be used to support transient power demands and reduce the generator and AC-to-DC converter size.

There are three fundamental conditions where power generation occurs.

- Normal No Load and Opposition Load conditions. Under these operating conditions, the motor rotation creates back electromotive force (EMF voltage), which is a function of motor speed. Under a no load condition, the motor will spin up in speed until the back EMF voltage (plus losses) is equal to the supply voltage. Under a load opposing condition, the motor will spin up in speed until the back EMF voltage (plus losses) plus load voltage equals the supply voltage. This condition can be compared to a car operating with its wheels off the ground (no load) or climbing a hill (opposing load).
- Normal Load Aiding condition. Under this operating condition, the motor is acting as a traditional generator and the motor has to react the generator voltage to control speed (and avoid run away). It is under this condition, that power can be recaptured by the vehicle power subsystem and instantly used or stored in batteries and/or capacitors. This condition can be compared with a car going down a hill.
- Abnormal Load Opposing Back Driving Condition. Under this condition, the opposing electromotive force (voltage) exceeds the motor supply voltage entering a condition of stall and actual motor back driving. This condition can be compared to a car that suddenly runs into a very steep road which it does not have the power to climb, slows and then starts to descend the hill backwards. This condition should never occur from a design perspective.

Regardless of beneficial load aiding regeneration power, the overall electrical power system must be able to manage regeneration power as it is inevitable. As such, there are a number of basic methods for management:

- Convert the Regen power to heat by using Regen Resistors (pure heat). With this design approach, a trade study is needed to evaluate the various options to accommodate the thermal dissipation not only in the Regen Resistors but also the electronic controllers. For example, air or liquid cooling.

- Charge up batteries, super capacitors or battery capacitor hybrids. With this approach the motor inverter has to be designed to allow the Regen power to flow back onto the DC bus.
- Bi-directional generator (motor generator). In this case, the Regen electrical power is actually put back onto the DC bus using the motor inverter and then the high bus voltage is use to drive the motor generator as a motor (via the motor/generator converter) and put mechanical power back into the mechanical drive system (reverse torque): Motor AC to DC via inverter and then DC to AC via motor generator converter and finally reverse torque.

In the event the system is designed to recapture or instantly use the regeneration power, the system should still include a Regen Resistor and/or over voltage protection system to cover converter or inverter failure conditions and to cover events where the system cannot absorb or use the Regen power fast enough.

Given UAM helicopter type vehicles, rotor drive and actuation systems seldom operate in a power regeneration condition (load aiding) because: (1) helicopters must continuously overcome opposing gravitational load, and (2) extreme rapid descent (auto-rotation/load aiding condition) is not passenger-friendly. However, regeneration power levels in an autorotative condition can be significant and must be managed. Relative to overall reliability, similar type and quantity of components are used in both systems (larger generator vs. regen system), except the buffered system requires the use of (and unreliability) of the super capacitor components. This is a system level trade study which includes power demand and regen analysis, thermal analysis along with weight, reliability and safety considerations:

- Super capacitor technology has a low TRL
- Need to be concerned about super capacitor charging safety (as with Li-ion batteries)

11.2.4 Li-ion Battery Failure Modes and Reliability

In addition to hardware reliability and the impacts or trades with high current or high voltage systems, understanding the failure modes of Li-ion batteries is vital to the future of safe and reliable battery designs for electric propulsion. These failure modes are well-understood, but are complex with many contributing factors, such as: cell chemistry, cell design and components, duty cycle, operating temperatures, external abuse (electrical, mechanical, or thermal), etc. The following failure modes of greatest concern to safety and reliability include:

- Cell Aging (capacity loss): Internal cell components my breakdown, with undesirable chemical reactions and gaseous byproducts increasing the internal pressure. This reduces the amount of active material, therefore reduces battery capacity, reduces cycle life, and may lead to thermal runaway due to overcharge or internal short (see below).
- Internal short circuit (ISC): Two types of shorts, soft and hard internal short. A soft ISC will cause some internal heating as a small amount of current leaks internally, reducing cell life, slowly draining the cell of capacity, local heat generated internally, and cell degradation discussed above. Hard ISC is a significant amount of internal leak current between electrodes that generates excessive heating and will end up causing cell degradation, soft short effects, or an intense energetic failure known as thermal runaway (below)

- Thermal Runaway (TR): TR is a result of self-sustaining chemical reactions of cell components. These reactions generate gases and heat at a rate such that the rate of heat generation exceeds heat dissipation, thus creating the self-sustained feedback loop of reactions. The temperature and pressure build up may result in rapid cell disassembly, fire/explosion, or release of flammable or toxic gases. As one cell experiences a violent TR failure, it is critical to prevent TR propagating to other cells, as the ejected contents of the failed cell may initiate TR in adjacent cells.

Additional work is needed to design the battery systems for these air vehicle configurations for both safety and reliability; however, the system can be operating at intended voltage/current at the pack-level and still exhibit cell-level failure modes. Individual cells of a pack may age slightly faster than surrounding cells, losing capacity over time. Upon cycling, these cells will discharge (or charge) faster than surrounding cells causing internal failure modes stated above. This can result in failure conditions like over-charging, over-discharging, or increased discharge rate for individual cells that may lead to the failure modes listed above. Refer to Table 18 for detailed information regarding these localized failure conditions, noting some may result from other, external factors (e.g., excessive discharging due to overloading electrical system). Battery management is essential to monitor and control the pack, to detect, prevent, or mitigate the failure modes above. Current best practices and continuous improvements in technology will be necessary for a smart battery design to provide active and passive control, from the cell to system level, and ultimately a safe and reliability system.

Table 18: Summary of Internal Failure Conditions of Li-ion Batteries

	Definition	Internal Effects	Failure Mode	Mitigations
Over-charge	Continue charging cell above voltage limits (~4.2 V*). Limits based on cell materials and supplier definition of 100% SOC. e.g., excess of 5V or 150% SOC, which requires a current source for long period of time	Electrolysis of electrolyte: - Produces gas and heat	Heat and pressure generated which can contribute to thermal runaway	**Active** Charge Control: - State-of-Charge (SoC) detection - Limit SoC below 100% Charge Power Control: - Overvoltage protection to disconnect cell **Passive** Pressure activated: - Current Interrupt Device/pressure fuse (permanently disconnect cell) - Cell vent (cell remains connected) Future cell developments (e.g., separator and electrolyte additives)
		Lithium plating (vs intercalation) at anode: - Generate nodes for possible dendrite formation - Decreases activation temperature of thermal runaway - Exothermic reactions	Latent failure	
		Cathode delithiation causing structural instability: - Triggers spontaneous exothermic breakdown; gases produced from side reactions and heat generated from increased internal resistance as well as side reactions - Heat generation is self-sustained as separator breakdown, introducing internal shorts and associated heating).	Thermal runaway	
		- Loss of active cathode material & increased impedance	Reduced cycle life	
	Overvoltage: Continue charging cell above voltage limits, but cathode not fully delithiated	Electrolyte decomposition and side reactions, producing excessive gas	Release of pressurized, hot, and potentially flammable or toxic gasses	Active overvoltage protection and passive mitigations above.
Over-discharge	Discharging cell to voltage below limit (~3.3V*)	Generate nodes for lithium dendrites: e.g., breakdown of anode current collector and plating at cathode upon subsequent cycling - Increased impedance	Latent internal short circuit failure	**Active discharge control** - SoC detection - Maintain minimum of 10% SoC Future cell developments (e.g., electrode and electrolyte materials)
		Dissolution of SEI produces gas, which consumes Li to reform upon subsequent cycling	Reduced cycle life	
		In multi-cell strings, possible cell "reversal," begins charging	Reduced cycle life	
High-rate Discharging	Discharging at faster rate than cell limitations	- Internal (resistive) heating which may lead to decomposition of cell components - Separator shrinkage could cause internal short circuit	Thermal runaway Reduced cycle life	Future cell developments (e.g., separators) **Active discharge control** - SoC detection - Limit SoC to 10% minimum **Passive**: - Positive Temperature Coefficient (PTC) Resistor (disconnects cell)
		- Cell component instability/degradation (e.g., loss of cathode active material)	Reduced cycle life	

*Based on COTS Li-ion 18650 cells

For Li-ion chemistry and information discussed above: “Linden’s Handbook of Batteries” 5th ed and “Electrochemical Power Sources: Fundamental, Systems, and Applications Li-Battery Safety”

11.2.5 Human Safety

In addition to hardware reliability and safety, implementation of high voltage/current systems presents inherent electrical hazards to people. At high voltage, adequate insulation is necessary to protect humans from contacting high power wiring. Moreover, currents as low as 10 mA may cause death.

OSHA standards and safe handling guidelines are available for those maintaining these systems. Relevant SAE safety standards for electric vehicle and electrified aircraft, such as SAE J2344, note potential gaps or considerations between automotive and aircraft.

The following subsections highlight some specific devices and guidance for human safety in design of high voltage and high current systems.

11.2.5.1 High Voltage Safety

Several safety design elements exist to protect humans from electrical hazards associated with high voltage. In all cases, adequate insulation is required.

- Dielectric withstanding and insulation resistance testing during build up.
- Warning and Caution placards in test procedures and on hardware/aircraft.
- It is common practice in electric vehicles and demonstrator electric aircraft for battery packs designed as sub-packs or modules of lower voltages (<60 VDC) for handling during maintenance.
- SAE J3108 contains high voltage safety info for electric vehicle labels to assist first and second responders and others.

11.2.5.2 High Current Safety

Several safety devices exists to protect humans from electrical hazard associated with high current. Circuit breakers (CBs) are a common electrical design component that break (open) the affected circuit in the event of current exceeding a threshold that is determined by the CB device. This prevents any current flow across that CB, but the threshold can still be a high current. CBs are devices primarily to protect electrical wiring and hardware components, not people.

Solid State Power Controllers (SSPCs) replace traditional electromechanical CB relays and thermal circuit breakers in power distribution systems, offering more accurate trip protection with solid-state reliability, while reducing overall vehicle-level weight. A SSPC is a smart devices that control voltage and/or current supplied to a load. They are more reliable and faster than electronic CBs in powering off to protect loads from dangerous faults, while CBs only trip above a certain current threshold. SSPC devices monitor overload conditions and can prevent short circuits. As such, the SSPCs are becoming state of the art in aerospace application.

The Ground Fault Current Interrupter (GFCI) is a circuit-interrupting device that is designed to protect high currents from conducting through people. GFCI devices compare the positive leg current to the negative leg current to detect ground fault current. GFCI devices help prevent human deaths as the GFCI detects abnormal current flowing in the circuit legs (e.g., due to contact with a person or water). In example, GFCI devices are used in households where nearby water sources (e.g., bathroom or kitchen) or cement flooring (e.g., basement or garage) are found. The GFCI outlet will trip and cut off power if human interference (conducting to ground, via water or concrete) to prevent a deadly electrical shock.

Arc Fault Interrupters (AFIs) are used to open the electrical circuit to prevent arcing fires caused by shorted circuits when the current frequency produces a unique arcing signature (bandwidth frequency). High frequency converters (e.g., motor controllers) can create high frequency signatures on the HVDC bus. As such, the AFIs can inadvertently false trip. Rigorous analysis and testing must be conducted to ensure the AFI risk reduction outweighs the inadvertent trip risk increase. It is also important to note that SSPC devices can incorporate both GFCI and AFI features. In addition to the devices mentioned above, proper grounding of electrical components is required.

11.3 Summary and Conclusions

This scope of design work did not include detailed development of the HVDC system, batteries, or other electrical system components. The HVDC concepts described should be used as stepping stones to more detailed design and analysis which are crucial next steps to validate reliability predictions through design, analysis, and test. In addition to HVDC system development, other trade spaces should be addressed in future work.

11.3.1 HVDC System Architecture

The conceptual electrical system layouts discussed were generated to support the fault tree analyses, Section 13; however, additional design is required to support further development of the RVLT propulsion architectures and specifically the buildup of HVDC reliability and safety analysis. In this study, no interconnectivity between battery packs across motors is considered, but only as individual "HVDC systems" per motor. This has several implications that must be evaluated for detailed design which may show required interconnectivity to meet safety targets.

Connecting packs across motors may be a more pragmatic design concept for redundancy than the values assumed for fault tree models, as discussed in Section 13.3.1. In addition to a more fault tolerant architecture, this would provide a means to handle emergency peak loading in the event of a battery pack failure. Additional issues rise as configurations without cross shafting (hexarotor and octorotor configurations) do not have the underlying, mechanically driven redundancy that would be possible in cross-shafted variants.

11.3.2 Batteries

Li-ion batteries require significant improvement in energy density, but also in the associated reliability predictions. Batteries are made up of hundreds or thousands of cells which may have some reliability data, but there are unique interdependencies associated with series/parallel configurations and how the batteries are operated as an integrated system (e.g., heat conduction between cells, cell aging/degradation rates, etc.) (ref. 37). Overall, battery architecture and redundancy management is a particular focus area for future work. Examples of redundant battery/motor architectures such that each motor may be powered by two or more independent batteries are in the public domain. In one example, battery management, automatic power rerouting, physical spacing, and spatial integration are utilized in an attempt to enhance reliability and fault tolerance (ref. 38).

Battery sizing will also play a critical role in future development to ensure there is sufficient capacity safety margin for each configuration. Moreover, the configurations developed in this study assumed the configurations had batteries local to each motor. Robust testing must be considered to ensure safety and reliability under such extreme environmental conditions (e.g., DO-160).

Additional work must also be done to incorporate battery charging, as charging requirements are different than discharging. For example, this may impact the system design and operation and change the thermal management requirements (e.g., upper temperature operating limits of 60°C during discharge compared to on 45°C for charging).

11.3.3 Regulatory Guidance

Further improvements in available regulatory guidance are necessary. Quantitative demonstration of compliance with safety and reliability requirements, such as those set forth in EASA SC-VTOL-01, do not sync with the guidance for these electric and hybrid propulsion systems set forth in EASA Proposed Special Condition E-19, specifically the qualitative safety analysis of these unique electrical systems, EHPS.80 Safety Assessment (ref. 11). Reliability data is not available for these new designs and equipment associated with distributed electric propulsion systems. Some components, such as Li-ion battery, have testing and installation guidelines, such as FAA AC 20-184, as a means of compliance to demonstrate the safety of the design and installation, but there is no readily available reliability data to quantify probability of failure per SC VTOL-01 requirements.

11.3.4 Recommended Areas for Future Trade Studies

- Optimum voltage level to balance wire weight, component reliability, and barriers to entry
- New monitoring techniques to evaluate in-service electronic degradation
- Transistor reliability and transistor thermal management
- Compact/reliable filtering design and capacitor reliability
- Power regeneration trades, including bidirectional generators/converters, super capacitors and regeneration resistors
- Battery, inverter and motor location relative to battery separation, wire separation, maintainability, cooling and weight
- Lightning
- Crash worthiness
- Ditching

12 SYSTEM RELIABILITY ANALYSIS

The aircraft and associated system reliability values were found using a combination of industry data, engineering judgment, and calculation methods. For this effort, the quadrotor FMECA from prior work (ref. 5) was refined and updated for each vehicle under study. Failure rate estimates were developed for the electrical power and distribution system, flight control system, drive and power system, and thermal management system. Each unique system warranted a different analysis approach, based on data source availability and design detail, to obtain pertinent failure modes and failure rate information. System reliability estimates were then rolled back into the aircraft FMECAs and then used to support initial verification and updates to the PSSA fault tree, see Section 13.

A FMECA is a tabular document containing postulated failure modes of the propulsion system under analysis. FMECAs were performed on the NASA RVLT concept vehicles to determine system failure modes and their associated criticality through bottom-up, function analysis. This examination of the system aids in applying failure rates to the preliminary safety assessment.

The FMECA process is a methodology for comprehensively identifying the failure modes for a system or component. It progresses and matures in line with the design process and increases in detail along with the design. The FMECA process begins in the conceptual design phase with initial planning and requirements review. This lays the groundwork for the actual FMECA, itself. System models and block diagrams are typically developed in this stage to aid in the identification of functions and functional decomposition. Figure 86 shows the approximate alignment of FMECA development with the design process.

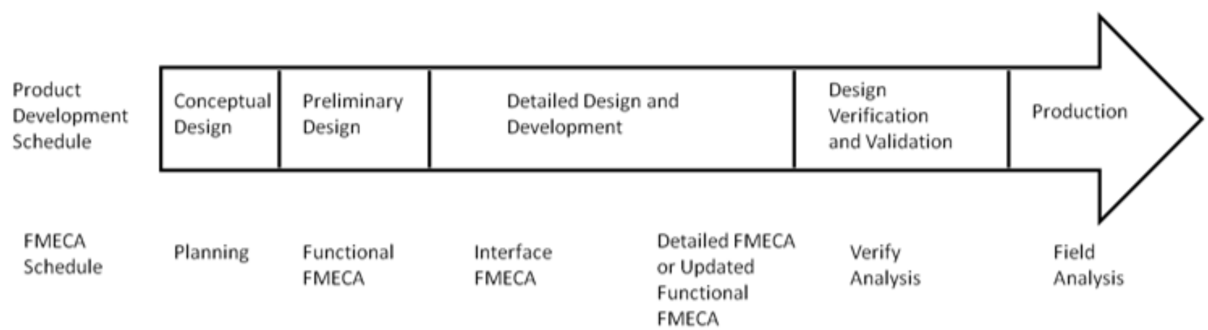


Figure 86: FMECA Development and Design Flow

Each phase of FMECA development follows a similar process after the FMECA is initially planned and functional requirements are analyzed (during the conceptual design phase). The planning and block diagrams are used to identify functions and function (one at a time). These are used to postulate failure modes for each functional failure (one at a time). These failure modes are analyzed for their consequences from which severity codes may be assigned.

Severity is divided into four categories as defined in MIL-STD-1629A (ref. 39). Severity category is assigned to provide a qualitative measure of the worst potential consequences resulting from design error or item failure. The severity classifications are described in Table 19.

Finally, criticality is determined for each failure mode. A flow chart of the FMECA development process which was adapted from SAE ARP4761 and ARP5580 (ref. 40) is shown in Figure 87. Prior work (ref. 5) determined failure rates of similar equipment, taken from various sources with applied environmental factors in accordance with MIL-HDBK-217 (ref. 41).

Table 19: Severity Classification Used in FMECA Worksheets

Category	Severity of Effect
I	Catastrophic: A failure which can cause death or system loss (i.e., aircraft, tank, missile, ship).
II	Critical: A failure which can cause severe injury, major property damage, or major system damage which will result in mission loss.
III	Marginal: A failure which may cause minor injury, minor property damage, or minor system damage which will result in delay or loss of availability or mission degradation.
IV	Minor: A failure not serious enough to cause injury, property damage, or system damage, but which will result in unscheduled maintenance or repair.

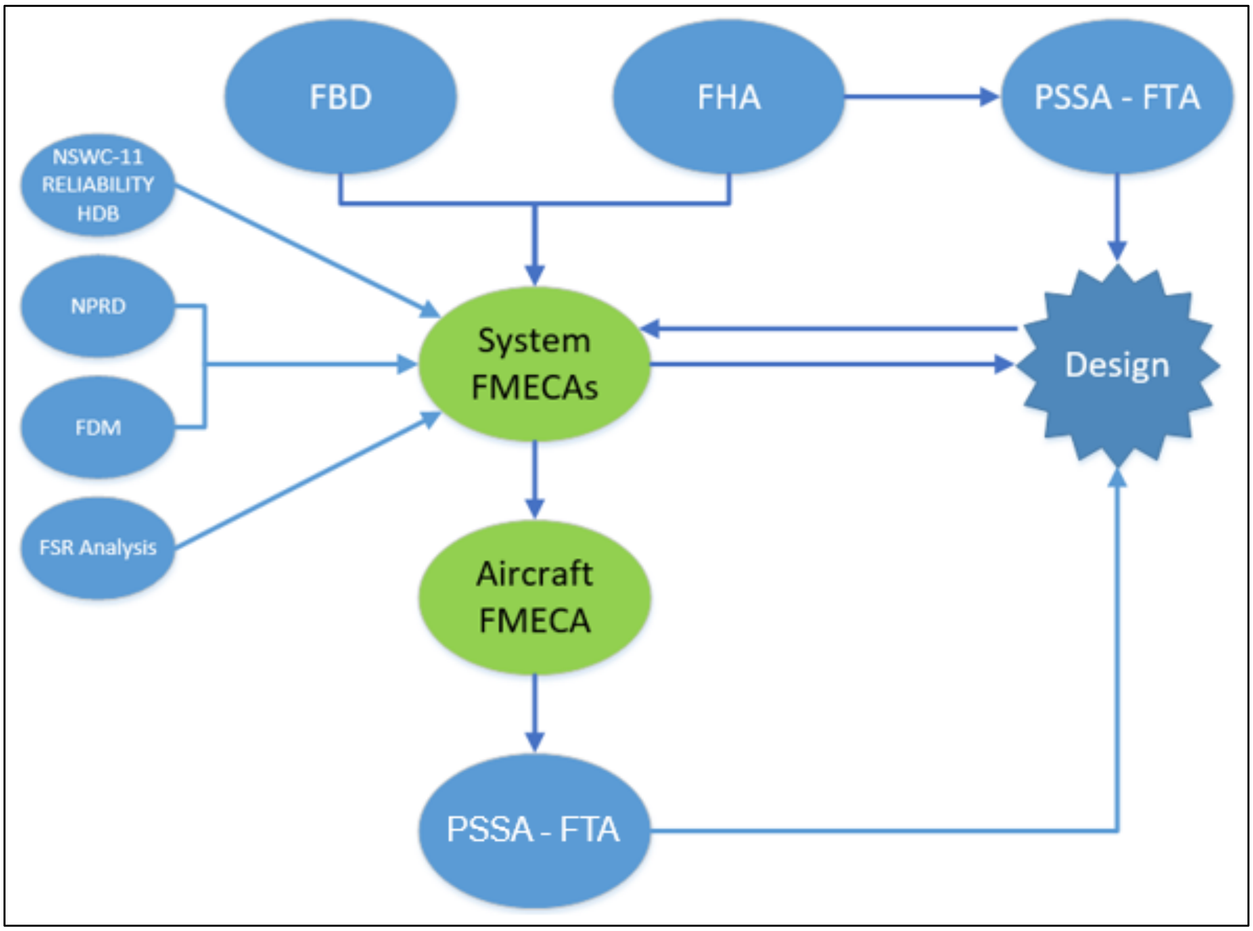


Figure 87: FMECA Development Flow Chart

12.1 Definitions of FMECA Worksheet Data Elements:

- **FMECA ID Code:** The FMECA identification is an indentured code which assigns a unique identifier for each failure mode. This is a combination of the Failure Mode Index (FMI) Function, FMI Mode, and FMI Cause, as defined here:

    - 1st character (1 to 9) identifies the function.
    - 2nd character (A to Z) identifies the failure mode.
    - 3rd character (1 to 9) identifies the cause.
- **Function:** A description of the function under analysis.
- **Failure Rate (λ):** The frequency (or rate) with which an item will be unable to perform its intended function in the operational environment for which it was designed. This is expressed as failures per flight hour.
- **Failure Mode:** A description of the functional or equipment failure. Failure modes are determined by examination of the functional outputs identified on the applicable Functional Block Diagram or based on the system description.
- **Failure Cause:** A description of the unique component or equipment or functional loss.
- **Mission Phase:** Describes the flight regime or maneuver that the aircraft is executing when the functional failure occurs. The default condition is "ALL," indicating that the failure mode is applicable to all phases of flight and ground maneuvering.
- **Local Failure Effect:** The consequence the failure has on the operation, function, or status of the specific item being analyzed.
- **Next Higher Effect:** The consequence the failure has on the operation, functions, failed equipment or other system components, including automatic enabling of backup and/or redundant systems.
- **End Effect:** This category defines the worst case end effect the defined failure has on the operation, function, or status of the aircraft. The effect shall be written in such a way that the severity determination is substantiated.
- **Detection Method:** The means or mechanism by which the defined failure can be discovered.
- **Compensating Provision (CP):** Design provisions or operator actions which circumvent or mitigate the effects of a failure. Compensating design provisions are features at any indenture level that will nullify the effects of a malfunction or failure, but do not prevent its occurrence.
- **Severity Code:** Severity is divided into four categories as defined in MIL-STD-1629A. Severity category is assigned to provide a qualitative measure of the worst potential consequences resulting from design error or item failure. The severity classifications are described in Table 19.
- **Failure Mode Ratio (α):** The fraction of item failures related to the failure mode under consideration. The sum of all failure modes related to a specific equipment or item will equal one (1). When a particular equipment or item has more than one possible failure effect, the mode ratio was is distributed evenly, except for the drive and power system, see Section 12.2.3.
- **Failure Effect Probability (β):** The conditional probability that the failure end effect will result in the identified severity classification given that the failure mode occurs. A 1.0 value indicates that the failure mode results in an actual loss in all instances. A value of less than 1.0 indicates a lower probability of actual loss.

- **Beta Mode Ratio Explanation:** The description of why/how a particular Failure Effect Probability was selected.
- **Failure Mode Criticality Number:** The probability that a failure for a component resulting from a particular failure mode will result in the severity classification identified. Criticality is the failure rate multiplied by failure mode ratio, failure effect probability, and operating time. For this analysis, the operating time was, by default, one hour.
- **Incipient Failure Symptoms:** Description of symptoms that can provide early warning before the failure event actually occurs, either through health monitoring, observation of abnormal condition, or scheduled maintenance inspections. Incipient failure symptoms help avoid worst case end effect, and are considered when determining Beta.

12.2 Analysis Approach

12.2.1 Electric Power and Distribution System and Thermal Management System Reliability Analysis

Failure rate predictions for each FMECA component were derived from an industry search for historical component failure rates from non-electric parts database (NPRD) (ref. 42), Quanterion's Failure Mode/Mechanical Distributions (FDM) (ref. 43), and prior work (ref. 5). Within those numbers, distributions were derived and associated with each of the postulated functional failures. A rate was determined for each functional failure mode specific to each sub-system or component. This rate was tabulated in the worksheet, along with the conditional probability that the failure effect results in the identified severity code and the failure effect probability (β), generating a mode criticality number for each failure condition. All failure effect probability values are based on engineering judgment.

12.2.2 Flight Control System Reliability Analysis

A similar approach to the electrical power and distribution system, see Section 12.3.1, was performed for the majority of the flight control system analysis. Failure rate predictions were taken from prior work (ref. 5) and carried through the FMECA process. However, the mechanical flight controls required additional scrutiny due to lower than expected failure rate budgets output from the initial PSSA fault tree.

The actuation system derived requirements from the PSSA included $\leq 10^{-10}$ catastrophic failures per flight hour, and VTOL.2250(c) requires that no single failure have a catastrophic effect upon the aircraft. The PSSA derived requirement is orders of magnitude lower than typical requirements for compliance with CS-27 or CS-29. AC 29-2C allows for a dual independent hydraulic system in lieu of direct compliance with an overarching 10^{-9} catastrophic failures per flight hour requirement and allows for single failures and jams if shown to be extremely improbable. However, SC-VTOL-01 does not provide an exception for dual independent hydraulic systems nor for single failures, and, therefore, must be in direct compliance with SC-VTOL-01, specifically, 10^{-9} catastrophic failures per flight hour and VTOL.2250(c) single failure criteria, or an exception is required and not guaranteed.

In order to fulfill contract scope, an actuator design was sought that would directly comply with the catastrophic failure criteria. Future work is required to further develop the design for direct compliance with VTOL.2250(c), but fielded designs with redundant load paths exist today. In order to validate that an actuator and mechanical flight control sub-system would be able to meet the

10^{-10} PSSA criteria, a legacy "Flight Safety Report" (FSR) method of reliability and safety analysis was used. The FSR method is similar to a fault tree analysis in that it uses "and" and "or" gates to calculate estimated failure rates of desired components or systems. Failure rates from prior work (ref. 5) were used to populate the FSR and sub-system schematics, Section 8-11, and engineering judgment were used to populate the interconnectivity within the redundant actuator system.

The FSR method for the triple hydraulic actuator system is shown in Figure 88. Visually, this is similar to a fault tree turned on its side. The FSR diagram is read from left to right with functional blocks in horizontal rows connected through "or" gates and the triangle representing an "and" gate. The red, shaded blocks represent mechanical jams and opens that are typically allowed per AC29-2C; however, additional work is required to create redundant load paths to reduce the probability of mechanical opens to an acceptably low level and eliminate single failures with catastrophic outcomes. Billions of flight hours and endurance testing have shown mechanical jams are not generally accepted as reasonable failure modes for hydraulic actuators. However, more work with regulating bodies is required to determine if mechanical jams in hydraulic actuators are reasonable and conceivable failure modes for this application and mission.

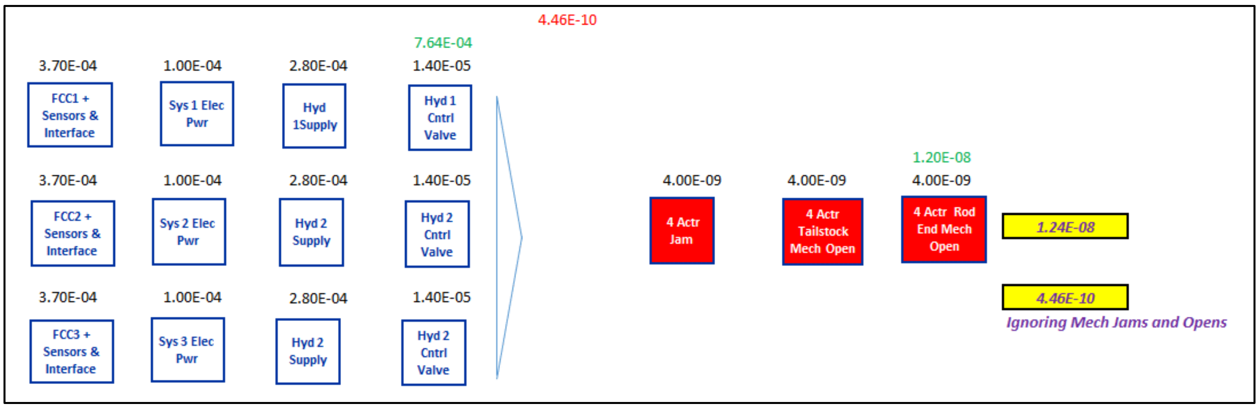


Figure 88: FSR Diagram for Triple Hydraulic Actuator

12.2.3 Drive and Power System Reliability Analysis

The drive and power system needed to demonstrate substantial decreases in failure rate in order to meet PSSA requirements and show compliance to SC-VTOL-01. Prior work (ref. 5) established generalized catastrophic failure rates of 5×10^{-6} and 3×10^{-4} for gearboxes and motors, respectively. However, outputs from PSSA budgeting required that gearbox and motor failure rates were $\leq 5 \times 10^{-11}$ and $\leq 10^{-6}$, respectively. To substantiate decreased failure rates, a similar method to that described by Smolders, et al (ref. 44) and in NSWC-11 (ref. 45) was employed to predict failure rates. Sub-system FMECAs were developed to tabulate and categorize failure rates according to their severity. Outcomes of the reliability analysis included additional derived design requirements and derived sensor requirements.

Functional block diagrams for drive and power sub-systems are shown in Figure 89 and Figure 90. The functional block diagrams decompose each sub-system into functional sub-systems, functions, and components (ref. 44). Failure modes for decomposed components were selected based on engineering judgment, using NSWC-11 as a guide, Table 20. Failure rates were calculated using equations from NSWC-11, see Table 21, and applied to each failure mode; assumptions used to calculate failure rates are captured as derived requirements that were fed back into the drive and power system design. Sub-system FMECAs were developed to tabulate the criticality of each failure mode against the associated severity. Derived sensing requirements were fed back into the drive and power system design. Finally, the criticality of each failure mode was rolled up into system failure rates and used to update the PSSA fault tree.

Most of the drive and power system failure modes distributed failure mode ratio, α, evenly across failure modes with more than one failure cause. Except gear tooth failures in which gear tooth bending failures were estimated to be 10% and gear tooth pitting failures were assumed to be 90% of the total gear tooth failures. Search data from Greaves, et al (Ref. 32) was used to substantiate this value, which catalogued twelve main gearbox failures. Four of which mention gears in the summary. Accident reports for the four noted gear-related incidents were evaluated. Only one of twelve, or 8% of, noted incidents were found to potentially include gear tooth bending failures of the main power gears, although none of the reports specifically identify gear tooth bending fatigue as the root cause. This also manifests itself as a derived requirement that gears must be pitting critical so that gear tooth failures are more likely to result in pitting failures as opposed to gear tooth bending failures.

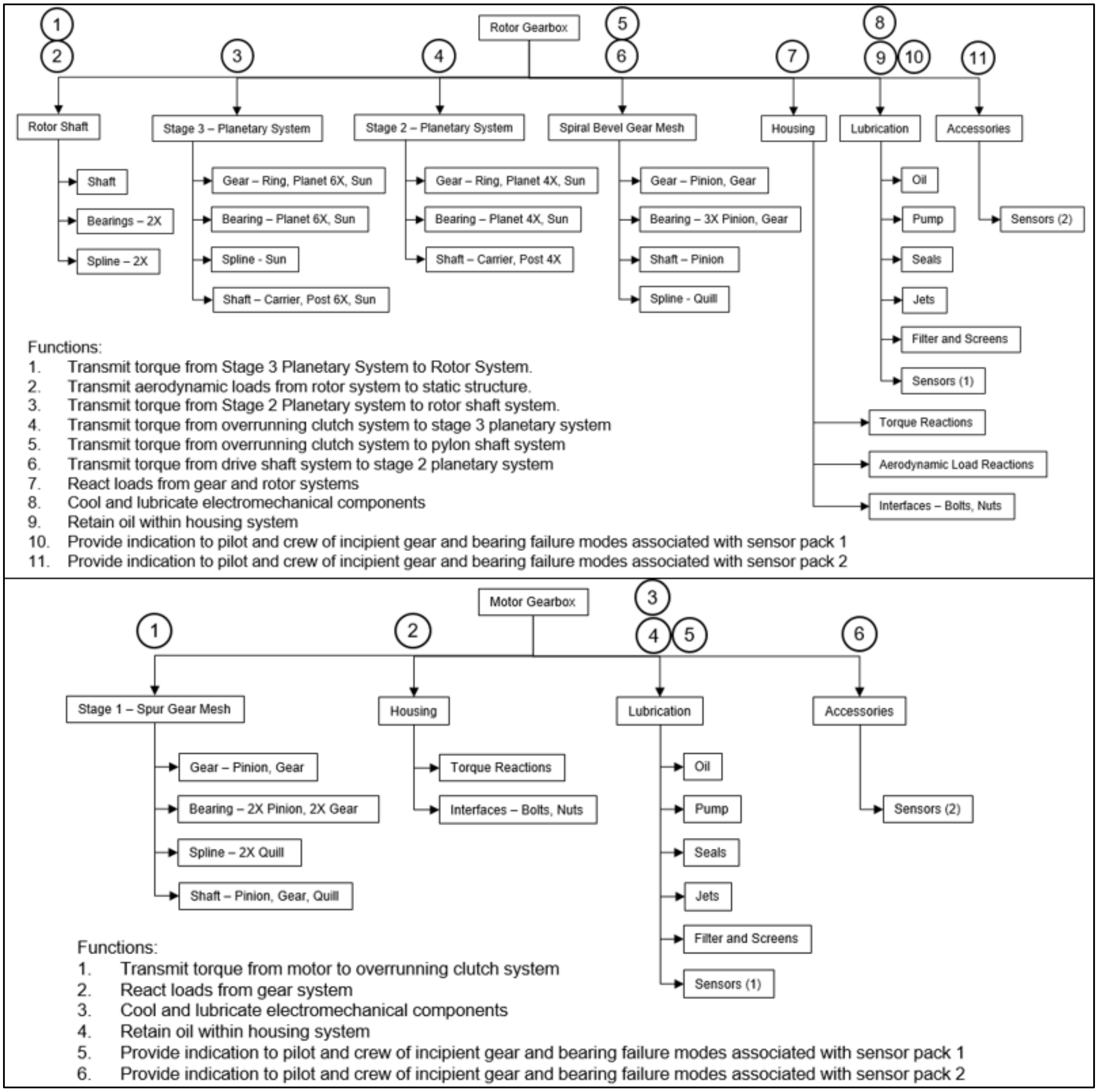


Figure 89: (Top) Rotor Gearbox Functional Block Diagram and (Bottom) Motor Gearbox Functional Block Diagram

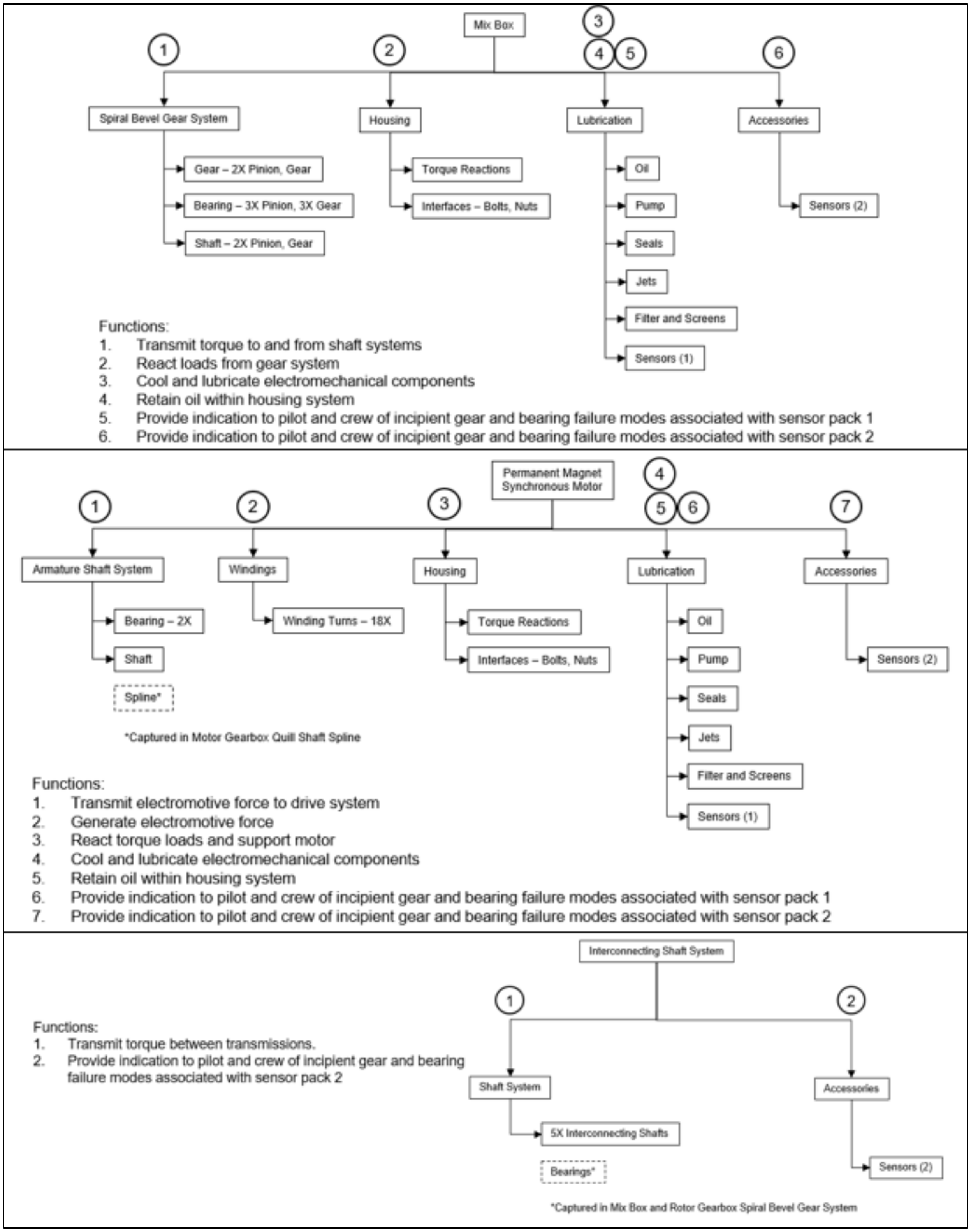


Figure 90: (Top) Mix Box FBD, (Middle) PMSM FBD, and (Bottom) Interconnecting Shaft System FBD

Table 20: Failure Modes for Drive and Power System Components

Component	Gears	Bearings	Splines	Shafts	Armatures
Reasonable and Conceivable Failure Modes	Pitting	Spalling	Wear	Fatigue	Shorted Winding
	Tooth Bending Fatigue	Brinelling		Fretting	Overheating
		Cracked Rings			
		Arc Burns			

Table 21: Calculated Failure Rates per NSWC-11 – Drive and Power System

Component	Failure Rate
Gears	3.22×10^{-8}
Shafts – Transmission	2.86×10^{-17}
Shafts – Interconnecting Drive	9.31×10^{-15}
Bearings	9.58×10^{-4}
Splines	9.10×10^{-9}
Windings	3.45×10^{-5}
PMSM Base Failure	3.50×10^{-6}

Derived design requirements:

- ADS-50-PRF gearbox qualification requirements, including 130% overload and 120% overspeed testing.
- Gear teeth shall be designed to be pitting critical, as opposed to bending critical.
- Material strength assurance of ≥99% probability with ≥95% confidence.
- Design practices must account for internal and external deflections and thermal expansion during operation.
- Bearing B_{10} life of ≥4,500 hours.
- Lubrication/cooling system must use MIL-PRF-7808 oil or better.
- Water content in oil must be kept to ≤25 wt. %.
- Lubrication/cooling media must include 10 micron filter or finer.
- Winding design life of ≥25,000 hours.
- Winding insulation temperature limit must be ≥10˚C greater than WCA temperature warning for oil cooled systems and ≥15˚C greater for air cooled systems.
- Phase winding voltage unbalance must be ≤1%.

Derived sensing requirements:

- All sensors are functioning and able to detect ≥99.9% of all applicable signals.
- Chip detectors, electrified debris screens, and vibration health monitoring is used inside the gearbox for detecting gear, bearing, and shaft failures.
- Torque ripple and vibration health monitoring is used inside the motor for detecting bearing and shaft failures.
- Oil condition monitors are used to detect oil quality.
- Temperature and pressure monitors are used for real time monitoring and fault detection.
- The PMSM requires a minimum of the following real-time cockpit displays and warning/caution/advisory (WCA) indications:
  - Torque cockpit display
  - Shaft speed cockpit display
  - Torque WCA
  - Shaft speed WCA
  - Oil temperature cockpit display
  - Oil pressure cockpit display
  - Oil temperature WCA
  - Oil pressure WCA
  - Torque ripple WCA
  - Bearing failure WCA
  - Insulation quality WCA
  - Short detection, shutoff, and WCA

12.3 Summary and Conclusions

Appendix C and Appendix D contain the FMECA worksheets for the drive and power system and NASA RVLT concept vehicles analyzed, respectively. Table 22 shows a failure rate summary. Failure rates developed for the eQuad were passed into the eQuad without cross-shafting, and failure rates developed for the RPM-controlled hexarotor were passed into the RPM-controlled octorotor due to the common features and large component reuse between each configuration.

Detailed system and sub-system analysis allowed simple replication of common failures across configurations. Replication allowed design changes to be efficiently worked into all configurations. Detailed design and reliability analysis provided realistically achievable failure rates with state-of-the-art technology. Failure rates of the Subsystem FMECAs were compiled at the aircraft level to feed into the PSSA for initial verification of compliance to SC-VTOL-01.

More detailed evaluations and sensitivity studies of mean time between maintenance, removal, and overhaul are recommended to develop UAM operating costs and safety guidelines. Designs developed under the current work are intended to meet SC-VTOL-01 safety requirements, but may incur undue unit or operating costs burdening the end user. Characterizing the sensitivity to design decisions in each major sub-system may uncover substantial operating and support cost savings.

Table 22: Failure Rate Summary

Category	**Table C Quadrotor**			**Table B Hexarotor**	
	All Electric	Hybrid	Turboshaft	Collective Control	RPM Control
I	1.84×10^{-9}	1.84×10^{-9}	1.80×10^{-9}	2.03×10^{-7}	2.29×10^{-8}
II	5.17×10^{-4}	6.90×10^{-4}	7.66×10^{-6}	4.43×10^{-4}	4.43×10^{-4}
III	1.43×10^{-5}	1.11×10^{-5}	0.00×10^{0}	2.13×10^{-5}	2.13×10^{-5}
IV	1.47×10^{-6}	1.47×10^{-6}	3.81×10^{-6}	2.01×10^{-6}	2.01×10^{-6}

13 PRELIMINARY SYSTEM SAFETY ANALYSIS

13.1 Fault Tree Analysis Methodology

Following the FHA, the resulting functional failures are consolidated into a more concise list of hazards. Each hazard is assigned a severity, in accordance with the severity of the functional failure(s) encompassed by the hazard, and is also assigned a hazard probability. Different techniques have been employed at the discretion of safety analysts, to determine the hazard probabilities for each hazard. For this study, a FTA was performed in order to model the connectivity between components, systems, and functions.

FTA is a top-down analysis used in this study to capture the propulsion and control systems as well as to examine their interrelationships and allow the definition of cut-sets to show areas where system improvements would improve the top-level number. The roll-up of the propulsion FTAs are done so that the aircraft level functional failure “loss of propulsion” also includes an element of loss of control. Loss of multiple propulsors may cause control problems that are considered a part of the loss of propulsion top level hazard. Two primary fault trees were used to model the pitch controlled (fixed speed) aircraft configurations to show compliance with VTOL.2510(a). One fault tree models the aircraft level “loss of propulsion” while the other models aircraft level “loss of pitch control”. The variable speed (fixed pitch) configurations use one fault tree to model loss of control as rolled-up in the loss of propulsion top level hazard. The FTA is meant to document a catastrophic or hazardous top level outcome, though lesser severity hazards may become evident due to FTA structure and execution.

Overall, the propulsion specific system failures rollup to the top-level hazard “loss of propulsion”; the collective control specific system failures rollup to the secondary top-level hazard “loss of pitch control” for pitch-controlled configurations. The unique and common mode DPFC aspects of the aircraft were captured in the FTA.

The top level hazards defined from the FHA were used to inform the top level of the fault tree. The FTA was done to a level of detail sufficient to show architectural impacts to the top-level hazard. The FTA may capture system effects that rollup to higher losses of functions captured in the FHA.

The FTAs performed for this study used a series of “AND” and “OR” gates to build the fault tree architecture in Reliability Workbench (RWB) (ref. 46). The symbols used in the associated fault tree diagrams are shown in Figure 91. RWB calculates the output of the “AND” and “OR” gates as shown in Equations (1) and (2), respectively. Equations (1) and (2) use the three (3) input examples shown in Figure 91, but more inputs may be applied by adding terms.

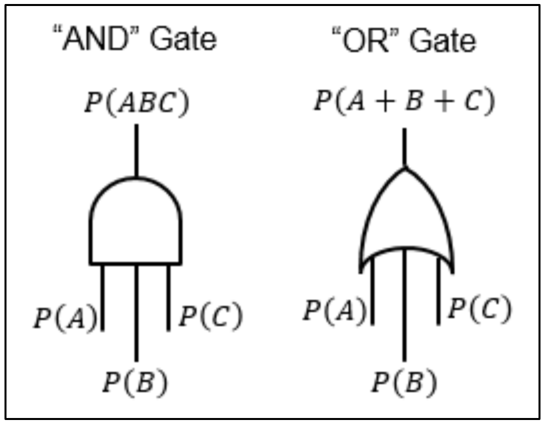


Figure 91: "AND" and "OR" Gate Symbols

$$P(ABC) = P(A) * P(B) * P(C) \tag{1}$$

$$P(A+B+C) = P(A) + P(B) + P(C) - P(A) * P(B) - P(A) * P(C) - \\ -P(B) * P(C) + P(A) * P(B) * P(C) \tag{2}$$

13.2 Fault Tree Analysis Results and Discussion

Using the Fault Tree architecture, system allocations for failure rate budgets are developed with the objective to meet a top-level aircraft failure ("loss of propulsion" or "loss of flight control path") to be within the range of the SC-VTOL-01, VTOL.2510(a) requirement of less that $1x10^{-9}$ catastrophic failure per fight hour (pfh). The safety derived requirements from the FHA, Section 7, combined with the probability budgets allocated for the systems in the initial FT architectures are then utilized by subsystem designers to incorporate into the design. The subsystems are then analyzed with the FMECA, Section 12. Final subsystem designs, with underpinning safety requirements and redundancy management, are incorporated into the FT architecture to accurately model the final design and systems interconnectivity, thus accurately modelling the roll-up of functional failures to the top-level hazard. FMECA failure rate data was incorporated into the FTAs to complete the final analysis. In the final FTAs, some undeveloped areas of consideration were probed further for "sensitivity" or highlight where improvements can be made (e.g., technological advancements, future system design and development, and future safety and reliability analysis). The summary of the FT top-level and next tier hazards are summarized in the following subsections.

13.2.1 Quadrotor PSSA

13.2.1.1 All-Electric Quadrotor

The eQuad (with cross-shafting) fault tree diagram may be found in Appendix E with a summary provided in Table 23. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The all-electric quadrotor (with cross shafting) FTA predicts $7.00x10^{-10}$ propulsion (power transmission) failures per flight hour and $1.78x10^{-9}$

collective control (Loss of Flight Path Control) failures per flight hour. This FTA prediction assumed individual battery pack failure rates of 10^{-5}, or 10^{-10} for HDVC system redundancy per motor.

Table 23: eQuad FTA Summary

All Electric Quadrotor with Cross Shafting		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***7.001E-10***
2	Any Dual Propulsor Failure	3.157E-10
4	Single Propulsor Failure	4.391E-06
5	Electric Motor Function Failure	4.346E-06
5	ESC Function Failure	5.491E-07
5	Power Loss to Motor	1.000E-10
2	Loss of Ability to Drive Any Rotor	3.844E-10
3	Single Propulsor Failure w/Cross Shafting Failure	3.393E-16
3	Loss of a Single Gearbox	3.844E-10
1 (Top)	***Loss of Flight Path Control***	***1.784E-09***
2	Loss of Collective Control of Any Rotor	1.784E-09
3	Loss of Pitch Control to a Rotor	4.460E-10
2	Loss of Hydraulic Power	4.098E-16
3	Loss of Hydraulic System	4.241 E-08

13.2.1.2 Hybrid-Electric Quadrotor

The hQuad fault tree diagram may be found in Appendix E with a summary provided in Table 24. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The hybrid-electric quadrotor FTA predicts 7.00×10^{-10} propulsion (Loss of Power Transmission) failures per flight hour and 1.78×10^{-9} collective control (Loss of Flight Path Control) failures per flight hour, which are the same the all-electric quadrotor top-level failures. This FTA prediction assumed individual battery pack failure rates of 10^{-5}, or 10^{-10} for HDVC system redundancy per motor, consistent with the eQuad.

The main in functional failure rates in the hQuad relative to the eQuad is identified in the "Power Loss to Motor" gate, which is significantly lower risk in the hQuad due to the fully redundant HVDC systems (turbogenerator and batteries).

Table 24: hQuad FTA Summary

Hybrid Electric Quadrotor		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***7.001E-10***
2	Any Dual Propulsor Failure	3.157E-10
4	Single Propulsor Failure	4.391E-06
5	Electric Motor Function Failure	4.346E-06

Hybrid Electric Quadrotor		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
5	ESC Function Failure	5.491E-07
5	Power Loss to Motor	1.800E-14
2	Loss of Ability to Drive Any Rotor	3.844E-10
3	Single Propulsor Failure w/Cross Shafting Failure	3.393E-16
3	Loss of a Single Gearbox	3.844E-10
1 (Top)	***Loss of Flight Path Control***	***1.784E-09***
2	Loss of Collective Control of Any Rotor	1.784E-09
3	Loss of Pitch Control to a Rotor	4.46E-10
2	Loss of Hydraulic Power	4.098E-19
3	Loss of Hydraulic System	4.241E-08

13.2.1.3 Turboshaft Quadrotor

The tQuad fault tree diagram may be found in Appendix E with a summary provided in Table 25. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The turboshaft quadrotor FTA predicts 5.10×10^{-10} propulsion (Loss of Power Transmission) failures per flight hour which is similar to that of the eQuad and hQuad. The tQuad fault tree architecture is significantly different from the eQuad and hQuad, since there are no electric propulsor branches and the “Loss of Power Transmission” is modeled to include a Dual Engine Fail gate. For the eQuad and hQuad, the motor torque is transferred to each rotor gearbox which drives the combiner gearboxes; however, in the tQuad, the two engines directly drive the combiner transmission and rotor gearboxes are driven off of the two combiner gearboxes. With this architecture, the combiner gearbox failures have a greater effect and outcome than in the eQuad and hQuad which can continue to drive rotor gearboxes by the motors. For the other electrified quads, only a combination in the loss of a single propulsor *and* a combiner transmission will contribute to the “Loss of Ability to Drive Any Rotor”. The tQuad FTA also predicts 1.78×10^{-9} collective control (Loss of Flight Path Control) failures per flight hour, consistent with the all-electric quadrotor top-level failures.

Table 25: tQuad FTA Summary

Turboshaft Quadrotor		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***5.100E-10***
2	Loss of Ability to Drive Any Rotor	4.063E-10
3	Loss of a Single Gearbox	3.844E-10
3	Loss of Power to Rotors	1.037E-10
2	Dual Engine Fail	1.915E-06
3	Loss of an Engine Output	5.100E-10
1 (Top)	***Loss of Flight Path Control***	***1.784E-09***
2	Loss of Collective Control of Any Rotor	1.784E-09
3	Loss of Pitch Control to a Rotor	4.46E-10

Turboshaft Quadrotor		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
2	Loss of Hydraulic Power	4.653E-16
3	Loss of Hydraulic System	5.287E-08

13.2.1.4 All-Electric Quadrotor without Cross Shafting

The eQuad (without cross-shafting) fault tree diagram may be found in Appendix E with a summary provided in Table 26. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The all-electric quadrotor (without cross shafting) FTA predicts $1.76x10^{-5}$ propulsion (power transmission) failures per flight hour and $1.78x10^{-9}$ collective control (Loss of Flight Path Control) failures per flight hour. This FTA prediction assumed individual battery pack failure rates of 10^{-5}, or 10^{-10} for HDVC system redundancy per motor. The failures used for this configuration were taken from the cross-shafted eQuad as the FMECAs were assumed to be similar (rather than generation of non-cross-shafted quad failure rates through an independent FMECA). The risk for loss of propulsion, top-level hazard is approximately 4 orders of magnitude greater that the cross-shafted variant.

The collective control top-level hazard is the same risk across the quadrotors due to the actuator failure rate number being the main variable driving the risk by several orders of magnitude with hazards. The actuators, while powered by hydraulic systems with varying drive inputs, are the same component with same failure rate across all configurations. The triplex hydraulic system for the cross-shafted configurations were designed with two hydraulic pumps on one combiner gearbox and the third pump on the other combiner gearbox. Without the combiner gearbox, the un-cross-shafted eQuad hydraulic pumps are all driven by independent rotor gearboxes, which removes the common failure mode. In the tables above, the cross shafted quadrotors show the complete loss of hydraulic power at significantly higher risk; however, this probability for any quadrotor relative to the actuator component failure rate, do not impact the top-level risk (Loss of Flight Path Control).

Table 26: eQuad (without cross shafting) FTA Summary

eQuad *without* Cross Shafting		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***1.756E-05***
2	Any Propulsor Failure	1.756E-05
3	Single Propulsor Failure	4.391E-06
5	Electric Motor Function Failure	4.346E-06
5	ESC Function Failure	5.491E-07
5	Power Loss to Motor	1.000E-10
2	Loss of Ability to Drive Any Rotor	3.844E-10
3	Single Propulsor Failure w/Cross Shafting Fail	n/a
3	Loss of a Single Gearbox	3.844E-10
1 (Top)	***Loss of Flight Path Control***	***1.784E-09***
2	Loss of Collective Control of Any Rotor	1.784E-09

eQuad *without* Cross Shafting		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
3	Loss of Pitch Control to a Rotor	4.46E-10
2	Loss of Hydraulic Power	1.076E-22
3	Loss of Hydraulic System	4.250E-08

Additional work is needed to evaluate the feasibility of a non-cross shafted quadrotor due to the high risk of catastrophic propulsion failure. An alternate Fault Tree model was completed to determine if this vehicle configuration could achieve the EASA 10^{-9} requirements at the top gate. A representative Fault Tree was developed to model a complete layer of redundancy at each propulsor, see Appendix E. As such, the adjusted FTA predicts 1.06×10^{-9} propulsion failures per flight hour by modeling each propulsor to contain the following level of redundancy:

- 2 Motors
- 4 ESCs
- 2 Cooling Systems
- 2 Overrunning Clutches

This exercise was solely modeled in the FT using the original FT architecture with the dual redundancy to simulate potential redundancy (Table 27) in attempt to meet the EASA top-level failure probably requirement. While this demonstrates that the adjusted prediction is within the range of the 10^{-9} requirement and only a slight increase in risk from the eQuad and hQuad, additional work is needed to determine feasibility of the system design, complexity, and additional volume and weight with the type of redundancy described above to be integrated into this aircraft design. It should be noted that if the redundant propulsors shared a common cooling system per rotor, “Loss of Propulsion” would improve to only 2.19×10^{-6} failures per flight hour.

Table 27: Redundancy-Adjusted eQuad (without cross shafting) FTA Summary

eQuad *without* Cross Shafting – Dual Redundant Propulsors per Rotor		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***1.062E-09***
2	Any Propulsor Failure	6.771E-10
3	Single Propulsor Failure	3.193E-10

13.2.2 Hexarotor PSSA

13.2.2.1 Pitch-Controlled Hexarotor

The Pitch eHex Fault Tree Diagram may be found in Appendix E with a summary provided in Table 28. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The pitch-controlled (fixed speed) FTA predicts 5.08×10^{-10} propulsion (power transmission) failures per flight hour and 1.50×10^{-11} collective control (Loss of

Flight Path Control) failures per flight hour. This FTA prediction assumed that overspeed conditions can be managed, but the placeholder for overspeed controls within the Fault Tree must be developed. This FTA prediction also assumed individual battery pack failure rates of 10^{-10} for HDVC system redundancy per motor. Refer to Section 13.3.2 on HVDC system sensitivity.

Table 28: Pitch-Controlled eHex FTA Summary

Pitch eHex		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***5.078E-10***
2	Loss of ability to provide thrust	4.769E-10
3	Any Dual Propulsor Failure	4.769E-10
5	Single Propulsor Failure	4.297E-06
6	Electric Motor Function Failure	4.296E-07
6	ESC Function Failure	5.469E-07
6	Power Loss to Motor	1.000E-10
2	Loss of Ability to Drive Any Rotor	3.090E-11
3	Dual GB Fail	1.104E-19
3	Single GB Fail Jam	3.090E-11
1 (Top)	***Loss of Flight Path Control***	***1.501E-11***
2	Loss of Collective Control of Two Rotors	1.501E-11
3	Loss of Pitch Control to a Rotor	1.000E-06
2	Loss of Hydraulic Power	7.669E-23
3	Loss of Hydraulic System	4.249E-08

13.2.2.2 RPM-Controlled Hexarotor

The Pitch eHex Fault Tree Diagram may be found in Appendix E with a summary provided in Table 29. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The speed-controlled (fixed pitch) FTA predicts 2.64×10^{-8} propulsion (power transmission) failures per flight hour. There is no secondary Fault Tree associated with collective control (Loss of Flight Path Control). This FTA prediction assumed individual battery pack failure rates of 10^{-10} for HDVC system redundancy per motor. The risk of catastrophic failure is three orders of magnitude greater than the Pitch eHex configuration due to the lack of overspeed control. However, a potential reduction in propulsion failures may be achieved with improvements to the main components that may contribute to overspeed conditions.

Table 29: RPM-controlled eHex FTA Summary

RPM eHex		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***2.642E-08***
2	Loss of ability to provide thrust	2.639E-08
3	Any Dual Propulsor Failure	2.969E-10
5	Single Propulsor Failure	5.497E-07

RPM eHex		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
6	Electric Motor Function Failure	5.460E-07
6	ESC Function Failure	5.460E-07
6	Power Loss to Motor	1.000E-10
3	Overspeed Condition	2.609E-08
2	Loss of Ability to Drive Any Rotor	3.090E-11
3	Dual GB Fail	7.455E-22
3	Single GB Fail Jam	3.090E-11

The specific functional failures that can result in an overspeed condition as modeled in the Fault Tree are shown in Figure 92.

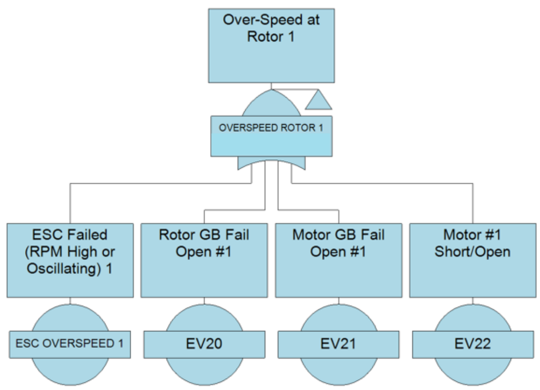


Figure 92: Overspeed Failure Conditions

The failure rates associated with each of these failures were based on the assumption that 1% of each components' failures would result in a potentially catastrophic overspeed condition. For example, out of all motor failures (failing open or short), it is assumed that 1% of the failures have a catastrophic overspeed effect. The failure rates associated with 1% of failures include:

- ESC Fail 5.40×10^{-10}
- Rotor GB Fail 5.15×10^{-12}
- Motor GB Fail 3.22×10^{-12}
- Motor open/short 3.8×10^{-9}

The motor (open/short) failure condition was several orders of magnitude more likely to occur than the other overspeed condition failures. To model the impact of motor improvements (to limit overspeed), the failure rate was bolstered to 1×10^{-11}. With this adjustment, re-allocated failure rate in the FTA shows a prediction of 3.61×10^{-9} propulsion (power transmission) failures per flight hour, shown in Table 30.

Table 30: Impacts to RPM-Hex Failures with Bolstered Overspeed Safety

RPM-Hex w/ 10^{-11} Probability of Motor Open/Short Failures (pfh)		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**

1 (Top)	***Loss of Power Transmission***	***3.607E-09***
3	Overspeed Condition	3.49E-09

13.2.3 Octorotor PSSA

The summary of the RPM-octorotor fault tree diagram is provided in Table 29. This table includes the top level hazards, shown as Gate Level 1 and several sub-tier hazards that roll-up to that top gate. The speed-controlled (fixed pitch) FTA predicts 3.49×10^{-8} propulsion (power transmission) failures per flight hour. This FTA prediction assumed individual battery pack failure rates of 10^{-10} for HDVC system redundancy per motor. There is no secondary Fault Tree associated with collective control (Loss of Flight Path Control). The failures used for this configuration were taken from the RPM-hex configuration as the FMECAs were assumed to be similar (rather than generation of octorotor failure rates through an independent FMECA). This is a slight increase in risk of catastrophic failure from the RPM-hex configuration due to the additional rotors to increase the likelihood of failure.

Table 31. RPM-Oct FTA Summary

RPM-Oct		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***3.486E-08***
2	Loss of Ability to Provide Thrust	3.482E-08
3	Any Dual Propulsor Failure	2.848E-11
5	Single Propulsor Failure	5.506E-07
6	Electric Motor Function Failure	5.980E-07
6	ESC Function Failure	5.467E-07
6	Power Loss to Motor	1.000E-10
3	Overspeed Condition	3.48E-08
2	Loss of Ability to Drive Any Rotor	4.120E-11
3	Dual GB Fail	1.392E-21
3	Single GB Fail Jam	4.12E-11

Similar to the RPM-hex, a potential improvement to 4.54×10^{-9} propulsion failures per flight hour is achievable if a motor open/short failure is assumed to occur at a rate of 10^{-11} pfh (see Table 32).

Table 32: Impacts to RPM-Oct Failures with Bolstered Overspeed Safety

RPM-Oct w/ 10^{-11} Probability of Motor Open/Short Failures (pfh)		
Gate Level	**Failure Gate Description**	**Failure Rate (pfh)**
1 (Top)	***Loss of Power Transmission***	***4.537E-09***
3	Overspeed Condition	4.47E-09

13.3 Areas for Future Work

13.3.1 Electrical Power and Distribution Modeling

As discussed in Section 11, the electrical system requires additional work to understand component details, interconnectivity with other systems, and substantiation of the failure rates for the HVDC/LVDC systems. The HVDC and LVDC systems were modeled in the FTAs with underdeveloped failure rates based on engineering judgment. The failure rate for the LVDC failures was considered a common system across the aircraft. The HVDC failures were treated with generic failure rates that were unique to each propulsor, and do not act as a common cause failure across other propulsors. For electrified configurations the "HVDC system" failure rate was identical (10^{-10} failures pfh) regardless of number of packs, e.g., for quad rotor configurations with two packs per motor, each pack was modeled as 10^{-5} failures pfh such that the complete loss of HVDC power at each motor was 10^{-10} pfh. For hexarotor and octorotor configurations with a single "pack" per motor, the pack safety objective was also 10^{-10} failures pfh. While there are no detailed systems to substantiate these failure rates at this time, using the same HVDC and LVDC system failure rates across configurations providing more direct focus of the safety and reliability trades from the other main systems which were the focus for development in this work.

13.3.2 Single Point Failures

During the evaluation of the hexarotor configurations, the depth of examination required to ascertain minor versus catastrophic failure conditions identified several single point failures that must be addressed in future work. Refer to the FHAs for the effect of these failures on various configurations. Additional work is needed, not only to analyze the failures that contribute to a catastrophic top-level failure, but also to incorporate across all designs in attempt to eliminate single failures to comply with VTOL.2250(c). Single point failures identified in this study are:

- Rotor and rotor attachment
- Mechanical jam or open in mechanical flight controls
- Mechanical jam or open in gearboxes
- Potential fire due to single motor jam

13.4 Summary

A summary of the FTA predictions of aircraft-level catastrophic failure per flight hour are shown in Figure 93 for propulsion (power transmission) failures and in Figure 94 for collective control (Loss of Flight Path Control) failures per flight hour. All of the collective controlled (fixed speed) vehicle configurations were able to meet the EASA SC-VTOL-01 10^{-9} catastrophic failure criteria. The eQuad without cross-shafting requires a feasible DPFC architecture discussed in Section 9.1.4 with dual redundant motor and ESC cooling, which can be architected using off-the-shelf components.

Adjustments to either overspeed failure conditions for the RPM-controlled eHex and eOct resulted in all vehicle configurations approaching the VTOL.2510(a) 10^{-9} catastrophic failure criteria. Overspeed prevention/mitigation and HVDC system must be future developed. Single point failure modes were identified that may increase risk as a function of the number of rotors in the design.

Across all configurations, redundancy management and architectural improvements will decrease probability of catastrophic failures. Additional detailed analysis, modeling, or simulation are required to verify aircraft configuration assumptions and designs should be iterated based on further evaluations.

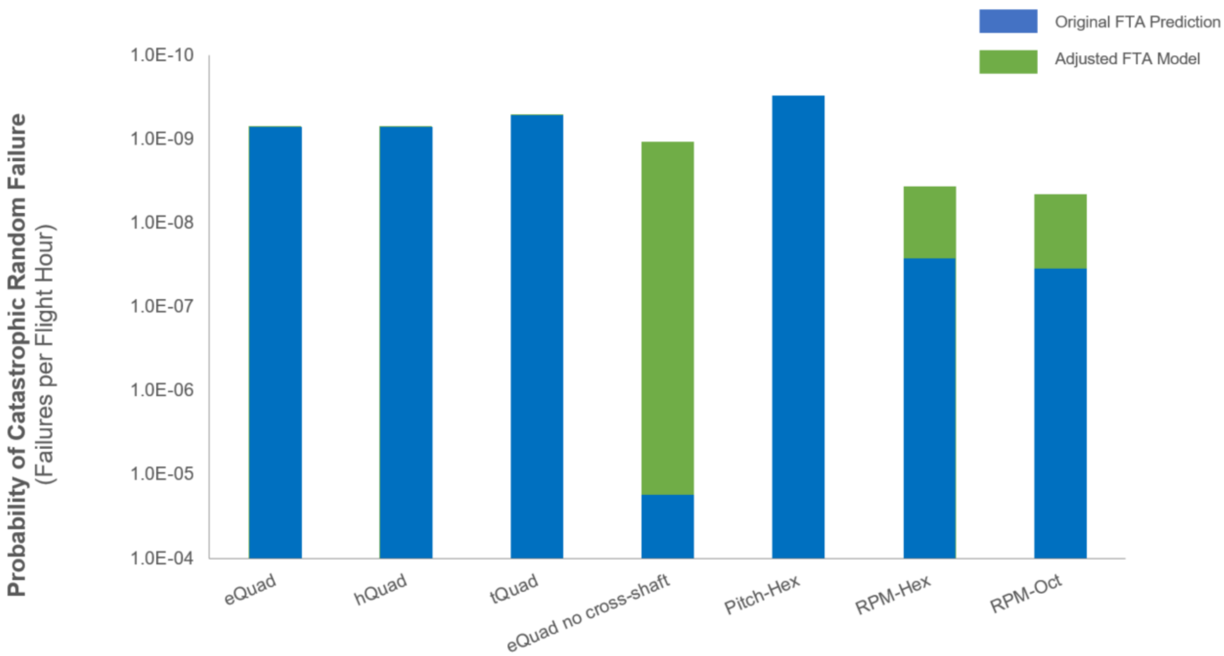


Figure 93: PSSA Summary – Probability of Catastrophic Propulsion Failure

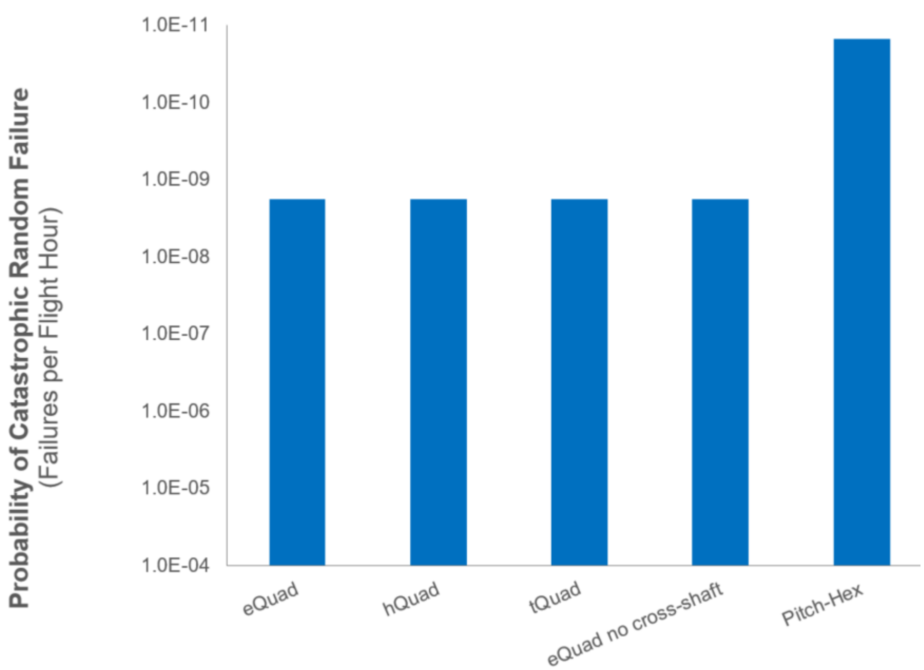


Figure 94: PSSA Summary – Probability of Catastrophic Collective Control Failure

14 DISCUSSION

The study results presented herein provided some particularly enlightening results that were not originally anticipated. NDARC component sizing and usage spectrums showed variability between architectures, but S&C simulations showed that NDARC component sizes may be unrealistic when less conventional architectures are employed. S&C simulations used simple LQR control schemes to compare aircraft attributes and highlighted engineering challenges associated with aircraft and architectural design decisions, such as using interconnecting shafting or distributed pitch-control, that should be fed back into the NDARC sizing routine. Additionally, system conceptual design teams found that complying with VTOL.2510(a) will increase system complexity when compared to legacy aircraft type certificated under CS-27 or CS-29 which include written exceptions to probabilistic failure criteria. Finally, only some of the aircraft were able to show paths to compliance with VTOL.2510(a) and redesign and additional coordination with EASA is required to comply with VTOL.2250(c). All of these factors sum together to show that modifications to the aircraft sizing routines are required to close on the safety case, and the aircraft cannot close on the design mission until the safety case closes.

14.1 Component Sizing and Usage Spectrum

As described in Section 4 the usage spectrum developed was based on one design mission that did not include maneuver power requirements. Additionally, the design mission was developed in absence of atmospheric disturbances, which is common for design missions, but the corresponding design criteria to size components for allowable transient response is not publically available and, therefore, requires simulation and flight testing to determine. In example, Section 9.1 uses a combination of NDARC sizing, ADS-50-PRF, and S&C simulations to develop power requirements to size the drive and power system. In the case of the pitch-controlled quadrotor, power requirements did not vary greatly from what was included in the original NDARC sizing routine. However, if that same sizing methodology was employed for the RPM-controlled octorotor, PMSM CRP would need to be increased by approximately nine times to account for maneuvering in an urban canyon environment, see Figure 24. That would require similar increases in the remainder of the drive and power system, not to mention the rotor system, blades, and supporting structure. Note, that doesn't include OMI performance, either.

An S&C simulation study was performed in order to reduce transient power requirements and found that active rotor braking can effectively reduce power transients, Figure 25. Active rotor braking may manifest itself in many forms, including regenerative braking to recharge the battery, heat sinking excess power to atmosphere through resistors or by mechanical brakes. S&C models for this study included regenerative braking capability, assuming effectively unlimited regeneration capability to approximate a lower limit on power transients. Active rotor braking has its own limitations on component design that were not studied. Limitations with the electrical or mechanical hardware performing the braking function were not evaluated and will have limitations, such as battery charging limits. Also, the rotor system and blades will see extremely low component lives if the abrupt changes in torque are not accurately captured in the rotor system usage spectrum.

14.2 S&C Simulation and Controller Type

Other methods may exist to reduce power transients for all study aircraft. The simple LQR controller was effectively utilized to compare aircraft attributes, but aircraft controller design and implementation has a large impact aircraft performance capability, including power transients for atmospheric disturbance rejection. Non-linear airframe models combined with more typical control theory used in aviation, plus elements such as power clipping or active rotor braking may show that power transients can be better managed than results indicate here.

14.3 Compliance with SC-VTOL-01

In addition to high power transients, compliance with SC-VTOL-01 has its own, novel engineering challenges. They manifest themselves into weight penalties that may not have been captured in initial aircraft sizing. VTOL.2250(c) and VTOL.2510(a) have the potential to drive significant weight and complexity into system designs.

Compliance with VTOL.2250(c) single failure criteria presents design challenges that must be overcome by novel designs, real-time monitoring, or both. Section 9.2.3 discusses impacts and potential design solutions as they related to the drive and power system, but a holistic review of the DPFC architectures presented needs to be performed, including, but not limited to, flight control actuators, mounting lugs, etc. Whichever system is under consideration, close collaboration with regulatory agencies, specifically EASA, is needed to ensure costly design revisions are not required.

Similarly, compliance with VTOL.2510(a), which outlines probabilistic failure criteria, has its own set of challenges. The flight control system for the quadrotors required more complexity in the form of a triplex hydraulic architecture to comply with VTOL.2510(a), Section 8.1.2. This same system was passed into the pitch-controlled hexarotor, but future evaluations may find that the hexarotor may comply with VTOL.2510(a) using dual redundant hydraulics or EMAs. Dual redundant hydraulics would result in a large reduction in complexity that would result in beneficial operating cost reductions. Unit cost and repair/replacement cost would be reduced and certification and inspection criteria would also be reduced. Furthermore, using EMAs on an all-electric aircraft would further reduce complexity, eliminating the need for a hydraulic system. This would result in even lower operating costs if the safety-case can close on an EMA design, see Section 8.2.

14.4 Benefits of Hybrid-Electric and Turboshaft Architectures

Another interesting finding from the presented study was the observed safety benefit of moving to a hybrid-electric power system. Historical industry data was used to estimate the failure rates of the turbogenerator system, both the gas-turbine and the generator, Section 12. The PSSA conducted on the series-hybrid architecture, Section 13.2.1.2, showed very low failure rates of "power loss to motor," which is the failure gate that captures the failure of the turbogenerator and the emergency battery network to send power to the PMSMs. Functionally, it can be thought of in a similar manner to dual engine capability. However, the series-hybrid architecture increases system complexity, requiring long wire runs with highly critical operating requirements and, in the configuration analyzed, the additional weight and complexity of the interconnecting shafts, Section 9.1.2.

The dual turboshaft configuration also showed promising top-level failure rates, Section 13.2.1.3. The tQuad is about 2,700 lbs. lighter than the eQuad and 1,400 lbs. lighter than the hQuad, Table 2. The turboshaft powerplant architecture is also largely simplified from that of the hybrid-electric system because it does not include the same electrical power and distribution system demands. The lighter weight vehicle and lower complexity will result in lower aircraft unit cost and likely lower maintenance costs, as reflected in the NDARC models.

There is likely an optimum design in which integrating dual engine capability for safety can be done with minimal increases in system complexity. The optimum design would result in low aircraft weight and low unit and recurring costs, one of the largest, overarching goals for UAM to be successful. An adaptation of the series-hybrid and dual-turboshaft designs is a parallel-hybrid system in which electric motor(s) are may be used in emergency conditions or to provide boost power for specific flight conditions. In either hybrid-electric architecture, series- or parallel-hybrid, the electric propulsion controller development is critical to the safe operation of the system. Control and management of shaft and electrical power to ensure that both systems are operating as intended requires further development. Additionally, testing of the emergency backup system is required to show that it will function when it is called upon.

14.5 Incorporation of Hydrogen Fuel Systems

Another, large, overarching goal for UAM is zero-carbon-emissions which were not studied under this effort. However, the hQuad and tQuad architectures present a promising opportunity for the incorporation of liquid or gaseous hydrogen as the primary fuel to power each aircraft. Hydrogen can be delivered to fuel cells to provide electrical energy to electric motors or it can be burned in internal combustion or gas-turbine engines, neither of which produce carbon or methane emissions. Hydrogen storage and distribution in the aircraft has its own engineering challenges and a clear path to compliance with SC-VTOL-01 should be developed. The safety analysis for the hQuad and tQuad, Sections 13.2.1.2 and 13.2.1.3, respectively, included an underdeveloped fuel system that should be further developed to substantiate initial reliability claims, which were based on engineering judgment.

14.6 Overspeed

The safety analysis included an overspeed hazard captured in the FHA and FTA. In the case of the RPM-controlled hexarotor and octorotor, this turned out to be the limiting hazard in the system, resulting in catastrophic failure rates at least two orders of magnitude higher than the pitch-controlled variants.

Initially, engineering judgment was used to apply failure rates to the failure modes leading to the overspeed hazard. One percent of the failure rates of each applicable failure mode was assumed to result in a potentially catastrophic or hazardous overspeed condition, effectively reducing the sensitivity to component reliability predictions by two orders of magnitude, see Section 13.2.2.

Overspeed hazards are not new or limited to electric propulsion system, though. Gas-turbines and other high speed machines have the potential to overspeed, but incorporate design practices, compensating provisions, and flight envelope restrictions to limit overspeed failures to acceptably low probabilities of occurrence.

Additional research is required to develop a comprehensive list of failure causes leading to overspeed, but three failure causes leading to overspeed are (i) mechanical opens, (ii) controller high failures, and (iii) environmentally induced overspeed of an uncontrolled rotor.

AC 33.27-1A (ref. AC 33.27-1A) provides guidance for demonstrating compliance with overspeed hazards related to mechanical opens. Mechanical opens, as they relate to the DPFC architectures are mechanical failures, such as shaft failures, gear web failures, or PMSM armature failures that would result in sudden unloading of the system. Unloading of the system results in an energy imbalance, requiring that some components increase their speed in order to absorb and balance the energy in the system.

Controller high failures are the result of undetected inverter, motor controller, or FCC failures that would command a higher speed setting. The baseline design uses independent RPM feedback and triplex FCCs to provide independent computational hardware and software monitoring of the control functions. In the event the inverters or motor controllers do not detect a failure and properly isolate themselves the three FCCs can remove all power from the failed rotor.

Additionally, overspeed may occur if a deenergized rotor, without independent rotor control, such as blade pitch control, is allowed to fail to a state of free-rotation. Environmentally induced overspeed may occur if combinations of gusts and maneuvers can spin the rotor. Analysis and flight testing are required to determine if steady winds, continuous turbulence, or gusts can create this condition within the flight envelope. If this condition is present within the flight envelope, then the flight envelope should be restricted after a single rotor is shut down or a probabilistic determination may be suitable if the flight condition(s) of concern are sufficiently infrequent.

Although not directly captured in the language of “overspeed,” the fully articulated rotors of the hexarotor and octorotor have the potential to result in excessive flapping as the rotors slow. Excessive flapping can cause uncontrolled blade strikes and additional pilot workload. It will occur when CF loads on the blade are reduced and airflow over the rotor creates lift on a single blade element.

Overspeed, as it relates to the NASA RVLT concept vehicles, must be considered further as the design matures. It is clear that overspeed hazards exist in each vehicle considered. The pitch-controlled quadrotors and hexarotor capture mechanical opens and controller high failures within component failure rates and fault tree architectures. Additionally, as the design matures, rotating components will need to show compliance to SC E-19, EHPS.240, Overspeed and Rotor Integrity. Environmentally induced overspeed is captured in the fault tree architecture, requiring that catastrophic failures occur in both the “loss of propulsion” and “loss of control” fault trees on the same rotor. However, additional work is required to characterize the probability of overspeed occurrence in the RPM-controlled hexarotor and octorotor which do not currently include independent rotor control for overspeed prevention after a PMSM system shut down.

14.7 Single Lifting System Failure

Both the hexarotor and octorotor aircraft assumed that failure of a single lifting system allowed for continued safe flight and landing. However, it may be possible to fail more than one lifting system and continue safe flight, which was not included in the scope of this study. It is unclear if the weight of a system sized to continue safe flight after two or more lifting system failures would provide a net benefit to the aircraft or safety, but could be considered in future work.

15 SUMMARY AND CONCLUSIONS

The statement of work, S&C simulations, and probability budgets were used to develop propulsion system architectures intended to meet SC-VTOL-01 with particular emphasis on VTOL.2510(a). Seven different vehicle configurations were analyzed: the all-electric quadrotor (eQuad), hybrid-electric quadrotor (hQuad), turboshaft quadrotor (tQuad), all-electric quadrotor without shafts, pitch all-electric hexarotor (Pitch eHex), RPM all-electric hexarotor (RPM eHex), and the RPM all-electric octorotor (RPM eOct). All of the vehicle configurations analyzed showed paths to comply with VTOL.2510(a). A fail-safe design philosophy lead to similar levels of safety for all configurations evaluated. The RPM-controlled hexarotor and octorotor showed a sensitivity to overspeed hazards, but more work in this area is needed to further evaluate how uncontrolled rotors will behave within a given flight envelope.

In conclusion, the RPM-controlled hexarotor had lower transient power spikes than the RPM-controlled octorotor. However, collective control schemes and interconnecting shafts were more effective means to reduce power spikes.

The pitch-controlled hexarotor showed the lowest probability of catastrophic failure of all aircraft evaluated. The pitch-controlled quadrotors all had similar probabilistic failure rates and all were within reach of compliance with VTOL.2510(a). The eQuad without interconnecting shafts required redundant PMSM stackups with overrunning clutches to keep all four rotors spinning.

S&C simulations indicate that collective-control schemes and interconnecting shafts will increase component reliability when maneuverability and disturbance rejection are incorporated into the usage spectrum. Evaluation of the usage spectrum found that resizing motors for each, individual rotor for the hex and octorotors may lead to lower reliability to cope with the more demanding cubic mean power.

In this research, through the three primary trade studies, aircraft systems were developed and refined. This includes the development and refinement of the flight control system, the drive and power system, the thermal management system, and the electrical power distribution system. Future refinement is recommended to validate early assumptions and the feasibility of derived requirements. The flight control system should further evaluate the use of EMAs, including EMA induced transient conditions and single failure criteria. Similarly, the drive and power system refinements should include the evaluation of the means to comply with single failure criteria, VTOL.2250(c), including further evaluation of real-time diagnostics and prognostics in a rotating frame, and further evaluation, evolution, and optimization of the hybrid-electric system. The thermal management system analysis should delve further into fan and winding attributes that may affect motor reliability. Finally, the electrical power system should be refined to further define the system architecture using state-of-the-art batteries, lithium ion or similar, and power regeneration.

16 RECOMMENDATIONS FOR FUTURE WORK

It is recommended that future work continues the evolution of the RVLT quadrotor and hexarotor for further comparison. The PSSA found that hexarotor and quadrotor configurations had the lowest and second lowest probability of catastrophic failure, respectively. The quadrotor has previously been sized for multiple propulsion types, allowing for incorporation of follow-on hybrid-electric studies and is a prime candidate for further trades to comply with VTOL.2250(c) and zero-emission fuel systems. The hexarotor has been configured for both RPM- and pitch-control schemes, allowing for additional evaluation of EMAs in place of hydraulic actuation and characterization of uncontrolled rotors post propulsion failure. The octorotor could be studied for continued safe flight and landing after multiple propulsion failures, but the current study has shown multiple engineering challenges associated with continued safe flight after a single propulsion failure which is likely a stepping stone to the ability to safely permit multiple propulsion failures in flight. The following tasks have been identified to compare differences between the quadrotor and hexarotor configurations.

- NDARC sizing and performance and optimization are recommended to incorporate the improved flight safety reliability as established in the current study. Resizing the quadrotor for series or parallel hybrid system using state-of-the-art batteries and optimizing the size of the battery system for emergency use. Similarly, using state-of-the-art batteries to resize the hexarotor or incorporate a similar hybrid-electric architecture.
- Develop low gross weight and low power margin design missions to compliment the mid-gross weight design mission studied here. Transient conditions, such as maneuvers, should be considered and incorporated into the usage spectrum.
- Pilot-in-the-loop simulations are recommended using non-linear models including rotor-to-rotor interference and additional control law development on par with aviation standards to ascertain impact of failure injection and pilot reaction. Non-linear models would also be able to facilitate varying mission parameters, such as multiple gross weight conditions (low gross weight v. low power margin hover). Pilot-in-the-loop simulations could also be used to compare architectures for how easy/costly it would be to train new pilots and maintain proficiency.
- Further evaluation of overspeed and uncontrolled, deenergized rotor systems is recommended. Incorporation of additional control features like brakes or individual blade control may be required to prevent cascading failures for the RPM-controlled system. Plus an individual blade control system will allow for noise or vibration reductions that would benefit the passenger.
- Series hybrid architectures have the potential of reducing the need and associated weight and geometric constraints of gearboxes and cross shafts. Conversely, parallel hybrid architecture have the potential of reducing the need and associated weight and geometric constraints of the HVDC distribution system. This warrants a trade study showing the weight and safety sensitivities to the choice of series or parallel hybrid propulsion system, and UAM and aircraft industry emissions goals warrant evaluation of zero-emission fuel systems, such as liquid or gaseous hydrogen. When looking at various hybrid architectures, characterizing the safety value of available range given a propulsive failure will help establish the severity of propulsion failures. To make hybrid systems feasible for

flight operations from a safety and reliability perspective, there needs to be demonstration and analysis of these systems to address these issues.

- Further evaluation of VTOL.2250(c) single failure criteria and its impact on rotating structural elements is needed. Analytical modeling of fatigue crack growth and detection means (in situ monitoring, visual inspections, etc.) in single and multiple load path structures could establish design criteria and means of compliance. The quadrotor and hexarotor propulsion architectures allow for evaluation of motor, bearing, gear, and rotor shaft failures and detection methods. To meet the intent of VTOL.2250(c), there needs to be further demonstration and analysis which can be paired with hybrid-electric system demonstrations discussed previously. It is recommended that a zero-emission, hybrid-electric propulsion system test bed be developed in which potential certification methods may be examined to guide development of applicable means of compliance.
- Current work has found that air cooling and liquid cooling at this scale are feasible, but future work should consider optimizing cooling system weight vs sub-component reliability to optimize system performance and unit/operating costs. The study should include reliability impacts to design requirements like maximum operating temperatures on winding or controller reliability.
- Continuing evolution of battery and electrical power system design including packaging studies in the outer mold line (OML) and around other systems including evaluations and technology gaps of placing large battery networks near fire zones, rotating systems, or other. Battery design should evaluate safety and reliability studies in which the notional subsystem design and components are developed to meet design mission requirements using state-of-the-art cell technology, given resized vehicle characteristics using state-of-the-art power/energy densities and nominal electrical power output requirements.
- More detailed evaluations and sensitivity studies of mean time between maintenance, removal, and overhaul guidelines for UAM operating costs and safety standards. Designs developed under the current work are intended to meet VTOL.2510(a) safety requirements, but may incur undue unit or operating costs burdening the end user. Characterizing the sensitivity to design decisions in each major sub-system may uncover substantial operating and support cost savings when comparing the quadrotor to the hexarotor.

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Appendix A Statement of Work Questions, Responses, and Cross-References

Table A1, Table A2, and Table A3 include the questions addressed in the statement of work (SOW), a brief response, and the corresponding report section(s).

Table A1: SOW questions addressed in the statement of work for SOW Table A.

SOW Question #	SOW Question	Response	Reference Section
A.1	What is the impact of the number of rotors on vehicle safety and reliability, or on aircraft/component design attributes required to achieve the required safety level (=<10^{-9} failures per flight hour)?	Safety analysis shows that the RPM-controlled octorotor requires higher reliability components or additional fail-safety in order to achieve similar levels of safety to the RPM-hexarotor. S&C simulation shows higher power transients for the RPM-controlled octorotor which will result in more frequent unscheduled repairs/replacements or lower component lives or both when compared to a similarly-designed RPM-controlled hexarotor.	5.2.3 13.2
A.2	How is the per-flight-hour failure rate of power system components for RPM control affected by the number of rotors? Does an increase in the number of rotors translate to a reduced duty cycle for some components? How do additional rotors for RPM control change the assumed electrical component duty cycles used when calculating failure rates? How do these compare to steady-state operation? Assess the impact of distributing the propulsion over more rotors on total vehicle reliability.	The NDARC sizing routine used the design mission in absence of atmospheric disturbances, typical for sizing routines, to estimate propulsion system component sizing. Usage spectrums based on the design mission were used to calculate cubic mean power. NSWC-11 guidance was used to estimate the reliability of propulsion system components sized to the design mission. Increases in cubic mean power would result in lower reliability for similarly sized components. The RPM-controlled hexarotor and octorotor did not exhibit discernable differences in per-flight-hour failure rates of propulsion system components using NDARC component sizing, cubic mean power, and NSWC-11 reliability estimating techniques. However, S&C simulation results showed excessive power transients for both aircraft, with the RPM-controlled octorotor showing more cases with higher PMSM power demands. S&C simulation results indicate that increasing rotors would tend to increase cubic mean power, a surrogate for duty-cycle, when aircraft maneuvers and atmospheric disturbances are considered.	4 5.2.3 9.2.1 12.2.3
A.3	Identify the peak transient power levels required per propulsion unit and assess the impact of these power transients on component reliability. Consider the transient power required to achieve the bandwidth required for vehicle trim, maneuvering, and disturbance rejection in an urban canyon environment with non-zero, average-day gusts and winds. Reassess for an individual single motor failure.	In the presence of steady winds, the vehicle response for both configurations is benign with respect to both the engine power required and vehicle output state errors. S&C simulation results shows the distribution of maximum transient power required for disturbance rejection in the presence of atmospheric disturbances representative of an urban canyon environment for both nominal and single engine failure conditions. The supplied bare-airframe models did not include rotor-to-rotor or rotor-airframe interference, so for the purposes of this analysis, a failure in Engine 1 was injected at a prescribed simulation time for the single engine failure scenarios. Because interference effects were not included in the analysis, Engine 1 failure injections were deemed representative of all rotor locations. See response to SOW Question B.3 for bandwidth study response.	5.2.3

SOW Question #	SOW Question	Response	Reference Section
A.4	Assume that motor sensors are required for safety-critical feedback to the flight control system in support of RPM control. Assume that the required sensors are to measure the following parameters: shaft speed, voltage, current, and temperature. Assess the reliability and redundancy requirements for these sensors.	Sensors requirements were included in the reliability and safety analysis. Sensors which measure/monitor PMSM torque, PMSM shaft speed, oil temperature, oil pressure, PMSM torque ripple, bearing failures, insulation quality, and electrical short dectection are included in the reliability analysis for the drive and power system. Reliability analysis assumed that each sensor needed to function and reliability detect 99.9% (50% confidence) of all failure causes that the specific sensor was designed to detect. Redundancy management was incorporated into the design, including multiple sensors where required to comply with safety levels identified in the FHA. Where practical, different sensor types were used to monitor critical components; in example, temperature probes and temperature switches were used to develop appropriate redundancy for temperature monitoring. The temperature probe continually monitors the temperature to indicate trends and allow the pilot/crew real-time visibility into system health. The temperature switch does not provide real-time system health to the pilot/crew, but will alert the pilot/crew when oil temperature reaches a specific threshold. The aircraft was assumed to be piloted; therefore, most sensors end function was to alert the pilot/crew. Short detection was the only sensor that is assumed to take action after a fault is detected. After a short is detected the affected PMSM needs to be deenergized rapidly to not create secondary failures, likely in too short of a window for the pilot to take appropriate action. The handoffs and authority to deenergize a PMSM need additional development but will likely reside with the triplex FCC.	10.1 12.2.3 13.2
A.5	Does an increase from 6 rotors to 8 rotors permit a reduction in component redundancy without a loss of overall system reliability? For example, if the 6-rotor configuration requires two motors per shaft, can the 8-rotor configuration achieve the same level of reliability with a single motor per shaft?	Assuming similar levels of fail-safety, the RPM-controlled octorotor has an increased risk of catastrophic failure compared to the RPM-controlled hexarotor due to the additional rotors which increase the likelihood of failure. To develop an RPM-controlled octorotor with equivalent or better catastrophic failure rates to that of the RPM-controlled hexarotor, the RPM-controlled octorotor will require higher reliability components or additional redundancy to overcome its inherent increase in likelihood of failure. The change from six to eight rotors does open up the possibility dual propulsor failures may not be catastrophic; however, this would require additional S&C simulation after two motor failures and increased component requirements so that the remaining components could withstand the loads from a dual propulsor failure event.	13.2.2 13.2.3

SOW Question #	SOW Question	Response	Reference Section
A.6	Address the impacts of voltage and current levels on reliability.	There are several benefits to increasing voltage in the electrification of aircraft, but the optimal voltage for each aircraft varies. Trends in current technology are discussed to highlight the impacts of higher voltages and currents on hardware reliability. The flow of electrical current generates heat due to Joule heating, especially in the PMSM windings. Therefore, equipment and wiring will have some current limits and additional weight may be required, e.g. insulation or thermal management equipment. Increasing the system voltage will reduce current. This offers several benefits to electrical components which will improve efficiency and reduce the need for cooling equipment. On the other hand, higher voltage can have potential negative impacts to power distribution components. Higher line voltage increases motor voltage spikes that can breakdown motor winding insulation. Use of higher voltage drives the need for improved insulation technology or thicker insulation which could increase weight.	11.2.1
A.7	Assess the reliability and safety impacts of liquid cooled vs. air cooled motors and motor controllers.	Liquid (oil) cooling vs. air cooling were evaluated using a PMSM designed for both oil and air cooling. A Safran ENGINeUS 100 PMSM was used to evaluate cooling performance of each type of system. Two cooling means were investigated, as well. Cool air was drawn through the heat exchanger (oil cooled system) or around the motor periphery (air cooled system) by either ram driven effects or by a dedicated electric cooling fan. A total of four system were evaluated, (1) oil cooled, ram driven, (2) oil cooled, fan driven, (3) air cooled, ram driven, and (4) air cooled, fan driven. A comparison was made to evaluate the relative reliability of each system and the components for the oil cooled system were integrated into the aircraft FMECA. A FHA was performed and each system was arranged for appropriate levels of reliability and redundancy to comply with SC-VTOL-01 and FHA severity classifications. The air cooled, ram driven system was determined to be the lowest weight system, but the oil cooled, ram driven system had the best overall performance when including cooling capability and reliability in the decision matrix.	10.2 10.3
A.8	What are the failure modes associated with repeated over-charging and excessive discharging of the batteries? How does repeated high-rate discharging affect battery reliability?	Battery failure modes are well-understood, but are complex with many contributing factors. Cell-aging, internal short circuit, and thermal runaway are the failure modes of greatest concern. Individual cells will age at different rates than surrounding cells. This can result in over-charging, over-discharging, or increased discharge rate for individual cells that may lead to the failure modes listed above.	11.2.4

SOW Question #	SOW Question	Response	Reference Section
A.9	Does the increase from 6 rotors to 8 rotors, and the associated choice of number of motors per rotor, significantly influence the requirements for the distributed electric propulsion (DEP) components and their associated supporting systems (such as cooling, health monitoring and/or control sensors, etc.)? For example, does the choice of fewer rotors require more challenging peak-torque/power requirements for the motor and power electronics and/or significantly change the performance requirements of the thermal management system? Does the choice of more rotors result in advantageous derived reliability-requirements for the DEP components?	As seen in the NDARC sizing models, assuming no atmospheric disturbances, adding more rotors reduces the power required per rotor. However, when atmospheric disturbances are applied, the transient power requirements increase as the number of rotors increases. The choice of more rotors requires more challenging transient power requirements for the motor and power electronics.	4 5.2.3 9.2.1

Table A2: SOW questions addressed in the statement of work for SOW Table B.

SOW Question #	SOW Question	Response	Reference Section
B.1	What is the impact of rotor thrust control (collective or RPM) on vehicle safety and reliability, or on aircraft/component design attributes required to achieve the required safety level (=<10^{-9} failures per flight hour)?	See response to SOW Question A.1 for additional background. S&C simulation results indicate that collective-control schemes will inherently improve component reliability due to less demanding power transients. Safety analysis results show an appreciable difference with the pitch-controlled hexarotor having notably fewer catastrophic failures per flight hour. Also, it is noteworthy that the pitch-controlled hexarotor had the fewest catastrophic failures per flight hour of all study aircraft.	4 5.2.2 9.2.1 13.2.2
B.2	How is the per-flight-hour failure rate of power system components affected by the use of RPM control vs. collective control? How do these compare to steady-state operation? How does the use of RPM control change the assumed electrical component duty cycles used when calculating failure rates? Assess the impact of RPM control on total vehicle reliability.	See response to SOW Question A.2 for additional background. The pitch- and RPM-controlled hexarotor did not exhibit discernable differences in per-flight-hour failure rates of propulsion system components using NDARC component sizing, cubic mean power, and NSWC-11 reliability estimating techniques. However, S&C simulation results showed excessive power transients for the RPM-controlled hexarotor. S&C simulation results indicate that increasing rotors would tend to increase cubic mean power, a surrogate for duty-cycle, when aircraft maneuvers and atmospheric disturbances are considered.	4 5.2.2 9.2.1 12.2.3

SOW Question #	SOW Question	Response	Reference Section
B.3	Identify the peak transient power levels required per propulsion unit and assess the impact of these power transients on component reliability. Consider the transient power required to achieve the bandwidth required for vehicle trim, maneuvering, and disturbance rejection in an urban canyon environment with non-zero, average-day gusts and winds. Reassess for an individual single motor failure.	The distribution and worst-case transient power requirements for the pitch-controlled hexarotor are substantially improved over the RPM-controlled hexarotor. S&C simulations also experimented with 3 Hz and 7.25 Hz PMSM bandwidths with the RPM-controlled hexarotor. The RPM-controlled hexarotor was used to study the effects of 3 vs. 7.25 Hz bandwidths. Study results for the RPM-controlled hexarotor showed a trend of lower power transients when a lower (3 Hz) motor bandwidth was used, but the difference between maximum power transients for each bandwidth considered was relatively small.	5.1.2 5.2.2
B.4	Assume that motor sensors are required for safety-critical feedback to the flight control system in support of RPM control. Assume that the required sensors are to measure the following parameters: shaft speed, voltage, current, and temperature. Assess the reliability and redundancy requirements for these sensors.	See response to SOW Question A.4.	-
B.5	Address the impacts of voltage and current levels on reliability.	See response to SOW Question A.6.	-
B.6	Assess the reliability and safety impacts of liquid cooled vs. air cooled motors and motor controllers.	See response to SOW Question A.7.	-
B.7	What are the failure modes associated with repeated overcharging and excessive discharging of the batteries? How does repeated high-rate discharging affect battery reliability?	See response to SOW Question A.8.	-
B.8	Does the choice of RPM or collective control significantly influence the requirements for the distributed electric propulsion (DEP) components and their associated supporting systems (such as cooling, health monitoring and/or control sensors, etc.)? For example, does the choice of RPM vs. collective control significantly influence peak-torque/power requirements for the motor and power electronics and/or significantly change the performance requirements of the thermal management system? Does the choice RPM vs. collective control result in advantageous derived-reliability-requirements for the DEP components and associated supporting systems?	As seen in the NDARC sizing models, assuming no atmospheric disturbances, RPM-control schemes may reduce vehicle by removing actuation related components. However, when atmospheric disturbances are applied, the transient power requirements increase when switching from pitch- to RPM-control schemes. Transient power requirements are significantly lower for the pitch-controlled hexarotor when compared to the RPM-controlled hexarotor. These results indicate that transient power requirements are largely a function of rotor inertia that must be overcome in RPM control schemes for disturbance rejection in this environment. The choice of RPM-control requires addressing more challenging transient power requirements for the motor and power electronics, leading to larger, heavier rotors, lower the ability to maintain desired heading, or both. Safety analysis also shows that the pitch-controlled hexarotor has notably fewer catastrophic failures per flight hour.	4 5.2.2 13.2.2

Table A3: SOW questions addressed in the statement of work for SOW Table C.

SOW Question #	SOW Question	Response	Reference Section
C.1	What is the impact of propulsion architecture (hybrid vs. all-electric vs turboshaft) on vehicle safety and reliability, or on aircraft/component design attributes required to achieve the required safety level (=<10^{-9} failures per flight hour)?	The three different propulsion systems studied, all-electric, hybrid-electric, and turboshaft, all demonstrate a path to comply with VTOL.2510(a), ≤10^{-9} catastrophic failure criteria. The all-electric quadrotor, however, requires a very reliable battery network that may reduce mission range and useful load. The hybrid-electric system demonstrated a more practical means to comply with VTOL.2510(a), using a small onboard battery for emergency conditions. The turboshaft system also showed a practical means to comply with VTOL.2510(a), using two turboshaft engines for redundancy and similar capability to CS-27 Category "A" Rotorcraft.	9.1 13.2.1
C.2	Address the impacts of voltage and current levels on reliability.	See response to SOW Question A.6.	-
C.3	Address the impacts of turbine-generator interface design choices on system reliability. Examples: number of spools, gearbox vs. direct drive.	Reliability estimates were developed for drive and power system components that required substantial increases in catastrophic and hazardous failure rates following preliminary failure rate budgeting. Reliability estimates focused on catastrophic and hazardous failure rates to support safety analysis. The number of components included in the given system effected the reliability of the system; in other words, the more components included in the system, the more failures per flight hour. However, component reliability can also be improved to improve system reliability. In general, very high reliability components will be required in order to comply with VTOL.2510(a) and, therefore, compliance with VTOL.2510(a) is less sensitive to part count. A direct drive turbo-generator was developed in order to balance weight, torque and switching demand.	9.1.2 9.2.2 12.2.3
C.4	Address the feasibility of replacing cross-shafting with component-level redundancy and discuss the associated challenges.	Without interconnecting shafts, a significant number of hazards increase in severity from minor to catastrophic relative to the variant with interconnecting shafts. To comply with VTOL.2510(a) two different PMSM system architectures were developed for the quadrotor with and without interconnecting shafts. The all-electric quadrotor with interconnecting shafts utilized a total of four single winding PMSMs with dual channel inverters, remotely located at each rotor. The all-electric quadrotor without interconnecting shafts required a total of eight single winding PMSMs with dual channel inverters, two remotely located at each of four rotors. Both designs required overrunning clutches to ensure that all four rotors continued to provide powered lift in the event of a single PMSM or dual inverter failure. Additionally, S&C simulation results show high transient power requirements for the aircraft without interconnecting shafts, particularly in high speed forward flight. The interconnecting shafts significantly reduce the maximum transient power required to reject atmospheric disturbances.	5.2.1 7.1 9.1.1 9.1.4 13.2.1
C.5	Assess the reliability and safety impacts of liquid cooled vs. air cooled motors and motor controllers.	See response to SOW Question A.7.	-

SOW Question #	SOW Question	Response	Reference Section
C.6	Collective control changes on the rotors will lead to fluctuations in the power required from the turbo-generator, increasing its duty cycle and likelihood of failure. Does an increase in battery capacity to serve as a buffer against power demand fluctuations on the turbogenerator lead to a more reliable or less reliable system? Explain and quantify.	This is a system level trade study which includes power demand/regen analysis, thermal analysis, accurate system modeling, weight, reliability and safety considerations. Batteries can serve as buffers to mitigate power spikes at the turbogenerator, and there are many flying examples of this type configuration. Also, the generator size can be increased to accommodate power spikes, and there are many flying examples of this type of configuration. Relative to overall reliability, similar type and quantity of components are used in both system configurations, but design practices and the ability to accurately predict and design equipment for power spikes will be the larger driver as related to overall reliability.	11.2.3
C.7	What are the failure modes associated with repeated overcharging and excessive discharging of the batteries? How does repeated high-rate discharging affect battery reliability?	See response to SOW Question A.8.	-

Appendix B Functional Hazard Analysis

Functional Hazard Analysis (FHA) tables for the electric quadrotor (eQuad), hybrid-electric quadrotor (hQuad), turboshaft quadrotor (tQuad), eQuad without cross-shafting, collective controlled hexarotor (Pitch eHex), and RPM-controlled hexarotor (RPM eHex) are provided in Table B1, Table B2, Table B3, Table B4, Table B5, and Table B6, respectively. An independent FHA table for the RPM-controlled electric octorotor (eOct) was not developed as the FHA for the RPM-Hex is identical based on current design assumptions.

The tables are split into functional hazards associated with the Loss of Propulsion and Loss of Collective Control (for applicable configurations) with orange and red color indicators (respectively) to organize these aircraft level function failure sections, consistent with the FBD and PSSA.

Table B1: Quad FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Any loss of single propulsor fail	All	Aircrew detects failure and compensates with remaining thrust to continue flight. Cross-shafting results in all rotors continuing to spin. Power available is greater than Power required (Pa>Pr). Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	
Any combination of Dual propulsor Fail	All	Failures are detected. Cross-shafting ensures all rotors are still spinning. Controllability still present. Reduced power available. Insufficient power to maintain level fight. Autorotative approach requires suitable landing area. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report	FTA	
FCC Fail	All	Failures are detected. Loss of reference speed from FCC. The ESC will default to programmed nominal speed. Degraded control and maneuverability. Increased pilot load.	Minor	NASA AHS report, FMECA	FTA	Failure classification Minor based on ground rule: ESC pre-programmed speed allows for reasonable control - land as soon as practical condition
LV DC Fail	All	Loss of power to all 4 ESC and FCC. Collective control of rotor lost. Loss of flight Path Control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Dual ESC Fail	Dual ESC Failed High: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Pilots will need to reduce engine power to land. If hover power can be properly managed, the pilot will ba able to land normally. Worst case feasible outcome is air-vehicle damage and occupant injury.	Severe	TBD, FMECA	TBD	
	Dual ESC Failed Low: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Reduced power available. Insufficient power to maintain level fight. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	TBD, FMECA	FTA	
Single ESC Fail	ESC Failed Hi: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Pilots will need to manually modulate engine power to a hover landing or a no hover landing with some forward speed to maximize Effective Translational Lift (ETL). Increased pilot load.	Minor	NASA AHS report; FMECA	TBD	
	ESC Failed Low: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Power available is greater than Power required (Pa>Pr). Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	
Single Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotor associated with that gearbox. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	Compliance with VTOL.2250(c). Consider dual load path design. Real time health monitoring to detect this and annunciate to crew - robust detection of prescribed failure modes.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Dual Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotors associated with those gearbox. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	
Combiner transmission/Cross shaft fail	All	Annunciated to pilot. Possible minor handling qualities impact, lack of redundancy available for follow-on propulsion single or dual failures. This fail in and of itself is not Catastrophic.	Minor	FMECA	FTA	Need proper anti-flail in place on driveshaft
Complete HVDC Fail	All	Loss of High Voltage Power to motors. Power available is less than Power required (Pa<Pr). Complete loss of propulsion. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report, layout diagrams	FTA	
HV Batt Thermal Runaway	All	Thermal runaway of cell(s). Worst case scenario is propagation throughout module. Results in release of fire, smoke or toxic gases. Adverse impact to adjacent flight critical systems, occupants, or ground personnel. Worst case scenario is loss of air-vehicle, occupant, or personnel.	Catastrophic	TBD, FMECA	FTA	Derived safety requirements such as those set forth in FAA SC Requirements AC 20-184 for 14 CFR Parts 23, 25, 27 & 29
Individual portions of HVDC Fail	All	Reduced power to one or more motors. Power available is greater than Power required (Pa>Pr). Aircrew detects failure and compensates with remaining thrust to continue flight. Cross-shafting results in all rotors continuing to spin.	Minor	FMECA	FTA	
Collective Control of Rotors						

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
FCC/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants. End effect could impact failure condition (fail fixed versus hard over).	Catastrophic	FMECA	FTA	Top level derived safety requirements: DAL A FCC detects and annunciates to crew - robust detection of actuation losses
LV DC Fail	All	Loss of power to all 4 ESC and FCC. Collective control of rotor lost. Loss of flight path control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.
Mechanical failure	All	Failures detected and annunciated to aircrew by real time health monitoring system. Loss of ability to actuate or interface with a single rotor system. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Critical parts list/inspection/life limit Objective: Real time health monitoring can detect this and DAL A FCCs to annunciate to crew - robust detection of actuation losses
Actuator failure	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to a single rotor (actuation system). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: DAL A FCC detects and annunciates to crew - robust detection of actuation losses
Actuation power failure	All	Loss of power to single rotor pitch control. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Note: system definition could influnce end effect (actuator float vs fail fixed vs fail to home)

Table B2: hQuad FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Any loss of single propulsor fail	All	Aircrew detects failure and compensates with remaining thrust to continue flight. Cross-shafting results in all rotors continuing to spin. Power available is greater than Power required (Pa>Pr). Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	
Any combination of Dual propulsor Fail	All	Failures are detected. Cross-shafting ensures all rotors are still spinning. Controllability still present. Reduced power available. Insufficient power to maintain level fight. Autorotative approach requires suitable landing area. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report	FTA	
FCC Fail	All	Failures are detected. Loss of reference speed from FCC. The ESC will default to programmed nominal speed. Degraded control and maneuverability. Increased pilot load.	Minor	NASA AHS report, FMECA	FTA	Failure classification Minor based on ground rule: ESC pre-programmed speed allows for reasonable control - land as soon as practical condition
LV DC Fail	All	Loss of power to all 4 ESC and FCC. Collective control of rotor lost. Loss of flight Path Control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Dual ESC Fail	Dual ESC Failed High: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Pilots will need to reduce engine power to land. If hover power can be properly managed, the pilot will ba able to land normally. Worst case feasible outcome is air-vehicle damage and occupant injury.	Severe	TBD, FMECA	TBD	
	Dual ESC Failed Low: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Reduced power available. Insufficient power to maintain level fight. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	TBD, FMECA	FTA	
Single ESC Fail	ESC Failed Hi: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Pilots will need to manually modulate engine power to a hover landing or a no hover landing with some forward speed to maximize Effective Translational Lift (ETL). Increased pilot load.	Minor	NASA AHS report; FMECA	TBD	
	ESC Failed Low: All phases	Failures are detected. Cross shafting ensures all rotors are still spinning. Controllability still present. Power available is greater than Power required (Pa>Pr). Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotor associated with that gearbox. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	Compliance with VTOL.2250(c). Consider dual load path design. Real time health monitoring to detect this and annunciate to crew - robust detection of prescribed failure modes.
Dual Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotors associated with those gearbox. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	
Combiner transmission/Cross shaft fail	All	Annunciated to pilot. Possible minor handling qualities impact, lack of redundancy available for follow-on propulsion single or dual failures. This fail in and of itself is not Catastrophic.	Minor	FMECA	FTA	Need proper anti-flail in place on driveshaft
Complete HVDC Fail	All	Complete loss of all high voltage power distribution to motors. Complete loss of propulsion. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report, layout diagrams	FTA	
HV Batt Thermal Runaway	All	Thermal runaway of cell(s). Worst case scenario is propagation throughout module. Results in release of fire, smoke or toxic gases. Adverse impact to adjacent flight critical systems, occupants, or ground personnel. Worst case scenario is loss of air-vehicle, occupant, or personnel.	Catastrophic	TBD, FMECA	FTA	Derived safety requirements such as those set forth in FAA SC Requirements AC 20-184 for 14 CFR Parts 23, 25, 27 & 29

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Individual portions of HVDC Fail	All	Reduced power to one or more motors. Power available is greater than Power required (Pa>Pr). Aircrew detects failure and compensates with remaining thrust to continue flight. Cross-shafting results in all rotors continuing to spin.	Minor	FMECA	FTA	
HV Batt Network Fail	All	Loss of high voltage power from battery systems to motors. Failure annunciated to pilot. Aircraft power defaults to turbo-generator system. Power available is greater than Power required (Pa>Pr). Worst case scenario is fire and potential loss of air vehicle/occupants.	Catastrophic	NASA AHS report, layout diagrams, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that generator alone can power a safe controlled landing; however, if energetic failure such as 1E4, the fire could be catastrophic
Turbo Shaft Engine Fail	All	Failure annunciated to pilot. Aircraft power defaults to battery systems. Battery power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that battery system alone can power a safe controlled landing
Engine Gearbox Fail	All	Failure annunciated to pilot. Aircraft power defaults to battery systems. Battery power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that battery system alone can power a safe controlled landing
AC Generator Fail	All	Failure annunciated to pilot. Aircraft power defaults to battery systems. Battery power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that battery system alone can power a safe controlled landing
AC/DC Converter Fail	All	Failure annunciated to pilot. Aircraft power defaults to battery systems. Battery power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that battery system alone

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
						can power a safe controlled landing
Battery to Generator Hand Off Fail	Ascent to Forward Flight Transition	Failure annunciated to pilot. Aircraft power defaults to battery systems. Power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	Define minimum battery charge for safe operation
Generator to Battery Hand Off Fail	Forward Flight to Descent Transition	Failure annunciated to pilot. Aircraft power defaults to turbo-generator system. Power available is greater than Power required (Pa>Pr).	Minor	FMECA, NDARC Model/sizing	FTA	NDARC model indicated/used to determine that generator alone can power a safe controlled landing
Collective Control of Rotors						
FCC/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants. End effect could impact failure condition (fail fixed versus hard over).	Catastrophic	FMECA	FTA	Top level derived safety requirements: DAL A FCC detects and annunciates to crew - robust detection of actuation losses
LV DC Fail	All	Loss of power to all 4 ESC and FCC. Collective control of rotor lost. Loss of flight path control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Mechanical failure	All	Failures detected and annunciated to aircrew by real time health monitoring system. Loss of ability to actuate or interface with a single rotor system. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Critical parts list/inspection/life limit Objective: Real time health monitoring can detect this and DAL A FCCs to annunciate to crew - robust detection of actuation losses
Actuator failure	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to a single rotor (actuation system). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: DAL A FCC detects and annunciates to crew - robust detection of actuation losses
Actuation power failure	All	Loss of power to single rotor pitch control. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Note: system definition could influnce end effect (actuator float vs fail fixed vs fail to home)

Table B3: tQuad FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Single Engine Fails	All	Failures detected and annunciated to aircrew. Loss of power from a single turboshaft engine. Cross-shafting with redundant turboshaft engine provide power available greater than power required. Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	Assumption: Engines are physically separated such that a rotorburst failure in one engine would not damage the other. Note: Can go OEI therefore loss of transmission from a single engine is not catastrophic Note: Hydraulic actuators are sized to handle load on single boost (loss of two hydraulic systems)
Dual Engines Fail	All	Failures detected and annunciated to aircrew. Loss of power from both turboshaft engines. Autorotative approach requires suitable landing area. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	FMECA	FTA	Assumption: Hydraulic power maintained as pumps are driven by rotor system. Battery power for control computers. Assumption: Pre-flight plan includes necessary altitude for autorotative recovery and controlled, emergency decent.
Single Engine Gearbox or Overrunning Clutch Fails Open	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of power transmission from a single turboshaft engine. Cross-shafting with redundant turboshaft engine provide power available greater than power required (Pa>Pr).	Minor	FMECA	FTA	Assumption: Turboshafts are sized for OEI therefore loss of transmission from a single engine is not catasttrophic Note: Hydraulic actuators are sized to handle load on single boost (loss of two hydraulic systems)

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Engine Gearbox Fails Closed	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to transmit torque out of gearbox. Potential for structural damage causing damaging to interconnecting shafting or combiner transmission. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	FMECA	FTA	Assumption: Turboshafts are sized for OEI therefore loss of transmission from a single engine is not catasttrophic Note: Hydraulic actuators are sized to handle load on single boost (loss of two hydraulic systems)
Dual Engine Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of power transmission from both turboshaft engines. Autorotative approach requires suitable landing area. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	FMECA	FTA	Assumption: Hydraulic power maintained as pumps are driven by rotor system. Battery power for control computers. Note: Hydraulic actuators are sized to handle load on single boost (loss of two hydraulic systems)
Combiner transmission/Cross shaft fail	All	Annunciated to pilot. Loss of ability to spin one or two rotors associated with that transmission or shaft. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Need proper anti-flail in place on driveshaft
Single Rotor Gearbox Fails	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotor associated with that gearbox. Potential for structural damage causing damaging to interconnecting shafting. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	Compliance with VTOL.2250(c). Consider dual load path design. Real time health monitoring to detect this and annunciate to crew - robust detection of prescribed failure modes.
Collective Control of Rotors						

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
FCC/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants. End effect could impact failure condition (fail fixed versus hard over).	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant flight control system. DAL A FCC detects and annunciates to crew - robust detection and fault reaction to flight control system failures.
Low Voltage Electrical Power Fail (DC)	All	Loss of power to all FCC. Collective control of rotor lost. Loss of flight path control and air vehicle	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant flight control system power.
Mechanical failure (Open or Jam)	All	Failures detected and annunciated to aircrew by real time health monitoring system. Loss of ability to actuate or interface with a single rotor system. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Critical Safety Item program/list/inspection/life limit. Consider dual load path design. Objective: Real time health monitoring can detect this and DAL A FCCs to annunciate to crew - robust detection of actuation losses.
Actuator failure	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to a single rotor (actuation system). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant flight control actuator power stages. DAL A FCC detects and annunciates to crew - robust detection of actuation losses

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Actuation power failure	All	Loss of power to single rotor pitch control. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: redundant hydraulic power sources.

Table B4: eQuad without Cross-Shafting FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Any loss of single propulsor fail	All	Failures detected and annunciated to aircrew. Loss of ability to power single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	Note: same effect for any combination of dual propulsor fail
Single motor failed electrically open (no power)		Low RPM. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Top level derived safety requirements: redundant Motor required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: same effect for any combination of dual fail of ESC function
Single motor failed internally shorted		Low RPM. Short acts as rotor brake. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Top level derived safety requirements: redundant Motor required. Reliable fault detection and isolation needed to ensure failed Motor is shut off. Remaining motor must have enough power to overcome short circuit damping drag load.
Single motor failed jammed		Rotor locked up. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants. Potential fire hazard.	Catastrophic			Top level derived safety requirements: redundant Motor required. Overrunning clutches incorporated to prevent torque spike. Reliable fault detection and isolation needed to ensure failed Motor is shut off.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
						Derived requirement: Need to detect, contain, and/or prevent fire. Best design practices (e.g., non-combustible insulation, lightning input, protection of phases).
FCC Fail	All	Failures are detected. Loss of reference speed from FCC. The ESC will default to programmed nominal speed. Degraded control and maneuverability. Increased pilot load.	Minor	NASA AHS report, FMECA	FTA	Failure classification Minor based on ground rule: ESC preprogrammed speed allows for reasonable control - land as soon as practical condition
LV DC Fail	All	Loss of power to all 4 ESCs and FCCs. Collective control of rotor lost. Loss of flight Path Control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.
ESC Failed (RPM High)	All	Failures are detected. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: same effect for any combination of dual fail of ESC function

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
ESC Failed (RPM Low)	All	Failures are detected. Autorotative landing required. Possible loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	FMECA	FTA	Top level derived safety requirements: redundant ESC required. Note: Incorporating overrunning clutches may reduce severity of motor jam. Note: same effect for any combination of dual fail of ESC function
ESC Failed (No power)	All	Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Top level derived safety requirements: redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: same effect for any combination of dual fail of ESC function
ESC Failed (Oscillating)	All	Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Loss of rotating machinery may also be attributed to transient torques, in addition to CF. Note: Potential motor over heating. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: same effect for any combination of dual fail of ESC function

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of ability to spin rotor associated with that gearbox. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants	Catastrophic	FMECA	FTA	
HVDC Fail	All	Loss of High Voltage Power to any motor. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report, layout diagrams	FTA	Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.
HV Batt Thermal Runaway	All	Thermal runaway of cell(s). Worst case scenario is propagation throughout module. Results in release of fire, smoke or toxic gases. Adverse impact to adjacent flight critical systems, occupants, or ground personnel. Worst case scenario is loss of air-vehicle, occupant, or personnel.	Catastrophic	TBD, FMECA	FTA	Derived safety requirements such as those set forth in FAA SC Requirements AC 20-184 for 14 CFR Parts 23, 25, 27 & 29
Collective Control of Rotors						
FCC/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants. End effect could impact failure condition (fail fixed versus hard over).	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant ESC required. Reliable fault detection and isolation needed to ensure failed FCC is shut off. DAL A FCC detects and annunciates to crew - robust detection of actuation losses
LV DC Fail	All	Loss of power to all 4 ESC and FCCs. Collective control of rotor lost. Loss of flight path control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	**Phase of Operation**	**Effect of the Failure condition on aircraft/crew**	**Classification of Failure Condition**	**Reference to supporting Mat'l**	**Verification**	**Derived Safety Requirements and Notes**
Mechanical failure (open or jam) in cockpit controls or actuation controls.	All	Failures detected and annunciated to aircrew by real time health monitoring system. Loss of ability to actuate or interface with a single rotor system. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Critical parts list/inspection/life limit Objective: Real time health monitoring can detect this and DAL A FCCs to annunciate to crew - robust detection of actuation losses
Actuator failure	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to a single rotor (actuation system). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant actuator power stages. DAL A FCC detects and annunciates to crew - robust detection of actuation losses
Actuation power failure	All	Loss of power to single rotor pitch control. Loss of ability to control pitch of a single rotor. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant actuator power systems. Note: system definition could influence end effect (actuator float vs fail fixed vs fail to home)

Table B5: Pitch-Hex FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Any loss of single propulsor fail	All	Failures detected and annunciated to aircrew. Loss of ability to power single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of a single rotor.
Loss of rotor drive (loss of power or open drive)		Low thrust to single rotor. All remaining rotors maintain flight path. Pitch control maintained. Degraded control and maneuverability. Increased pilot work load.	Minor			Assume control system at that rotor can manage rotor system speed to prevent over-speed and undesirable thrust.
Single rotor motor function failed internally shorted		Low RPM to single rotor. Short acts as rotor brake. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor			Assume control system at that rotor can manage rotor system speed to prevent over-speed and undesirable thrust. Note: assumes this is a transient condition. Eventually this effect may result in motor seizure from overheating. May need to shut down motor drive.
Single rotor motor function failed jammed		Rotor locked up. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential fire hazard. Worst case feasible outcome is loss of air-vehicle/occupant due to fire hazard.	Catastrophic			Assume hexacopter configuration can continue safe flight and landing with loss of a single rotor. Derived requirement: Need to detect, contain, and/or prevent fire. Best design practices (e.g., non-combustible insulation, lightning input, protection of phases).

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Any dual propulsor fail	All	Failures detected and annunciated to aircrew. Loss of ability to power two rotors. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Assume hexacopter configuration cannot continue safe flight and landing with loss of two or more rotors.
Dual Rotor Failure of any Motor Functions	All	Any combination of motor failures to two rotors. Failures are detected. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: redundant Motor required. Reliable fault detection and isolation needed to ensure failed ESC is shut off.
FCS Fail	All	Failures are detected. Loss of reference speed from FCC. The ESCs default to programmed nominal speed. Degraded control and maneuverability. Increased pilot load.	Minor	NASA AHS report, FMECA	FTA	Failure classification Minor based on ground rule: ESC preprogrammed speed allows for reasonable control - land as soon as practical condition
LV DC Fail	All	Loss of power to all 6 ESCs and FCCs. Collective control of rotors lost. Loss of flight Path Control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure, redundancy needs to be addressed during system design.
Single Rotor ESC Function Failed (RPM High)	All	Failures are detected. Possible loss of rotating machinery parts. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: same effect for any combination of dual fail of ESC function
Single Rotor ESC Function Failed (RPM Low)	All	Failures are detected. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of of rotor.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Rotor ESC Function Failed (No power)	All	Failures are detected. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential reduction in component fatigue life.	Minor			Assume control system at that rotor can manage rotor system speed to prevent over-speed and undesirable thrust.
Single Rotor ESC Function Failed (Oscillating)	All	Failures are detected. Rotor drive will be shut off. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to transient torques or over-speed condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Derived safety requirements: Reliable fault detection and isolation needed to ensure failed ESC is shut off. Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., mechanical brake). Over-speed condition detection limit should account for oscillatory failure modes such that values less than over-speed do not result in structural exceedance. Investigate feasibility if collective control may preclude need for mechanical braking.
Dual Failure of any ESC Functions	All	Any combination of ESC failure to two or more rotors. Failures are detected. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Need redundant ESC's.
Single Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). All remaining rotors maintain flight path. Degraded	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of of rotor.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
		control and maneuverability. Increased pilot work load.				
Dual Gearbox Fail	All	Dual Gearbox Fail. Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	
Single Gearbox Fail (Jam or rotor shaft mechanical open)	All	Loss of ability to spin rotor associated with that gearbox. Potential for structural damage causing damage to interconnecting shafting. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic			Compliance with VTOL.2250(c). Consider dual load path design. Real time health monitoring to detect this and annunciate to crew - robust detection of prescribed failure modes.
Single Gearbox Fail (Gear system mechanical open)		Loss of ability to transmit torque. Failures are detected. Possible loss of rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Compliance with VTOL.2250(c). Consider dual load path design. Real time health monitoring to detect this and annunciate to crew - robust detection of prescribed failure modes.
Loss of HVDC to a Single Motor	All	Loss of High Voltage Power to any motor function to single rotor. Low thrust to single rotor. All remaining rotors maintain flight path. Pitch control maintained. Degraded control and maneuverability. Increased pilot work load.	Minor	NASA AHS report, layout diagrams	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of a single rotor. Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Loss of HVDC to Two or More Motors	All	Loss of High Voltage Power to two or more motor functions. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.
Complete HVDC System Failure	All	Complete loss of High Voltage Power System. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.
HV Batt Thermal Runaway	All	Thermal runaway of cell(s). Worst case scenario is propagation throughout module. Results in release of fire, smoke or toxic gases. Adverse impact to adjacent flight critical systems, occupants, or ground personnel. Worst case scenario is loss of air-vehicle, occupant, or personnel.	Catastrophic	TBD, FMECA	FTA	Derived safety requirements such as those set forth in FAA SC Requirements AC 20-184 for 14 CFR Parts 23, 25, 27 & 29
Collective Control of Rotors						
Single Rotor Functional FCS/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of a single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Top level derived safety requirements: Redundant ESC required. Reliable fault detection and isolation needed to ensure failed FCC is shut off. DAL A FCC detects and annunciates to crew - robust detection of actuation losses

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Dual Rotor Functional FCS/Interface Fail	All	Failures detected and annunciated to aircrew. Loss of ability to control pitch of two rotors. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant ESC and FCS required. Reliable fault detection and isolation needed to ensure failed FCC's are shut off. DAL A FCC detects and annunciates to crew - robust detection of actuation losses
Complete LV DC Fail	All	Loss of power to all 6 ESC and FCCs. Collective control of rotors lost. Loss of flight path control and air vehicle	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure, redundancy needs to be addressed during system design.
Single Mechanical open failure in rotor controls	All	Failures detected (e.g., blade pitch sensor) and annunciated to aircrew by real time health monitoring system. Loss of ability to control blade pitch on a single rotor system. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot load.	Minor	TBD	TBD	Assume hexacopter configuration can continue safe flight and landing with loss of of rotor.
Dual Mechanical failure (open) actuation controls	All	Failures detected (e.g., blade pitch sensor) and annunciated to aircrew by real time health monitoring system. Loss of ability to control blade pitch on a two rotor systems. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Critical parts list/inspection/life limit Objective: Real time health monitoring can detect this and DAL A FCCs to annunciate to crew - robust detection of actuation losses

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single mechanical jam failure in rotor controls	All	Failures detected and annunciated to aircrew by real time health monitoring system. Loss of ability to actuate or interface with a single rotor system. Large bias thrust on one rotor. If rotor drive shut off, overspeed may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	TBD	TBD	Derived requirement: DAL A FCCs to detect and isolate failure and provide adaptive control laws as needed. Need to manage rotor system speed to prevent over-speed and undesirable bias thrust.
Actuator control failure to a Single Rotor (hardover or oscillation)	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to a single rotor (actuation system). All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of one rotor.
Actuator control failure to Dual Rotors (hardover or oscillation)	All	Failures detected and annunciated to aircrew. Loss of ability for control system to send a derived input to two rotors (actuation system). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant actuator power stages. DAL A FCC detects and annunciates to crew - robust detection of actuation losses
Actuation power failure to a Single Rotor	All	Failures detected (e.g., blade pitch sensor) and annunciated to aircrew by real time health monitoring system. Loss of ability to control blade pitch on a single rotor system. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of of rotor. Blade design needs to ensure stable blade operation perhaps in a trim position when unretained in the pitch axis. Note: system definition could influnce end effect (actuator

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
						float vs fail fixed vs fail to home)
Actuation power failure to Dual Rotors	All	Failures detected (e.g., blade pitch sensor) and annunciated to aircrew by real time health monitoring system. Loss of ability to control blade pitch on a dual rotor systems. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Top level derived safety requirements: Redundant actuator power systems.

Table B6: RPM-Hex FHA Table

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Transmit Adequate Power to Rotors						
Any loss of single propulsor fail	All	Failures detected and annunciated to aircrew. Loss of ability to power single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of a single rotor.
Loss of rotor drive (loss of power or open drive)		No applied thrust to single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential over-speed rotor condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Derived requirement: Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., brake).
Single motor failed internally shorted		Short acts as rotor brake. Failures are detected. Rotor drive will be shut off. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition. Potential motor over-heating. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			The short acting as a rotor brake may not adequately slow rotor to a safe speed. Derived requirements: Need ability to open motor drive circuits. Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., mechanical brake). Eventually this effect may result in motor seizure from overheating. Need ability to manage motor over-heating.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single motor failed jammed		Rotor locked up. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential fire hazard. Worst case feasible outcome is loss of air-vehicle/occupant due to fire hazard.	Catastrophic			Assume hexacopter configuration can continue safe flight and landing with loss of a single rotor. Derived requirement: Need to detect, contain, and/or prevent fire. Best design practices (e.g., non-combustible insulation, lightning input, protection of phases).
Any dual propulsor fail	All	Failures detected and annunciated to aircrew. Loss of ability to power two rotors. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	Assume hexacopter configuration cannot continue safe flight and landing with loss of two or more rotors.
Dual Failure of any Motor Functions	All	Any combination of motor failures to two rotors. Failures are detected. Autorotative landing required. Possible loss rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: redundant Motor required. Reliable fault detection and isolation needed to ensure failed ESC is shut off.
Complete FCS fail	All	Loss of complete FCS. Loss of air-vehicle/occupant.	Catastrophic			Derived Safety Require: Redundant FCS required.
Single Rotor Functional FCS/Interface Fail	All	Failures are detected. ESC loses RPM loop closure commands from FCS. Loss of reference speed from FCC. The ESCs default to zero RPM. Degraded control and maneuverability. Increased pilot load.	Minor	NASA AHS report, FMECA	FTA	Failure classification Minor based on ground rule: ESC will command rotor to zero RPM and act as rotor brake - land as soon as practical condition

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Dual Rotor Functional FCS/Interface Fail	All	Failures are detected. ESC loses RPM loop closure commands from FCS. Loss of reference speed from FCC. The ESC default to zero RPM at two rotors. Autorotative landing required. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report, FMECA	FTA	Derived Safety Requirement: Redundant FCS channels required. Reliable fault detection and isolation needed. Note: Investigate feasibility of operating on four rotors (e.g., motor sizing) or auto-rotation.
LVDC Fail	All	Loss of power to all ESCs and FCCs. Loss of flight Path Control and air vehicle.	Catastrophic	FMECA	FTA	Underdeveloped LV Battery as a single point catastrophic failure, redundancy needs to be addressed during system design.
Single Rotor ESC Function Failed (RPM High)	All	Failures are detected. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: Redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Need to manage rotor system speed to prevent over-speed (e.g., brake). Note: same effect for any combination of dual rotor fail of ESC function
Single Rotor ESC Function Failed (RPM Low)	All	Failures are detected. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of single rotor.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Rotor ESC Function Failed (No power)	All	No applied thrust to single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential over-speed rotor condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Derived requirement: Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., brake)
Single Rotor ESC Function Failed (Oscillating)	All	Failures are detected. Rotor drive will be shut off. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to transient torques or over-speed condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Derived safety requirements: Reliable fault detection and isolation needed to ensure failed ESC is shut off. Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., mechanical brake). Over-speed condition detection limit should account for oscillatory failure modes such that values less than over-speed do not result in structural exceedance.
Dual Failure of any ESC Functions	All	Baseline assumption: Any combination of ESC failure to two or more rotors results in catastrophic loss of air-vehicle/occupant.	Catastrophic	NASA AHS report; FMECA	TBD	Top level derived safety requirements: Redundant ESC required. Reliable fault detection and isolation needed to ensure failed ESC is shut off. Note: Investigate feasibility of operating on four rotors (e.g., motor sizing) or auto-rotation.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Gearbox Fail	All	Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load.	Minor	FMECA	FTA	Assume hexacopter configuration can continue safe flight and landing with loss of single rotor.
Dual Gearbox Fail	All	Dual Gearbox Fail. Failures detected and annunciated to aircrew (chip light, temp/ pressure indications). Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic	FMECA	FTA	
Single Gearbox Fail (Jam or rotor shaft mechanical open)	All	Loss of ability to spin rotor associated with that gearbox. Potential for structural damage causing damaging to interconnecting shafting. Loss of flight-path control and subsequent catastrophic loss of air vehicle/occupants.	Catastrophic			Flight Control system shall allow aircraft to be reconfigured during a single gearbox failure scenario such that safe flight to landing will be able to be accomplished. Propulsion system shall allow aircraft to be reconfigured during a single gearbox failure scenario such that safe flight to landing will be able to be accomplished.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
Single Gearbox Fail (Gear system mechanical open)		Loss of ability to transmit torque. Failures are detected. Possible loss of rotating machinery parts or loss of control may occur. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic			Flight Control system shall allow aircraft to be reconfigured during a single gearbox failure scenario such that safe flight to landing will be able to be accomplished. Propulsion system shall allow aircraft to be reconfigured during a single gearbox failure scenario such that safe flight to landing will be able to be accomplished.
Loss of HVDC to a Single Rotor Motor Function	All	No applied thrust to single rotor. All remaining rotors maintain flight path. Degraded control and maneuverability. Increased pilot work load. Potential over-speed rotor condition. Worst case feasible outcome is loss of air-vehicle/occupant.	Catastrophic	NASA AHS report, layout diagrams	FTA	Derived requirement: Need to manage rotor system speed to prevent over-speed and undesirable thrust (e.g., brake) Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.
Complete HVDC System Failure	All	Loss of power to all motors. Loss of flight Path Control and air vehicle.	Catastrophic			Underdeveloped HV Battery as a single point catastrophic failure needs to be addressed during system design.

Failure Conditions	Phase of Operation	Effect of the Failure condition on aircraft/crew	Classification of Failure Condition	Reference to supporting Mat'l	Verification	Derived Safety Requirements and Notes
HV Batt Thermal Runaway	All	Thermal runaway of cell(s). Worst case scenario is propagation throughout module. Results in release of fire, smoke or toxic gases. Adverse impact to adjacent flight critical systems, occupants, or ground personnel. Worst case scenario is loss of air-vehicle, occupant, or personnel.	Catastrophic	TBD, FMECA	FTA	Derived safety requirements such as those set forth in FAA SC Requirements AC 20-184 for 14 CFR Parts 23, 25, 27 & 29

Appendix C Drive and Power System Failure Modes, Effects, and Criticality Analysis

The drive and power system needed to substantiate substantial decreases in failure rate in order to meet preliminary system safety assessment (PSSA) requirements. Outputs from PSSA budgeting required that gearbox and motor failure rates needed to be less than or equal to 5 x 10^{-11} and less than or equal to 1 x 10^{-6}, respectively. Literature searches were not able to find existing hardware with that met PSSA budgets; however, anecdotal experience has shown helicopter gearboxes to have appreciably low failure rates so a search for a method to estimate failure rates based on initial architecture, functions, and design assumptions (or requirements) was performed. A similar method to that described by Smolders, et al[1] and in NSWC-11[2] was employed to predict failure rates using information available during conceptual design and creating derived requirements where additional information was required to complete the analysis.

The drive and power system developed a series of system and sub-system criticality tables to capture the failure rate of the specific failure mode, the criticality and severity of each failure mode, the number of parts within each system that could reasonably be expected to exhibit the given failure mode, and the failure rate and severity of the given system. Functional block diagrams were developed to appropriate levels so that failure modes and associated failure rates could be developed using NWC-11 guidance, see Section 12. The failure modes and failure rates were then tabulated and assigned severities and probability of detection using a series of failure mode, effects, and criticality analyses (FMECAs).

Each gearbox, motor and shaft system are summarized for reference purposes, in order to review system-by-system sensitivities. Table C1, Table C3, Table C5, Table C7, and Table C9 summarize the estimated failure rates for the rotor gearbox, the motor gearbox, PMSM, mix box, and interconnecting shaft systems, respectively. Table C2, Table C4, Table C6, Table C8, and Table C10 summarize the number of parts that could reasonably be expected to exhibit the given failure mode and the associated failure rate for the rotor gearbox, the motor gearbox, PMSM, mix box, and interconnecting shaft systems, respectively. The criticality from the applicable FMECA failure modes was multiplied by the number of parts to calculate the failure rates presented in those tables.

Table C11 is the compiled drive and power system FMECA. The gearbox, motor, and shaft system FMECAs were compiled for convenience. The mode ratio is the inverse of the sum of the applicable failure mode (i.e. bearing failure). Sensors are used to detect specified failure causes and alert the pilot and crew. Senor reliability is captured in the Failure Effect Probability and the sensor type was captured in Beta Mode Explanation. For initial reliability analysis and requirements definition each sensor is assumed to be able to accurately detect 99.9% of the noted failure cause, with 50% confidence, and subsequently alert the pilot and crew. Multiple sensors may be employed for a single failure cause, in which case the sensor reliability was "anded" together to calculate the failure effect probability. A failure cause with a single sensor will have 1E-3 probability that the end effect will result in the identified severity. A failure cause with two sensors will have a 1E-6 probability that the end effect will result in the identified severity. And a failure cause with three sensors will have a 1E-9 probability that the end effect will result in the identified severity.

[1] Smolders, K., Long, H., Feng, Y., and Tavner, P., "Reliability Analysis and Prediction of Wind Turbine Gearboxes," European Wind Energy Conference (EWEC 2010), Warsaw, Poland, 2010.

[2] Naval Surface Warfare Center, Carderock Division, "Handbook of Reliability Prediction Procedures for Mechanical Equipment," NSWC-11, 2011.

The electric quadrotor (eQuad) functional hazard analysis (FHA) was used to inform the drive and power system FMECA. The remaining aircraft analyzed elsewhere in this study used the criticality and failure rates from the eQuad drive and power system FMECA and adjusted the severity according to the applicable aircraft's FHA and was then captured in the applicable aircraft's FMECA. The modular drive and power system allowed for similar parts to be used for each functional sub-system, which allowed for easy transition from one aircraft architecture to the next.

Table C1: Rotor Gearbox Criticality Summary

Rotor Gearbox System	Criticality by Severity Classification			
	I	II	III	IV
Rotor Shaft System	0	2.01E-11	(a)	(a)
Stage 3 - Planetary System	2.58E-11	1.58E-11	(a)	(a)
Stage 2 - Planetary System	1.93E-11	4.79E-12	(a)	(a)
Spiral Bevel Gear Mesh	6.44E-12	3.83E-12	(a)	(a)
Housing	(b)	(b)	(a)	(a)
Lubrication	(c)	(c)	(a)	(a)
Accessories	(c)	(c)	(a)	(a)
Total	**5.15E-11**	**4.45E-11**	**(a)**	**(a)**

Table C2: Rotor Gearbox System Criticality Summary

System	Failure Mode	Part Count	Criticality by Severity Classification I	II	III	IV
Rotor Shaft	Shaft Failure	1	0	2.54E-17	(a)	(a)
	Bearing Failure	2	0	1.92E-12	(a)	(a)
	Spline Failure	2	0	1.82E-11	(a)	(a)
Stage 3 Planetary	Gear Tooth Failure	8	2.58E-11	2.32E-16	(a)	(a)
	Shaft Failure	8	0	1.02E-16	(a)	(a)
	Bearing Failure	7	0	6.71E-12	(a)	(a)
	Spline Failure	1	0	9.10E-12	(a)	(a)
Stage 2 Planetary	Gear Tooth Failure	6	1.93E-11	1.74E-16	(a)	(a)
	Shaft Failure	5	0	6.35E-17	(a)	(a)
	Bearing Failure	5	0	4.79E-12	(a)	(a)
Spiral Bevel Gear	Gear Tooth Failure	2	6.44E-12	5.80E-17	(a)	(a)
	Shaft Failure	1	0	1.27E-17	(a)	(a)
	Bearing Failure	4	0	3.83E-12	(a)	(a)
	Spline Failure	1	0	9.10E-18	(a)	(a)

Table C3: Motor Gearbox Criticality Summary

Motor Gearbox System	**Criticality by Severity Classification**			
	I	**II**	**III**	**IV**
Stage 1 - Spur Gear System	6.44E-12	3.83E-12	(a)	(a)
Housing	(b)	(b)	(a)	(a)
Lubrication	(c)	(c)	(a)	(a)
Accessories	(c)	(c)	(a)	(a)
Total	**6.44E-12**	**3.83E-12**	**(a)**	**(a)**

Table C4: Motor Gearbox System Criticality Summary

System	**Failure Mode**	**Part Count**	**Criticality by Severity Classification**			
			I	**II**	**III**	**IV**
Stage 1 Spur Gear	Gear Tooth Failure	2	6.44E-12	5.80E-17	(a)	(a)
	Shaft Failure	3	0	3.81E-17	(a)	(a)
	Bearing Failure	4	0	3.83E-12	(a)	(a)
	Spline Failure	2	0	1.82E-17	(a)	(a)

Table C5: PMSM Criticality Summary

PMSM System	**Criticality by Severity Classification**			
	I	**II**	**III**	**IV**
Motor Base Failure Rate	3.50E-06	3.50E-06	3.50E-06	3.50E-06
Armature Shaft System	0	0	1.92E-09	(a)
Winding System	2.94E-07	0.00E+00	2.94E-13	(a)
Housing System	1.00E-09	1.00E-09	1.00E-09	1.00E-09
Lubrication	(c)	(c)	(a)	(a)
Accessories	(c)	(c)	(a)	(a)
Total	**3.80E-06**	**3.50E-06**	**3.50E-06**	**3.50E-06**

Table C6: PMSM System Criticality Summary

System	**Failure Mode**	**Part Count**	**Criticality by Severity Classification**			
			I	**II**	**III**	**IV**
Armature Shaft	Shaft Failure	1	0	0	2.54E-20	(a)
	Bearing Failure	2	0	0	1.92E-09	(a)
Winding	Short Winding	12	2.94E-07	0	(a)	(a)
	Winding Overheating	12	0	0	2.94E-13	(a)

Table C7: Mix Box Criticality Summary

Mix Box System	**Criticality by Severity Classification**			
	I	**II**	**III**	**IV**
Spiral Bevel Gear System	9.66E-12	8.62E-09	(a)	(a)
Housing	(b)	(b)	(a)	(a)
Lubrication	(c)	(c)	(a)	(a)
Accessories	(c)	(c)	(a)	(a)
Total	**9.66E-12**	**8.62E-09**	**(a)**	**(a)**

Table C8: Mix Box System Criticality Summary

System	**Failure Mode**	**Part Count**	**Criticality by Severity Classification**			
			I	**II**	**III**	**IV**
Spiral Bevel Gear	Gear Tooth Failure	3	9.66E-12	8.69E-14	(a)	(a)
	Shaft Failure	3	0	7.62E-17	(a)	(a)
	Bearing Failure	9	0	8.62E-09	(a)	(a)

Table C9: Interconnecting Shafts Criticality Summary

Interconnecting Shaft System	**Criticality by Severity Classification**			
	I	**II**	**III**	**IV**
Drive Shaft System	0	4.66E-17	(a)	(a)
Accessories	(c)	(c)	(a)	(a)
Total	**0**	**4.66E-17**	**(a)**	**(a)**

Table C10: Interconnecting Shaft System Criticality Summary

System	**Failure Mode**	**Part Count**	**Criticality by Severity Classification**			
			I	**II**	**III**	**IV**
Drive Shaft System	Shaft Failure	5	0	4.66E-17	(a)	(a)

Notes:

a) Drive and Power System FMECA focused on Category I and II severity failure modes. See aircraft FMECA for more information on Category III and IV severity failure modes.

b) Stationary structures were not captured in FMECA, unless specifically recommended in supporting literature.

c) Lubrication and sensor reliability was evaluated as part of the failure effect probability in drive and power system FMECA and in the thermal management system reliability analysis.

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Rotor Shaft System	Transfer Stage 3 Output Torque to Rotor	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01	1.00E-09	Chip Detector, Elec. Screen, VHMS	4.79E-13
Rotor Shaft System	Transfer Rotor Loads to Static Structure	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Rolling element bearings utilized for slow, progressive failure that has probability of being detection.	II	5.00E-01	1.00E-09	Chip Detector, Elec. Screen, VHMS	4.79E-13
Rotor Shaft System	Transfer Stage 3 Output Torque to Rotor	9.10E-09	Spline Failure	Wear Initiated by Shaft Windup and Edge Loading.	All	Tooth Surface Damage.	Metallic debris generated, redistribution of load along spline, increased vibration.	Edge loading is alleviated. Failure is detected.	See Beta Mode Ratio Explanation.	Two rows of splines and fixed spline clamp up means that torque can still be transmitted with single spline failure.	II	5.00E-01	1.00E-03	VHMS	4.55E-12
Rotor Shaft System	Transfer Rotor Loads to Static Structure	9.10E-09	Spline Failure	Wear Initiated by Shaft Windup and Edge Loading.	All	Tooth Surface Damage.	Metallic debris generated, redistribution of load along spline, increased vibration.	Edge loading is alleviated. Failure is detected. Friction from fixed spline takes some torque load.	See Beta Mode Ratio Explanation.	Two rows of splines and fixed spline clamp up means that torque can still be transmitted with single spline failure.	II	5.00E-01	1.00E-03	VHMS	4.55E-12
Rotor Shaft System	Transfer Stage 3 Output Torque to Rotor	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Edge loading is alleviated. Failure is detected. Friction from fixed spline takes some torque load.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	2.50E-01	1.00E-03	VHMS	6.35E-18
Rotor Shaft System	Transfer Rotor Loads to Static Structure	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	2.50E-01	1.00E-03	VHMS	6.35E-18
Rotor Shaft System	Transfer Stage 3 Output Torque to Rotor	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	2.50E-01	1.00E-03	VHMS	6.35E-18
Rotor Shaft System	Transfer Rotor Loads to Static Structure	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	2.50E-01	1.00E-03	VHMS	6.35E-18

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	3.22E-08	Gear Tooth Failure	Crack Initiated by Gear Tooth Bending Fatigue	All	Decreased backlash, coast flank contact, potential topland interference.	Sudden increase in torque at mesh point, high loads imparted to stationary structure. Potential structural damage.	Inability to transmit torque from a single motor to rotor systems	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	I	1.00E-01	1.00E-03	VHMS	3.22E-12
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	3.22E-08	Gear Tooth Failure	Pitting Initiated by Cyclic Contact Stress Thru Lubrication Film	All	Tooth surface damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	2.90E-17
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.58E-13
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	1.27E-23
Stage 3 planetary system	Transfer Stage 2 Output Torque to Rotor Shaft	9.10E-09	Spline Failure	Wear Initiated by Shaft Windup, Edge Loading, or Micromotion.	All	Tooth Surface Damage.	Metallic debris generated, redistribution of load along spline, increased vibration.	Edge loading is alleviated. Failure is detected. Friction from fixed spline takes some torque load.	See Beta Mode Ratio Explanation.	Fixed spline clamp up means that torque can still be transmitted if wear initiates.	II	1.00E+00	1.00E-03	VHMS	9.10E-12
Stage 2 planetary system	Transfer Sun Gear Shaft Torque to Stage 3 Planetary System	3.22E-08	Gear Tooth Failure	Crack Initiated by Gear Tooth Bending Fatigue	All	Decreased backlash, coast flank contact, potential topland interference.	Sudden increase in torque at mesh point, high loads imparted to stationary structure. Potential structural damage.	Inability to transmit torque from a single motor to rotor systems	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	I	1.00E-01	1.00E-03	VHMS	3.22E-12

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Stage 2 planetary system	Transfer Sun Gear Shaft Torque to Stage 3 Planetary System	3.22E-08	Gear Tooth Failure	Pitting Initiated by Cyclic Contact Stress Thru Lubrication Film	All	Tooth surface damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	2.90E-17
Stage 2 planetary system	Transfer Sun Gear Shaft Torque to Stage 3 Planetary System	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.58E-13
Stage 2 planetary system	Transfer Sun Gear Shaft Torque to Stage 3 Planetary System	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Stage 2 planetary system	Transfer Sun Gear Shaft Torque to Stage 3 Planetary System	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	1.27E-23
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	3.22E-08	Gear Tooth Failure	Crack Initiated by Gear Tooth Bending Fatigue	All	Decreased backlash, coast flank contact, potential topland interference.	Sudden increase in torque at mesh point, high loads imparted to stationary structure. Potential structural damage.	Inability to transmit torque from a single motor to rotor systems	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	I	1.00E-01	1.00E-03	VHMS	3.22E-12
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	3.22E-08	Gear Tooth Failure	Pitting Initiated by Cyclic Contact Stress Thru Lubrication Film	All	Tooth surface damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	2.90E-17
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.58E-13
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	9.10E-09	Spline Failure	Wear Initiated by Lubrication Breakdown.	All	Tooth Surface Damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Fixed Spline has Clamp-Up from Lock Nut to Continue to Transmit Load.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.10E-18

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Spiral Bevel Gear System – Rotor Gearbox	Transfer Torque to and from Drive Shaft Systems	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	1.27E-23
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	3.22E-08	Gear Tooth Failure	Crack Initiated by Gear Tooth Bending Fatigue	All	Decreased backlash, coast flank contact, potential topland interference.	Sudden increase in torque at mesh point, high loads imparted to stationary structure. Potential structural damage.	Inability to transmit torque from a single motor to rotor systems	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	I	1.00E-01	1.00E-03	VHMS	3.22E-12
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	3.22E-08	Gear Tooth Failure	Pitting Initiated by Cyclic Contact Stress Thru Lubrication Film	All	Tooth surface damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	2.90E-17
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.58E-13
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	9.10E-09	Spline Failure	Wear Initiated by Inadequate Lubrication Film.	All	Tooth Surface Damage.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Fixed Spline has Clamp-Up from Lock Nut to Continue to Transmit Load.	II	1.00E+00	1.00E-09	Chip Detector Elec. Screen VHMS	9.10E-18
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Stage 1 Spur Gear System	Transfer motor output torque to overrunning clutch	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-09	Chip Detector Elec. Screen VHMS	1.27E-23
Armature Shaft System	Transfer electromotive force to drive system	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Chip migration to air gap of armature/stator, increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	VHMS and torque ripple monitoring. Overrunning clutch allows motor to be shut down upon detection.	III	5.00E-01	1.00E-06	VHMS Torque Ripple	4.79E-10

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Armature Shaft System	Transfer electromotive force to drive system	9.58E-04	Bearing Failure	Arc Burns Initiated by Presence of Electric Currents.	All	Fatigue Damage of Ball or Raceway.	Metallic debris generated, increased heat generation, shaft misalignment.	Chip migration to air gap of armature/stator, increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	VHMS and torque ripple monitoring. Overrunning clutch allows motor to be shut down upon detection. Grounding rings to alleviate electric potential.	III	5.00E-01	1.00E-06	VHMS Torque Ripple	4.79E-10
Armature Shaft System	Transfer electromotive force to drive system	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Armature misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	VHMS and torque ripple monitoring. Overrunning clutch allows motor to be shut down upon detection.	III	5.00E-01	1.00E-06	VHMS Torque Ripple	1.27E-20
Armature Shaft System	Transfer electromotive force to drive system	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Gear head misalignment, metallic debris generated.	Chip migration to air gap of armature/stator, increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	VHMS and torque ripple monitoring. Overrunning clutch allows motor to be shut down upon detection.	III	5.00E-01	1.00E-06	VHMS Torque Ripple	1.27E-20
Winding System	Generate electromotive force	2.45E-05	Short Winding	Insulation degradation	All	Insulation breakdown and loss of insulation quality.	Short circuit and uncontrolled torque application.	Inadvertent torque transmission	See Beta Mode Ratio Explanation.	Oil cooled system insulates field windings. Oil condition monitoring system monitors insulation quality of the oil.	I	1.00E+00	1.00E-03	Oil Condition Monitoring System	2.45E-08
Winding System	Generate electromotive force	2.45E-05	Winding Overheating	Incorrect supply voltage or voltage imbalance	All	High temperatures in field wire.	Damage to field wire insulation, increased resistance. Reduced efficiency. Failure is detected.	Inability to transmit torque from a single motor to rotor systems	See Beta Mode Ratio Explanation.	Voltage monitoring, torque ripple monitoring, resistance monitoring. Overrunning clutch allows motor to be shut down upon detection.	III	1.00E+00	1.00E-09	Voltage Monitoring Torque Ripple Resistance Monitoring	2.45E-14
Spiral Bevel Gear System - Mix Box	Transfer Torque to and from Drive Shaft Systems	3.22E-08	Gear Tooth Failure	Crack Initiated by Gear Tooth Bending Fatigue	All	Tooth Failure.	Sudden increase in torque at mesh point, high loads imparted to stationary structure. Potential structural damage.	Failure to Transfer Torque to Shaft System	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	I	1.00E-01	1.00E-03	VHMS	3.22E-12
Spiral Bevel Gear System - Mix Box	Transfer Torque to and from Drive Shaft Systems	3.22E-08	Gear Tooth Failure	Pitting Initiated by Cyclic Contact Stress Thru Lubrication Film	All	Tooth Surface Damage and Eventual Tooth Failure.	Metallic debris generated, increased heat generation.	Increased vibrations, decreased efficiency. Failure is detected.	See Beta Mode Ratio Explanation.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.00E-01	1.00E-06	Chip Detector Elec. Screen VHMS	2.90E-14
Spiral Bevel Gear System - Mix Box	Transfer Torque to and from Drive Shaft Systems	9.58E-04	Bearing Failure	Spalling Initiated by Cyclic Contact Stress.	All	Metallic debris generated, increased heat generation, shaft misalignment.	Increased vibrations, decreased efficiency. Failure is detected.	Failure to Transfer Torque to Shaft System	Multiple detection methods reduce severity.	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00	1.00E-06	Chip Detector Elec. Screen VHMS	9.58E-10

Table C11: Drive and Power System FMECA

Component Nomenclature	Function	Failure Rate (λ)	Failure Mode	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
Spiral Bevel Gear System - Mix Box	Transfer Torque to and from Drive Shaft Systems	2.54E-14	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Gear head misalignment, potential edge loading of roller bearing.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	Gear Tooth Surface Damage and Eventual Tooth Failure, Bearing Fatigue Damage of Ball or Raceway, or Eventual Shaft Failure.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Spiral Bevel Gear System - Mix Box	Transfer Torque to and from Drive Shaft Systems	2.54E-14	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Gear head misalignment, potential edge loading of roller bearing, metallic debris generated.	Gear Tooth Surface Damage, Bearing Fatigue Damage of Ball or Raceway. Failure is Detected.	Gear Tooth Surface Damage and Eventual Tooth Failure, Bearing Fatigue Damage of Ball or Raceway, or Eventual Shaft Failure.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected.	II	5.00E-01	1.00E-03	VHMS	1.27E-17
Drive Shaft System	Transfer torque to and from gearboxes	9.31E-15	Shaft Failure	Crack Initiated by Shaft Bending Fatigue.	All	Reduced shaft stiffness.	Increased vibrations.	Inability to transmit torque from a single motor to rotor systems. Rotors are desynchronized. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	5.00E-01	1.00E-03	VHMS	4.66E-18
Drive Shaft System	Transfer torque to and from gearboxes	9.31E-15	Shaft Failure	Crack Initiated by Fretting Fatigue.	All	Faying surface damage and reduced shaft stiffness.	Increased vibrations.	Inability to transmit torque from a single motor to rotor systems. Rotors are desynchronized. Failure is Detected.	See Beta Mode Ratio Explanation.	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	5.00E-01	1.00E-03	VHMS	4.66E-18

Appendix D Failure Modes, Effects, and Criticality Analysis

Failure modes, effects, and criticality analysis (FMECA) tables for the electric quadrotor (eQuad), hybrid-electric quadrotor (hQuad), turboshaft quadrotor (tQuad), collective controlled, electric hexarotor (eHex), and RPM-controlled eHex are provided in Table D1, Table D2, Table D3, Table D4, Table D5, respectively. FMECA tables for the eQuad without interconnecting shafting and for the RPM-controlled electric octorotor (eOct) were not developed. The FMECA table for the eQuad without interconnecting shafting would have been similar to the eQuad with interconnecting shafting, except that the functional interconnection line items would have been removed. Similarly, the RPM-controlled eOct FMECA table would have been similar to the RPM-controlled eHex FMECA table, except that an additional two groups of motor/rotor functionality would have been applied for Rotor #7 and #8. As such, failure rates for the eQuad without interconnecting shafting were taken from the eQuad FMECA Table, Table D1, and failure rates for the RPM-controlled eOct were taken from the RPM-controlled eHex FMECA Table, Table D5.

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1A1	Provide HVDC power to electric motors	1.00E-06	A	Failure to provide HVDC electrical energy to ESC #1	1	HV Battery output failure or associated wiring. Loss of output to ESC #1 only.	All	No power to ESC #1; Motor #1 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1B1	Provide HVDC power to electric motors	1.00E-06	B	Failure to provide HVDC electrical energy to ESC #2	1	HV Battery output failure or associated wiring. Loss of output to ESC #2 only.	All	No power to ESC #2; Motor #2 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1C1	Provide HVDC power to electric motors	1.00E-06	C	Failure to provide HVDC electrical energy to ESC #3	1	HV Battery output failure or associated wiring. Loss of output to ESC #3 only.	All	No power to ESC #3; Motor #3 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #3 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1D1	Provide HVDC power to electric motors	1.00E-06	D	Failure to provide HVDC electrical energy to ESC #4	1	HV Battery output failure or associated wiring. Loss of output to ESC #4 only.	All	No power to ESC #4; Motor #4 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	HB	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1E1	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	1	Battery cell failure - no runaway.	All	Loss of output from single branch within battery network. Battery output voltage slightly reduced. Increased current draw from remaining battery cells	Output voltage slightly reduced to one or more motors.	Reduced range and/or slight degradation of motor performance.	B	Visual and audible warning provided to pilot.	Battery monitoring system must detect and isolate the fault. Continued operation with failed cell may put additional stress on other battery cells which must be managed to prevent catastrophic failure	IV	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E2	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	2	Battery cell failure - Thermal runaway. - contained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery monitoring system detects failure, disconnects and isolates the defective battery cell.	Reduced battery system capacity, slight degradation of battery output voltage provided to one or more electric motors. Battery may catch fire. Excess heat generated may affect adjacent battery cells.	Reduced range and/or degradation electric motor performance.	B	Visual and audible warning provided to pilot.	Battery cooling and fire protection system must contain battery temperature to prevent loss of aircraft	III	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E3	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	3	Battery cell failure internal short - thermal runaway - uncontained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery catches fire. Loss of all HVDC output.	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes	1.00E-02	It is assumed that in most cases, battery failure will occur gradually giving the pilot time to land safely.	2.50E-09
1E4	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	4	Complete HV battery failure; low voltage or no voltage output. (Battery discharged)	All	Loss of all HVDC output	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes. Battery system is assumed to have redundancy where multiple internal failures	1.00E-02	It is assumed that in most cases, battery voltage would decrease gradually, giving the pilot time to land safely.	2.50E-09

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																must occur for complete loss of HVDC to all 4 motors.			
2A1	Convert HV electrical energy to shaft torque	2.70E-04	A	Failure to provide output torque from Motor #1 to Gearbox #1	1	ESC #1 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2A2	Convert HV electrical energy to shaft torque	2.70E-04	A	Failure to provide output torque from Motor #1 to Gearbox #1	2	ESC #1 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2A3	Convert HV electrical energy to shaft torque	3.50E-06	A	Internal failure of Motor #1	3	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of Engine #1	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2A4	Convert HV electrical energy to shaft torque	3.50E-06	A	Internal failure of Motor #1	4	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of motor #1	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.50E-06
2A5	Convert HV electrical energy to shaft torque	3.80E-06	A	Internal failure of Motor #1	5	Internal failure of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2A6	Convert torque from motor to rot	4.20E-07	A	Failure to provide output torque from Motor #1 to Gearbox #1	6	Clutch #1 failure - failure to engage	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2A7	Convert HV electrical energy to shaft torque	4.20E-07	A	Failure to provide output torque from Motor #1 to Gearbox #1	7	Clutch #1 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #1 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2B1	Convert HV electrical energy to shaft torque	2.70E-04	B	Failure to provide output torque from Motor #2 to Gearbox #2	1	ESC #2 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2B2	Convert HV electrical energy to shaft torque	2.70E-04	B	Failure to provide output torque from Motor #2 to Gearbox #2	2	ESC #2 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2B3	Convert HV electrical energy to shaft torque	3.50E-06	B	Internal failure of Motor #2	3	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of Engine #2	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2B4	Convert HV electrical energy to shaft torque	3.50E-06	B	Internal failure of Motor #2	4	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of motor #2	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.50E-06
2B5	Convert HV electrical energy to shaft torque	3.80E-06	B	Internal failure of Motor #2	5	Internal failure of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2B6	Convert HV electrical energy to shaft torque	4.20E-07	B	Failure to provide output torque from Motor #2 to Gearbox #2	6	Clutch #2 failure - failure to engage	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2B7	Convert HV electrical energy to shaft torque	4.20E-07	B	Failure to provide output torque from Motor #2 to Gearbox #2	7	Clutch #2 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #2 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2C1	Convert HV electrical energy to shaft torque	2.70E-04	C	Failure to provide output torque from Motor #3 to Gearbox #3	1	ESC #3 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and	5.40E-05

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
													energy stored in rotors to provide soft landing (autorotate)					controllability of aircraft with limited power	
2C2	Convert HV electrical energy to shaft torque	2.70E-04	C	Failure to provide output torque from Motor #3 to Gearbox #3	2	ESC #3 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2C3	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #3	3	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of Engine #3	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2C4	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #3	4	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of motor #3	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2C5	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #3	5	Internal failure of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2C6	Convert HV electrical energy to shaft torque	4.20E-07	C	Failure to provide output torque from Motor #3 to Gearbox #3	6	Clutch #3 failure - failure to engage	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #3 through combiner gearbox. Available power reduced	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2C7	Convert HV electrical energy to shaft torque	4.20E-07	C	Failure to provide output torque from Motor #3 to Gearbox #3	7	Clutch #3 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #3 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2D1	Convert HV electrical energy to shaft torque	2.70E-04	D	Failure to provide output torque from Motor #4 to Gearbox #4	1	ESC #4 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2D2	Convert HV electrical energy to shaft torque	2.70E-04	D	Failure to provide output torque from Motor #4 to Gearbox #4	2	ESC #4 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2D3	Convert HV electrical energy to shaft torque	3.50E-06	D	Internal failure of Motor #4	3	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of Engine #4	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2D4	Convert HV electrical energy to shaft torque	3.50E-06	D	Internal failure of Motor #4	4	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of motor #4	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2D5	Convert HV electrical energy	3.80E-06	D	Internal failure of Motor #4	5	Internal failure of Motor #4	All	Unable to transfer	No torque from Motor #4. Torque from the	Flying qualified unaffected. Pilot	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	1.00E+00		1.00E+00		3.80E-06

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	to shaft torque							torque gearbox #4.	remaining three motors distributed to rotors by the combiner gearbox.	alerted, land as soon as practical		provided to pilot	failure to pilot and crew.						
2D6	Convert HV electrical energy to shaft torque	4.20E-07	D	Failure to provide output torque from Motor #4 to Gearbox #4	6	Clutch #4 failure - failure to engage	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2D7	Convert HV electrical energy to shaft torque	4.20E-07	D	Failure to provide output torque from Motor #4 to Gearbox #4	7	Clutch #4 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #4 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
3A1	Transfer motor torque to rotors	3.83E-12	A	Internal failure of gearbox system Gearbox #1	1	Internal failure of gearbox #1	All	Increased vibrations, decreased efficiency of rotor gearbox #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3A2	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	2	Gearbox #1 Failure, failure to transfer torque from motor #1 or combiner gearbox to Rotor #1	All	Unable to transfer torque from motor #1	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3B1	Transfer motor torque to rotors	3.83E-12	B	Internal failure of gearbox system Gearbox #2	1	Internal failure of gearbox #2	All	Increased vibrations, de-	Excess drag on gearbox #2 consumes power	Failure is detected, landed as soon as practical	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	1.00E+00		1.00E+00		3.83E-12

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								creased efficiency of rotor gearbox #2	from remaining 3 motors.			provided to pilot	failure to pilot and crew.						
3B2	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	2	Gearbox #2 Failure, failure to transfer torque from motor #2 or combiner gearbox to Rotor #2	All	Unable to transfer torque from motor #2	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3C1	Transfer motor torque to rotors	3.83E-12	C	Internal failure of gearbox system Gearbox #3	1	Internal failure of gearbox #3	All	Increased vibrations, decreased efficiency of rotor gearbox #3	Excess drag on gearbox #3 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3C2	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	2	Gearbox #3 Failure, failure to transfer torque from motor #3 or combiner gearbox to Rotor #3	All	Unable to transfer torque from motor #3	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3D1	Transfer motor torque to rotors	3.83E-12	D	Internal failure of gearbox system Gearbox #4	1	Internal failure of gearbox #4	All	Increased vibrations, decreased efficiency of rotor gearbox #4	Excess drag on gearbox #4 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3D2	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	2	Gearbox #4 Failure, failure to transfer torque from motor #4 or combiner gearbox to Rotor #4	All	Unable to transfer torque from motor #4	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3E1	Transfer torque between drive gearboxes	8.62E-09	E	Internal Failure of combiner gearbox #1	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents	II	1.00E+00		1.00E+00		8.62E-09

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
													damage to neighboring systems.						
3E2	Transfer torque between drive gearboxes	9.66E-12	E	Internal Failure of combiner gearbox #1	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of interconnecting drive shafts.	Potential damage to interconnecting drive shafts and rotor gearboxes.	Worst case pluasible outcome: Seized gearbox causes cascading failures of the drive system. Loss of power to rotor, loss of aircraft and crew		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		9.66E-12
3F1	Transfer torque between drive gearboxes	8.62E-09	F	Internal Failure of combiner gearbox #2	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	1.00E+00		1.00E+00		8.62E-09
3F2	Transfer torque between drive gearboxes	9.66E-12	F	Internal Failure of combiner gearbox #2	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of interconnecting drive shafts.	Potential damage to interconnecting drive shafts and rotor gearboxes.	Seized gearbox causes cascading failures of the drive system. Loss of power to rotor, loss of aircraft and crew		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		9.66E-12
3G1	Transfer torque between drive gearboxes	4.66E-17	G	Failure of combiner gearbox interconnecting drive shaft	1	Interconnecting driveshaft shear	All	Loss of ability to transfer torque between drive gearbox and combiner gearbox	Loss of ability to cross-shaft power in the event of a motor 1 failure	Rotors are desynchronized.; In the event of motor 1 torque output - Loss of power to rotors 1. Loss of control of aircraft. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	1.00E+00		1.00E+00		4.66E-17
4A1	Transfer torque from gearbox to rotor	4.45E-11	A	Internal failure of rotor #1 Transmission	1	Internal failure of rotor #1 Transmission	All	Increased vibrations, decreased efficiency of rotor #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4A2	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	2	Internal failure of rotor #1 Transmission	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4B1	Transfer torque from gearbox to rotor	4.45E-11	B	Internal failure of rotor #2 Transmission	1	Internal failure of rotor #2 Transmission	All	Increased vibrations, decreased efficiency of rotor #2	Excess drag on gearbox #2 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4B2	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	2	Internal failure of rotor #2 Transmission	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
4C1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #3 Transmission	1	Internal failure of rotor #3 Transmission	All	Increased vibrations, decreased efficiency of rotor #3	Excess drag on gearbox #3 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4C2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	2	Internal failure of rotor #3 Transmission	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
4D1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #4 Transmission	1	Internal failure of rotor #4 Transmission	All	Increased vibrations, decreased efficiency of rotor #4	Excess drag on gearbox #4 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4D2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #4 Transmission	2	Internal failure of rotor #4 Transmission	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
5A1	Provide hydraulic control of Rotor pitch	4.46E-10	A	Failure to transfer commanded input to Rotor #1 pitch	1	Failure of triplex actuator #1	All	Loss of ability for control system to command triplex Actuator #1 position	Loss of ability for control system to command a derived pitch input to Rotor #1	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #1 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10

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5B1	Provide hydraulic control of Rotor pitch	4.46E-10	B	Failure to transfer commanded input to Rotor #2 pitch	1	Failure of triplex actuator #2	All	Loss of ability for control system to command triplex Actuator #2 position	Loss of ability for control system to command a derived pitch input to Rotor #2	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #2 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5C1	Provide hydraulic control of Rotor pitch	4.46E-10	C	Failure to transfer commanded input to Rotor #3 pitch	1	Failure of triplex actuator #3	All	Loss of ability for control system to command triplex Actuator #3 position	Loss of ability for control system to command a derived pitch input to Rotor #3	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #3 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5D1	Provide hydraulic control of Rotor pitch	4.46E-10	D	Failure to transfer commanded input to Rotor #4 pitch	1	Failure of triplex actuator #4	All	Loss of ability for control system to command triplex Actuator #4 position	Loss of ability for control system to command a derived pitch input to Rotor #4	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #4 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5E1	Provide hydraulic control of Rotor pitch	4.98E-07	E	Failure of hydraulic pump 1	1	Failure of hydraulic pump #1 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5E2	Provide hydraulic control of	4.98E-07	E	Failure of hydraulic pump 1	2	Failure of hydraulic pump #1 resulting	All	Internal failure of pump	Loss of Flow through Pump #1 compensated by	None	B	Visual and audible alert	Flight control system compensates for lack	II	8.50E-02	FM distribution based on	1.00E+00	Beta will be highly dependent	4.24E-08

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	Rotor pitch					in loss of sufficient flow		results in severely degraded output/ no output	two remaining pumps			provided to pilot	of flow with two remaining pumps. System designed such that control can be maintained with single pump			FDM -91 pump hydraulic		upon detailed design and controllability of aircraft with limited power	
5F1	Provide hydraulic control of Rotor pitch	4.98E-07	F	Failure of hydraulic pump 2	1	Failure of hydraulic pump #2 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5F2	Provide hydraulic control of Rotor pitch	4.98E-07	F	Failure of hydraulic pump 2	2	Failure of hydraulic pump #2 resulting in loss of sufficient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow through Pump #2 compensated by two remaining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two remaining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5G1	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 3	1	Failure of hydraulic pump #3 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5G2	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 3	2	Failure of hydraulic pump #3 resulting in loss of sufficient flow	All	Internal failure of pump results in severely de-	Loss of Flow through Pump #3 compensated by two remaining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two remaining pumps. System designed such that control can be	II	8.50E-02	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and	4.24E-08

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								graded output/ no output					maintained with single pump					controllability of aircraft with limited power	
6A1	Transfer heat generated during operation of motor gearbox	3.62E-07	A	TMS #1 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6B1	Transfer heat generated during operation of motor gearbox	2.73E-07	B	Degraded oil flow through TMS #1	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6C1	Transfer heat generated during operation of motor gearbox	1.49E-05	C	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6D1	Transfer heat generated during operation of motor gearbox	2.73E-07	D	Loss of excess oil return path preventing flow to motor #1	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6E1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #1/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6F1	Transfer heat generated during operation of	3.62E-07	F	TMS #2 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	motor gearbox																		
6G1	Transfer heat generated during operation of motor gearbox	2.73E-07	G	Degraded oil flow through TMS #2	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6H1	Transfer heat generated during operation of motor gearbox	1.49E-05	H	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6!1	Transfer heat generated during operation of motor gearbox	2.73E-07	!	Loss of excess oil return path preventing flow to motor #2	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6J1	Transfer heat generated during operation of motor gearbox	2.73E-07	J	Severely Degraded Flow of oil through TMS #2/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6K1	Transfer heat generated during operation of motor gearbox	3.62E-07	K	TMS #3 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6L1	Transfer heat generated during operation of motor gearbox	2.73E-07	L	Degraded oil flow through TMS #3	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6M1	Transfer heat generated during operation of motor gearbox	1.49E-05	M	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6N1	Transfer heat generated during operation of motor gearbox	2.73E-07	N	Loss of excess oil return path preventing flow to motor #3	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6O1	Transfer heat generated during operation of motor gearbox	2.73E-07	O	Severely Degraded Flow of oil through TMS #3/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6P1	Transfer heat generated during operation of motor gearbox	3.62E-07	P	TMS #4 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6Q1	Transfer heat generated during operation of motor gearbox	2.73E-07	Q	Degraded oil flow through TMS #4	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6R1	Transfer heat generated during operation of motor gearbox	1.49E-05	R	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6S1	Transfer heat generated during operation of	2.73E-07	D	Loss of excess oil return path preventing	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07

Table D1: eQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	motor gearbox			flow to motor #4															
6T1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #4/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1A1	Provide HVDC power to electric motors	1.00E-06	A	Failure to provide HVDC electrical energy to ESC #1	1	HV Battery output failure or associated wiring. Loss of output to ESC #1 only.	All	No power to ESC #1; Motor #1 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1B1	Provide HVDC power to electric motors	1.00E-06	B	Failure to provide HVDC electrical energy to ESC #2	1	HV Battery output failure or associated wiring. Loss of output to ESC #2 only.	All	No power to ESC #2; Motor #2 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1C1	Provide HVDC power to electric motors	1.00E-06	C	Failure to provide HVDC electrical energy to ESC #3	1	HV Battery output failure or associated wiring. Loss of output to ESC #3 only.	All	No power to ESC #3; Motor #3 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #3 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1D1	Provide HVDC power to electric motors	1.00E-06	D	Failure to provide HVDC electrical energy to ESC #4	1	HV Battery output failure or associated wiring. Loss of output to ESC #4 only.	All	No power to ESC #4; Motor #4 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	HB	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1E1	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	1	Battery cell failure - no runaway.	All	Loss of output from single branch within battery network. Battery output voltage slightly reduced. Increased current draw from remaining battery cells	Output voltage slightly reduced to one or more motors.	Reduced range and/or slight degradation of motor performance.	B	Visual and audible warning provided to pilot.	Battery monitoring system must detect and isolate the fault. Continued operation with failed cell may put additional stress on other battery cells which must be managed to prevent catastrophic failure	IV	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E2	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	2	Battery cell failure - Thermal runaway. - contained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery monitoring system detects failure, disconnects and isolates the defective battery cell.	Reduced battery system capacity, slight degradation of battery output voltage provided to one or more electric motors. Battery may catch fire. Excess heat generated may affect adjacent battery cells.	Reduced range and/or degradation electric motor performance.	B	Visual and audible warning provided to pilot.	Battery cooling and fire protection system must contain battery temperature to prevent loss of aircraft	III	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E3	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	3	Battery cell failure internal short - thermal runaway - uncontained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery catches fire. Loss of all HVDC output.	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes	1.00E-02	It is assumed that in most cases, battery failure will occur gradually giving the pilot time to land safely.	2.50E-09
1E4	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	4	Complete HV battery failure; low voltage or no voltage output. (Battery discharged)	All	Loss of all HVDC output	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes. Battery system is assumed to have redundancy where multiple internal failures must occur	1.00E-02	It is assumed that in most cases, battery voltage would decrease gradually, giving the pilot time to land safely.	2.50E-09

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																for complete loss of HVDC to all 4 motors.			
1F1	Provide HVDC for propulsion and to charge batteries	2.67E-06	F	Failure to provide shaft power to gerbox	1	Turbo-shaft engine failure	All	Loss of HVDC electrical power to drive lift and thrust motors and to charge batteries	Electric motors operate from battery power only. Increased power demand from batteries. Batteries rapidly discharge.	Loss of propulsion if pilot cannot land safely before batteries are discharged. Loss of aircraft.	B	Visual and audible alert provided to pilot	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries discharge 40% of the time.	1.60E-06
1G1	Provide HVDC for propulsion and to charge batteries	5.00E-06	G	Failure to provide shaft power to AC generator	1	Gearbox failure	All	Loss of HVDC electrical power to drive lift and thrust motors and to charge batteries	Electric motors operate from battery power only. Increased power demand from batteries. Batteries rapidly discharge.	Loss of propulsion if pilot cannot land safely before batteries are discharged. Loss of aircraft.	B	Visual and audible warning provided to pilot.	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries discharge 40% of the time.	3.00E-06
1H1	Provide HVDC for propulsion and to charge batteries	9.24E-05	H	Failure to convert shaft power to AC electrical power	1	AC Generator failure	All	Loss of HVDC electrical power to drive lift and thrust motors and to charge batteries	Electric motors operate from battery power only. Increased power demand from batteries. Batteries rapidly discharge.	Loss of propulsion if pilot cannot land safely before batteries are discharged. Loss of aircraft.	B	Visual and audible warning provided to pilot.	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries	5.54E-05

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		discharge 40% of the time.	
1I1	Provide HVDC for propulsion and to charge batteries	2.00E-04	I	Failure to convert AC electrical power to HVDC electrical power	1	AC/DC converter failure	All	Loss of HVDC electrical power to drive lift and thrust motors and to charge batteries	Electric motors operate from battery power only. Increased power demand from batteries. Batteries rapidly discharge.	Loss of propulsion if pilot cannot land safely before batteries are discharged. Loss of aircraft.	B	Visual and audible alert provided to pilot	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries discharge 40% of the time.	1.20E-04
2A1	Convert HV electrical energy to shaft torque	2.70E-04	A	Failure to provide output torque from Motor #1 to Gearbox #1	1	ESC #1 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2A2	Convert HV electrical energy to shaft torque	2.70E-04	A	Failure to provide output torque from Motor #1 to Gearbox #1	2	ESC #1 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2A3	Convert HV electrical energy to shaft torque	3.50E-06	A	Internal failure of Motor #1	3	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of Engine #1	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	3.30E-01		1.00E+00		1.16E-06

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2A4	Convert HV electrical energy to shaft torque	3.50E-06	A	Internal failure of Motor #1	4	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of motor #1	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	3.30E-01		1.00E-02		1.16E-08
2A5	Convert HV electrical energy to shaft torque	3.80E-06	A	Internal failure of Motor #1	5	Internal failure of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2A6	Convert torque from motor to rot	4.20E-07	A	Failure to provide output torque from Motor #1 to Gearbox #1	6	Clutch #1 failure - failure to engage	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2A7	Convert HV electrical energy to shaft torque	4.20E-07	A	Failure to provide output torque from Motor #1 to Gearbox #1	7	Clutch #1 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #1 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2B1	Convert HV electrical energy to shaft torque	2.70E-04	B	Failure to provide output torque from Motor #2 to Gearbox #2	1	ESC #2 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2B2	Convert HV electrical energy to shaft torque	2.70E-04	B	Failure to provide output torque from Motor #2 to Gearbox #2	2	ESC #2 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2B3	Convert HV electrical energy to shaft torque	3.50E-06	B	Internal failure of Motor #2	3	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of Engine #2	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2B4	Convert HV electrical energy to shaft torque	3.50E-06	B	Internal failure of Motor #2	4	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of motor #2	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	3.30E-01		1.00E-02		1.16E-08
2B5	Convert HV electrical energy to shaft torque	3.80E-06	B	Internal failure of Motor #2	5	Internal failure of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2B6	Convert HV electrical energy to shaft torque	4.20E-07	B	Failure to provide output torque from Motor #2 to Gearbox #2	6	Clutch #2 failure - failure to engage	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air craft/hard landing possible	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2B7	Convert HV electrical energy to shaft torque	4.20E-07	B	Failure to provide output torque from Motor #2 to Gearbox #2	7	Clutch #2 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque	Torque from other 3 motors forces motor #2 to continue to spin. In case of motor-stator contact, friction causes excessive	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of	9.70E-12

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								from other motors.	heat to be generated, causing motor to catch fire.									motor failure rate.	
2C1	Convert HV electrical energy to shaft torque	2.70E-04	C	Failure to provide output torque from Motor #3 to Gearbox #3	1	ESC #3 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2C2	Convert HV electrical energy to shaft torque	2.70E-04	C	Failure to provide output torque from Motor #3 to Gearbox #3	2	ESC #3 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2C3	Convert HV electrical energy to shaft torque	3.12E-06	C	Internal failure of Motor #3	3	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of Engine #3	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.12E-06
2C4	Convert HV electrical energy to shaft torque	1.00E-09	C	Internal failure of Motor #3	4	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of motor #3	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	3.30E-01		1.00E-02		3.30E-12
2C5	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #3	5	Internal failure of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2C6	Convert HV electrical energy to	4.20E-07	C	Failure to provide output torque from Motor	6	Clutch #3 failure - failure to engage	All	No output from ESC #3 to Motor #3. Motor #3 fails	Torque from other 3 motors is transferred to gearbox #3	Limited flight envelope. Reduced maximum speed and insufficient	B	Visual and audible warn-	Pilot can avoid flight conditions which require maximum torque. Pilot can use	II	5.00E-01	Assume sprag clutch failure modes	2.00E-01	Beta will be highly dependent	4.20E-08

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	shaft torque			#3 to Gearbox #3				to provide output torque.	through combiner gearbox. Available power reduced	power to take off or hover at max weight. Loss of air craft/hard landing possible		ing provided to pilot.	energy stored in rotors to provide soft landing (autorotate)			evenly distributed.		upon detailed design and controllability of aircraft with limited power	
2C7	Convert HV electrical energy to shaft torque	4.20E-07	C	Failure to provide output torque from Motor #3 to Gearbox #3	7	Clutch #3 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #3 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2D1	Convert HV electrical energy to shaft torque	2.70E-04	D	Failure to provide output torque from Motor #4 to Gearbox #4	1	ESC #4 A failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2D2	Convert HV electrical energy to shaft torque	2.70E-04	D	Failure to provide output torque from Motor #4 to Gearbox #4	2	ESC #4 B failure	All	No output from ESC to motor.	None	Loss of single drive mottor possible with second ESC failure.	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	1.00E+00		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.40E-05
2D3	Convert HV electrical energy to shaft torque	3.12E-06	D	Internal failure of Motor #4	3	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of Engine #4	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.12E-06

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2D4	Convert HV electrical energy to shaft torque	1.00E-09	D	Internal failure of Motor #4	4	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of motor #4	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	3.30E-01		1.00E-02		3.30E-12
2D5	Convert HV electrical energy to shaft torque	3.80E-06	D	Internal failure of Motor #4	5	Internal failure of Motor #4	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Torque from the remaining three motors distributed to rotors by the combiner gearbox.	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.80E-06
2D6	Convert HV electrical energy to shaft torque	4.20E-07	D	Failure to provide output torque from Motor #4 to Gearbox #4	6	Clutch #4 failure - failure to engage	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air craft/hard landing possible	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08
2D7	Convert HV electrical energy to shaft torque	4.20E-07	D	Failure to provide output torque from Motor #4 to Gearbox #4	7	Clutch #4 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Torque from other 3 motors forces motor #4 to continue to spin. In case of motor-stator contact, friction causes excessive heat to be generated, causing motor to catch fire.	Aircraft fire. Substantial damage to aircraft, Possible loss of aircraft.	N	None	Fire detection and suppression system must contain fire.	I	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
3A1	Transfer motor torque to rotors	3.83E-12	A	Internal failure of gearbox system Gearbox #1	1	Internal failure of gearbox #1	All	Increased vibrations, decreased efficiency of rotor gearbox #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3A2	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	2	Gearbox #1 Failure, failure to transfer torque from motor #1 or com-	All	Unable to transfer torque from motor #1	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
						biner gearbox to Rotor #1				loss of aircraft and occupants									
3B1	Transfer motor torque to rotors	3.83E-12	B	Internal failure of gearbox system Gearbox #2	1	Internal failure of gearbox #2	All	Increased vibrations, decreased efficiency of rotor gearbox #2	Excess drag on gearbox #2 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3B2	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	2	Gearbox #2 Failure, failure to transfer torque from motor #2 or combiner gearbox to Rotor #2	All	Unable to transfer torque from motor #2	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3C1	Transfer motor torque to rotors	3.83E-12	C	Internal failure of gearbox system Gearbox #3	1	Internal failure of gearbox #3	All	Increased vibrations, decreased efficiency of rotor gearbox #3	Excess drag on gearbox #3 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3C2	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	2	Gearbox #3 Failure, failure to transfer torque from motor #3 or combiner gearbox to Rotor #3	All	Unable to transfer torque from motor #3	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3D1	Transfer motor torque to rotors	3.83E-12	D	Internal failure of gearbox system Gearbox #4	1	Internal failure of gearbox #4	All	Increased vibrations, decreased efficiency of rotor gearbox #4	Excess drag on gearbox #4 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3D2	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	2	Gearbox #4 Failure, failure to transfer torque from motor #4 or combiner gearbox to Rotor #4	All	Unable to transfer torque from motor #4	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical. Loss of 2nd gearbox could result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
3E1	Transfer torque between drive gearboxes	8.62E-09	E	Internal Failure of combiner gearbox #1	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail	II	1.00E+00		1.00E+00		8.62E-09

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
													system prevents damage to neighboring systems.						
3E2	Transfer torque between drive gearboxes	9.66E-12	E	Internal Failure of combiner gearbox #1	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of interconnecting drive shafts.	Potential damage to interconnecting drive shafts and rotor gearboxes.	Seized gearbox causes cascading failures of the drive system. Loss of power to rotor, loss of aircraft and crew		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		9.66E-12
3F1	Transfer torque between drive gearboxes	8.62E-09	F	Internal Failure of combiner gearbox #2	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	1.00E+00		1.00E+00		8.62E-09
3F2	Transfer torque between drive gearboxes	9.66E-12	F	Internal Failure of combiner gearbox #2	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of interconnecting drive shafts.	Potential damage to interconnecting drive shafts and rotor gearboxes.	Seized gearbox causes cascading failures of the drive system. Loss of power to rotor, loss of aircraft and crew		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		9.66E-12
3G1	Transfer torque between drive gearboxes	4.66E-17	G	Failure of combiner gearbox interconnecting drive shaft	1	Interconnecting driveshaft shear	All	Loss of ability to transfer torque between drive gearbox and combiner gearbox	Loss of ability to cross-shaft power in the event of a motor 1 failure	Rotors are desynchronized.; In the event of motor 1 torque output - Loss of power to rotors 1. Loss of control of aircraft. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		4.66E-17
4A1	Transfer torque from gearbox to rotor	4.45E-11	A	Internal failure of rotor #1 Transmission	1	Internal failure of rotor #1 Transmission	All	Increased vibrations, decreased efficiency of rotor #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4A2	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	2	Internal failure of rotor #1 Transmission	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4B1	Transfer torque from gearbox to rotor	4.45E-11	B	Internal failure of rotor #2 Transmission	1	Internal failure of rotor #2 Transmission	All	Increased vibrations, decreased efficiency of rotor #2	Excess drag on gearbox #2 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4B2	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	2	Internal failure of rotor #2 Transmission	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
4C1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #3 Transmission	1	Internal failure of rotor #3 Transmission	All	Increased vibrations, decreased efficiency of rotor #3	Excess drag on gearbox #3 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4C2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	2	Internal failure of rotor #3 Transmission	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
4D1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #4 Transmission	1	Internal failure of rotor #4 Transmission	All	Increased vibrations, decreased efficiency of rotor #4	Excess drag on gearbox #4 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4D2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #4 Transmission	2	Internal failure of rotor #4 Transmission	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable. Unable to maintain level flight.	Worst case plausible outcome loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
5A1	Provide hydraulic control of Rotor pitch	4.46E-10	A	Failure to transfer commanded input to Rotor #1 pitch	1	Failure of triplex actuator #1	All	Loss of ability for control system to command triplex Actuator #1 position	Loss of ability for control system to command a derived pitch input to Rotor #1	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #1 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
5B1	Provide hydraulic control of Rotor pitch	4.46E-10	B	Failure to transfer commanded input to Rotor #2 pitch	1	Failure of triplex actuator #2	All	Loss of ability for control system to command triplex Actuator #2 position	Loss of ability for control system to command a derived pitch input to Rotor #2	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #2 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5C1	Provide hydraulic control of Rotor pitch	4.46E-10	C	Failure to transfer commanded input to Rotor #3 pitch	1	Failure of triplex actuator #3	All	Loss of ability for control system to command triplex Actuator #3 position	Loss of ability for control system to command a derived pitch input to Rotor #3	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #3 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5D1	Provide hydraulic control of Rotor pitch	4.46E-10	D	Failure to transfer commanded input to Rotor #4 pitch	1	Failure of triplex actuator #4	All	Loss of ability for control system to command triplex Actuator #4 position	Loss of ability for control system to command a derived pitch input to Rotor #4	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #4 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5E1	Provide hydraulic control of Rotor pitch	4.98E-07	E	Failure of hydraulic pump 1	1	Failure of hydraulic pump #1 resulting in minor performance degredation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5E2	Provide hydraulic control of Rotor pitch	4.98E-07	E	Failure of hydraulic pump 1	2	Failure of hydraulic pump #1 resulting in loss of suffecient flow	All	Internal failure of pump results in severely de-	Loss of Flow trhough Pump #1 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and	4.24E-08

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								graded output/ no output					be maintained with single pump					controllability of aircraft with limited power	
5F1	Provide hydraulic control of Rotor pitch	4.98E-07	F	Failure of hydraulic pump 2	1	Failure of hydraulic pump #2 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5F2	Provide hydraulic control of Rotor pitch	4.98E-07	F	Failure of hydraulic pump 2	2	Failure of hydraulic pump #2 resulting in loss of suffecient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #2 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5G1	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 3	1	Failure of hydraulic pump #3 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5G2	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 3	2	Failure of hydraulic pump #3 resulting in loss of suffecient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #3 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6A1	Transfer heat generated during operation of motor gearbox	3.62E-07	A	TMS #1 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6B1	Transfer heat generated during operation of motor gearbox	2.73E-07	B	Degraded oil flow through TMS #1	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6C1	Transfer heat generated during operation of motor gearbox	1.49E-05	C	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6D1	Transfer heat generated during operation of motor gearbox	2.73E-07	D	Loss of excess oil return path preventing flow to motor #1	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6E1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #1/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6F1	Transfer heat generated during operation of motor gearbox	3.62E-07	F	TMS #2 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6G1	Transfer heat generated during operation of motor gearbox	2.73E-07	G	Degraded oil flow through TMS #2	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6H1	Transfer heat generated during operation of motor gearbox	1.49E-05	H	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6I1	Transfer heat generated during operation of motor gearbox	2.73E-07	I	Loss of excess oil return path preventing flow to motor #2	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6J1	Transfer heat generated during operation of motor gearbox	2.73E-07	J	Severely Degraded Flow of oil through TMS #2/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6K1	Transfer heat generated during operation of motor gearbox	3.62E-07	K	TMS #3 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6L1	Transfer heat generated during operation of motor gearbox	2.73E-07	L	Degraded oil flow through TMS #3	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6M1	Transfer heat generated during operation of motor gearbox	1.49E-05	M	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6N1	Transfer heat generated during operation of motor gearbox	2.73E-07	N	Loss of excess oil return path preventing flow to motor #3	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07

Table D2: hQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6O1	Transfer heat generated during operation of motor gearbox	2.73E-07	O	Severely Degraded Flow of oil through TMS #3/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6P1	Transfer heat generated during operation of motor gearbox	3.62E-07	P	TMS #4 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6Q1	Transfer heat generated during operation of motor gearbox	2.73E-07	Q	Degraded oil flow through TMS #4	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6R1	Transfer heat generated during operation of motor gearbox	1.49E-05	R	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6S1	Transfer heat generated during operation of motor gearbox	2.73E-07	D	Loss of excess oil return path preventing flow to motor #4	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6T1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #4/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1A1	Provide propulsive power	2.67E-06	A	Failure to provide shaft power to gearbox #1	1	Turbo-shaft engine #1 failure	All	Loss of torque output from engine #1	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motor	Loss of second engine or combiner gearbox may result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries discharge 40% of the time.	1.60E-06
1B1	Provide propulsive power	3.83E-12	B	Internal failure of gearbox system Gearbox #1	1	Internal failure of gearbox #1	All	Increased vibrations, decreased efficiency of rotor gearbox #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
1B2	Provide propulsive power	6.44E-12	B	Internal failure of gearbox system Gearbox #1	2	Gearbox #1Failure, failure to transfer torque from motor #1 or combiner gearbox to Rotor #1	All	Unable to transfer torque from motor #1	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
1B3	Provide propulsive power	6.44E-12	B	Internal failure of gearbox system Gearbox #1	3	Gearbox #1 Jam - Failure, failure to transfer torque from motor #1 or combiner gearbox to Rotor #1	All	Unable to transfer torque from motor #1	Continued Engine torque on gearbox causes cascading failures/engine fries	Worst case outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		2.00E-01		1.29E-12
1C1	Provide propulsive power	4.20E-07	C	Failure to provide output torque from Motor #1 to Gearbox #1	1	Clutch #1 failure - failure to engage	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motor	Loss of second engine or combiner gearbox may result in loss of aircraft and occupants	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.20E-08

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1C2	Provide propulsive power	4.20E-07	C	Failure to provide output torque from Motor #1 to Gearbox #1	2	Clutch #1 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motor	Loss of second engine or combiner gearbox may result in loss of aircraft and occupants	N	None	Fire detection and suppression system must contain fire.	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
1D1	Provide propulsive power	2.67E-06	D	Failure to provide shaft power to gearbox	1	Turbo-shaft engine #2 failure	All	Loss of torque output from engine #2	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motor	Loss of second engine or combiner gearbox may result in loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Pilot follows emergency procedures to find safe landing area and land immediately	II	1.00E+00		6.00E-01	Assume that pilot will be able fo find a safe landing area within before batteries discharge 40% of the time.	1.60E-06
1E1	Provide propulsive power	3.83E-12	E	Internal failure of gearbox system Gearbox #2	1	Internal failure of gearbox #2	All	Increased vibrations, decreased efficiency of rotor gearbox #2	Excess drag on gearbox #2 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
1E2	Provide propulsive power	6.44E-12	E	Internal failure of gearbox system Gearbox #2	2	Gearbox #2 failure to transfer torque from motor #2 or combiner gearbox to Rotor #2	All	Unable to transfer torque from motor #2	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motors	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		6.44E-12
1E3	Provide propulsive power	6.44E-12	E	Internal failure of gearbox system Gearbox #2	3	Gearbox #2 Jam - Failure, failure to transfer torque from motor #2 or combiner gearbox to Rotor #2	All	Unable to transfer torque from motor #2	Continued Engine torque on gearbox causes cascading failures/engine fries	Worst case outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		2.00E-01		1.29E-12
1F1	Provide propulsive power	4.20E-07	F	Failure to provide output torque	1	Clutch #2 failure - failure to engage	All	No output from ESC #2 to Motor #2. Motor	Combiner gearbox distributes	Loss of second engine or combiner gearbox may result	B	Visual and audi-	Pilot can avoid flight conditions which require maximum	II	5.00E-01	Assume sprag	2.00E-01	Beta will be highly dependent	4.20E-08

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
				from Motor #2 to Gearbox #2				#2 fails to provide output torque.	torque from remaining drive systems. Increased load on other motor	in loss of aircraft and occupants		ble warning provided to pilot.	torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)			clutch failure modes evenly distributed.		upon detailed design and controllability of aircraft with limited power	
1F2	Provide propulsive power	4.20E-07	F	Failure to provide output torque from Motor #2 to Gearbox #2	2	Clutch #2 failure - failure to disengage	All	No effect on normal operation. In the event of motor failure, clutch failure will transfer torque from other motors.	Combiner gearbox distributes torque from remaining drive systems. Increased load on other motor	Loss of second engine or combiner gearbox may result in loss of aircraft and occupants	N	None	Fire detection and suppression system must contain fire.	II	5.00E-01	Assume sprag clutch failure modes evenly distributed.	4.62E-05	Probability of motor bearing failure or motor-stator contact assumed to be half of motor failure rate.	9.70E-12
2A1	Transfer torque between drive gearboxes	8.62E-09	A	Internal Failure of combiner gearbox #1	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	1.00E+00		1.00E+00		8.62E-09
2A2	Transfer torque between drive gearboxes	9.66E-12	A	Internal Failure of combiner gearbox #1	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of interconnecting drive shafts.	Potential damage to interconnecting drive shafts and rotor gearboxes.	Seized gearbox causes cascading failures of the drive system. Loss of power to rotors. Worst case outcome, loss of aircraft and crew		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		9.66E-12
2B1	Transfer torque between drive gearboxes	8.62E-09	B	Internal Failure of combiner gearbox #2	1	Internal failure of combiner gearbox	All	Increased vibrations, decreased efficiency.	None	Failure is detected, landed as soon as practical		Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	II	1.00E+00		1.00E+00		8.62E-09
2B2	Transfer torque between	9.66E-12	B	Internal Failure of combiner gearbox #2	2	Combiner gearbox seized	All	Combiner gearbox stops turning and causes sudden stop of	Potential damage to interconnecting drive	Seized gearbox causes cascading failures of the drive system. Loss of		Visual and audible alert	VHMS annunciates failure to crew; 30 minute get home capability after crack is	I	1.00E+00		1.00E+00		9.66E-12

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	drive gearboxes							interconnecting drive shafts.	shafts and rotor gearboxes.	power to rotors. Worst case outcome, loss of aircraft and crew		provided to pilot	detected. Anti-flail system prevents damage to neighboring systems.						
2C1	Transfer torque between drive gearboxes	6.52E-17	C	Failure of combiner gearbox interconnecting drive shaft	1	Interconnecting driveshaft shear	All	Loss of ability to transfer torque between drive gearbox and combiner gearbox	Loss of ability to cross-shaft power in the event of a motor 1 failure	Rotors are desynchronized. Loss of power to rotor(s). Loss of control of aircraft. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	VHMS annunciates failure to crew; 30 minute get home capability after crack is detected. Anti-flail system prevents damage to neighboring systems.	I	1.00E+00		1.00E+00		6.52E-17
3A1	Transfer torque from gearbox to rotor	4.45E-11	A	Internal failure of rotor #1 Transmission	1	Internal failure of rotor #1 Transmission	All	Increased vibrations, decreased efficiency of rotor #1	Excess drag on gearbox #1 consumes power from remaining Drive systems.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
3A2	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	2	Internal failure of rotor #1 Transmission	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable. Unable to maintain level flight.	Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
3B1	Transfer torque from gearbox to rotor	4.45E-11	B	Internal failure of rotor #2 Transmission	1	Internal failure of rotor #2 Transmission	All	Increased vibrations, decreased efficiency of rotor #2	Excess drag on gearbox #2 consumes power from remaining Drive systems.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
3B2	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	2	Internal failure of rotor #2 Transmission	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable. Unable to maintain level flight.	Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
3C1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #3 Transmission	1	Internal failure of rotor #3 Transmission	All	Increased vibrations, decreased efficiency of rotor #3	Excess drag on gearbox #3 consumes power from remaining Drive systems.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
3C2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	2	Internal failure of rotor #3 Transmission	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable. Unable to maintain level flight.	Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
3D1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #4 Transmission	1	Internal failure of rotor #4 Transmission	All	Increased vibrations, decreased efficiency of rotor #4	Excess drag on gearbox #4 consumes power from remaining Drive systems.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
3D2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #4 Transmission	2	Internal failure of rotor #4 Transmission	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable. Unable to maintain level flight.	Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E+00		1.00E+00		5.15E-11
4A1	Provide hydraulic control of Rotor pitch	4.46E-10	A	Failure to transfer commanded input to Rotor #1 pitch	1	Failure of triplex actuator #1	All	Loss of ability for control system to command triplex Actuator #1 position	Loss of ability for control system to command a derived pitch input to Rotor #1	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #1 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
4B1	Provide hydraulic control of Rotor pitch	4.46E-10	B	Failure to transfer commanded input to Rotor #2 pitch	1	Failure of triplex actuator #2	All	Loss of ability for control system to command triplex Actuator #2 position	Loss of ability for control system to command a derived pitch input to Rotor #2	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #2 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
4C1	Provide hydraulic control of Rotor pitch	4.46E-10	C	Failure to transfer commanded input to Rotor #3 pitch	1	Failure of triplex actuator #3	All	Loss of ability for control system to command triplex Actuator #3 position	Loss of ability for control system to command a derived pitch input to Rotor #3	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #3 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4D1	Provide hydraulic control of Rotor pitch	4.46E-10	D	Failure to transfer commanded input to Rotor #4 pitch	1	Failure of triplex actuator #4	All	Loss of ability for control system to command triplex Actuator #4 position	Loss of ability for control system to command a derived pitch input to Rotor #4	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #4 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
4E1	Provide hydraulic control of Rotor pitch	4.98E-07	E	Failure of hydraulic pump 1	1	Failure of hydraulic pump #1 resulting in minor performance degradation	All	Internal failure of pump results in degraded flow	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
4E2	Provide hydraulic control of Rotor pitch	4.98E-07	E	Failure of hydraulic pump 1	2	Failure of hydraulic pump #1 resulting in loss of sufficient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow through Pump #1 compensated by two remaining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two remaining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
4F1	Provide hydraulic control of Rotor pitch	4.98E-07	F	Failure of hydraulic pump 1	1	Failure of hydraulic pump #2 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
4F2	Provide hydraulic control of	4.98E-07	F	Failure of hydraulic pump 1	2	Failure of hydraulic pump #2 resulting	All	Internal failure of pump	Loss of Flow through Pump #2 compensated by	None	B	Visual and audible alert	Flight control system compensates for lack	II	8.50E-02	FM distribution based on	1.00E+00	Beta will be highly dependent	4.24E-08

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	Rotor pitch					in loss of sufficient flow		results in severely degraded output/ no output	two remaining pumps			provided to pilot	of flow with two remaining pumps. System designed such that control can be maintained with single pump			FDM -91 pump hydraulic		upon detailed design and controllability of aircraft with limited power	
4G1	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 1	1	Failure of hydraulic pump #3 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenance required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
4G2	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 1	2	Failure of hydraulic pump #3 resulting in loss of sufficient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow through Pump #3 compensated by two remaining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two remaining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribution based on FDM -91 pump hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5A1	Transfer heat generated during operation of motor gearbox	2.73E-07	A	TMS #1 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		2.73E-07
5B1	Transfer heat generated during operation of motor gearbox	1.84E-06	B	Degraded oil flow through TMS #1	1	Leaking	All	Oil leak	Minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		1.84E-06

Table D3: tQuad FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
5C1	Transfer heat generated during operation of motor gearbox	1.84E-06	C	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.84E-06
5E1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #1/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
5F1	Transfer heat generated during operation of motor gearbox	2.73E-07	F	TMS #2 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		2.73E-07
5G1	Transfer heat generated during operation of motor gearbox	1.84E-06	G	Degraded oil flow through TMS #2	1	Leaking	All	Oil leak	Minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		1.84E-06
5H1	Transfer heat generated during operation of motor gearbox	1.84E-06	H	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.84E-06
5J1	Transfer heat generated during operation of motor gearbox	2.73E-07	J	Severely Degraded Flow of oil through TMS #2/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1A1	Provide HVDC power to electric motors	1.00E-06	A	Failure to provide HVDC electrical energy to ESC #1	1	HV Battery output failure or associated wiring. Loss of output to ESC #1 only.	All	No power to ESC #1; Motor #1 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	I	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1B1	Provide HVDC power to electric motors	1.00E-06	B	Failure to provide HVDC electrical energy to ESC #2	1	HV Battery output failure or associated wiring. Loss of output to ESC #2 only.	All	No power to ESC #2; Motor #2 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	I	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1C1	Provide HVDC power to electric motors	1.00E-06	C	Failure to provide HVDC electrical energy to ESC #3	1	HV Battery output failure or associated wiring. Loss of output to ESC #3 only.	All	No power to ESC #3; Motor #3 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #3 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	I	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1D1	Provide HVDC power to electric motors	1.00E-06	D	Failure to provide HVDC electrical energy to ESC #4	1	HV Battery output failure or associated wiring. Loss of output to ESC #4 only.	All	No power to ESC #4; Motor #4 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Limited flight envelope. Reduced maximum speed and insufficient power to take off or hover at max weight. Loss of air raft Possible hard landing if failure occurs while in OMI avoid region (< 20 kts)	HB	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	I	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1E1	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	1	Battery cell failure - no runaway.	All	Loss of output from single branch within battery network. Battery output voltage slightly reduced. Increased current draw from remaining battery cells	Output voltage slightly reduced to one or more motors.	Reduced range and/or slight degradation of motor performance.	B	Visual and audible warning provided to pilot.	Battery monitoring system must detect and isolate the fault. Continued operation with failed cell may put additional stress on other battery cells which must be managed to prevent catastrophic failure	IV	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E2	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	2	Battery cell failure - Thermal runaway. - contained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery monitoring system detects failure, disconnects and isolates the defective battery cell.	Reduced battery system capacity, slight degradation of battery output voltage provided to one or more electric motors. Battery may catch fire. Excess heat generated may affect adjacent battery cells.	Reduced range and/or degradation electric motor performance.	B	Visual and audible warning provided to pilot.	Battery cooling and fire protection system must contain battery temperature to prevent loss of aircraft	III	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E3	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	3	Battery cell failure internal short - thermal runaway - uncontained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery catches fire. Loss of all HVDC output.	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes	1.00E-02	It is assumed that in most cases, battery failure will occur gradually giving the pilot time to land safely.	2.50E-09
1E4	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	4	Complete HV battery failure; low voltage or no voltage output. (Battery discharged)	All	Loss of all HVDC output	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes. Battery system is assumed to have redundancy where multiple internal failures must occur	1.00E-02	It is assumed that in most cases, battery voltage would decrease gradually, giving the pilot time to land safely.	2.50E-09

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																for complete loss of HVDC to all 4 motors.			
2A1	Convert HV electrical energy to shaft torque	2.70E-04	A	ESC #1 A failure	1	ESC #1 A fail low- fail off	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2A2	Convert HV electrical energy to shaft torque	2.70E-04	A	ESC #1 A failure	2	ESC #1 A fail high	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2B1	Convert HV electrical energy to shaft torque	2.70E-04	B	ESC #1 B failure	1	ESC #1 A fail low- fail off	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2B2	Convert HV electrical energy to shaft torque	2.70E-04	B	ESC #1 B failure	2	ESC #2 A fail high	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and	2.70E-10

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
													energy stored in rotors to provide soft landing (autorotate)					controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	
2C1	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #1	1	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of Engine #1	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2C2	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #1	2	Minor Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of motor #1	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.50E-06
2C3	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	3	Internal failure of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2C4	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	4	Motor #1 fails Open	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2C5	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	5	Internal jam of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Power continually applied to failed gearbox raising internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2D1	Convert HV electrical energy to	2.70E-04	D	ESC #2 A failure	1	ESC #2 A fail low- fail off	All	No output from ESC #2 to Motor #2. Motor #2 fails	2nd ESC compensates for loss of function	Loss of air craft/hard landing	B	Visual and audible warn-	Pilot can avoid flight conditions which require maximum torque. Pilot can use	II	5.00E-01		2.00E-01	Beta will be highly dependent	2.70E-05

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	shaft torque							to provide output torque.		possible with second ESC failure of FCC/sensor failure		ing provided to pilot.	energy stored in rotors to provide soft landing (autorotate)					upon detailed design and controllability of aircraft with limited power	
2D2	Convert HV electrical energy to shaft torque	2.70E-04	D	ESC #2 A failure	2	ESC #2 A fail high	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2E1	Convert HV electrical energy to shaft torque	2.70E-04	E	ESC #2 B failure	1	ESC #2 A fail low- fail off	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2E2	Convert HV electrical energy to shaft torque	2.70E-04	E	ESC #2 B failure	2	ESC #2 A fail high	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight	2.70E-10

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		control/sensor failure	
2F1	Convert HV electrical energy to shaft torque	3.50E-06	F	Internal failure of Motor #2	1	Minor Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of Engine #2	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2F2	Convert HV electrical energy to shaft torque	3.50E-06	F	Internal failure of Motor #2	2	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of motor #2	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2F3	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	3	Internal failure of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2F4	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	4	Motor #2 fails Open	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2F5	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	5	Internal jam of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2G1	Convert HV electrical energy to shaft torque	2.70E-04	G	ESC #3 A failure	1	ESC #3 A fail low- fail off	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2G2	Convert HV electrical energy to shaft torque	2.70E-04	G	ESC #3 A failure	2	ESC #3 A fail high	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2H1	Convert HV electrical energy to shaft torque	2.70E-04	H	ESC #3 B failure	1	ESC #3 A fail low- fail off	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2H2	Convert HV electrical energy to shaft torque	2.70E-04	H	ESC #3 B failure	2	ESC #3 A fail high	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2I1	Convert HV electrical energy to shaft torque	3.50E-06	I	Internal failure of Motor #3	1	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of Engine #3	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2I2	Convert HV electrical energy to shaft torque	3.50E-06	I	Internal failure of Motor #3	2	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of motor #3	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.50E-08
2I3	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	3	Internal failure of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.75E-06
2I4	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	4	Motor #3 fails Open	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.80E-09
2I5	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	5	Internal jam of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.80E-08
2J1	Convert HV electrical energy to shaft torque	2.70E-04	J	ESC #4 A failure	1	ESC #4 A fail low- fail off	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and	2.70E-05

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
													energy stored in rotors to provide soft landing (autorotate)					controllability of aircraft with limited power	
2J2	Convert HV electrical energy to shaft torque	2.70E-04	J	ESC #4 A failure	2	ESC #4 A fail high	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2K1	Convert HV electrical energy to shaft torque	2.70E-04	K	ESC #4 B failure	1	ESC #4 A fail low- fail off	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2K2	Convert HV electrical energy to shaft torque	2.70E-04	K	ESC #4 B failure	2	ESC #4 A fail high	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2L1	Convert HV electrical energy to shaft torque	3.50E-06	L	Internal failure of Motor #4	1	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of Engine #4	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2L2	Convert HV electrical energy to shaft torque	3.50E-06	L	Internal failure of Motor #4	2	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of motor #4	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2L3	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	3	Internal failure of Motor #4	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2L4	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	4	Motor #4 fails Open	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2L5	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	5	Internal jam of Motor #4	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2M1	Convert HV electrical energy to shaft torque	2.70E-04	M	ESC #5 A failure	1	ESC #5 A fail low- fail off	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2M2	Convert HV elec-	2.70E-04	M	ESC #5 A failure	2	ESC #5 A fail high	All	No output from ESC #5 to Motor #5.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or	B	Visual and audi-	Pilot can avoid flight conditions which require maximum	II	5.00E-01		2.00E-06	Beta will be highly dependent	2.70E-10

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	trical energy to shaft torque							Motor #5 fails to provide output torque.		FCC/sensor failure; pitch control can manage rotor speed		ble warning provided to pilot.	torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)					upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	
2N1	Convert HV electrical energy to shaft torque	2.70E-04	N	ESC #5 B failure	1	ESC #5 A fail low- fail off	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2N2	Convert HV electrical energy to shaft torque	2.70E-04	N	ESC #5 B failure	2	ESC #5 A fail high	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2O1	Convert HV electrical energy to shaft torque	3.50E-06	O	Internal failure of Motor #5	1	Internal failure of Motor #5	All	Increased vibrations, decreased efficiency of Engine #5	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2O2	Convert HV electrical energy to	3.50E-06	O	Internal failure of Motor #5	2	Internal failure of Motor #5	All	Increased vibrations, de-	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	1.00E+00		1.00E-02		3.50E-08

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	shaft torque							creased efficiency of motor #5				provided to pilot	failure to pilot and crew.						
2O3	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	3	Internal failure of Motor #5	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2O4	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	4	Motor #5 fails Open	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2O5	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	5	Internal jam of Motor #5	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2P1	Convert HV electrical energy to shaft torque	2.70E-04	P	ESC #6 A failure	1	ESC #6 A fail low- fail off	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2P2	Convert HV electrical energy to shaft torque	2.70E-04	P	ESC #6 A failure	2	ESC #6 A fail high	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes	2.70E-10

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		1e-5 flight control/sensor failure	
2Q1	Convert HV electrical energy to shaft torque	2.70E-04	Q	ESC #6 B failure	1	ESC #6 A fail low- fail off	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2Q2	Convert HV electrical energy to shaft torque	2.70E-04	Q	ESC #6 B failure	2	ESC #6 A fail high	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2R1	Convert HV electrical energy to shaft torque	3.50E-06	R	Internal failure of Motor #6	1	Internal failure of Motor #6	All	Increased vibrations, decreased efficiency of Engine #6	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2R2	Convert HV electrical energy to shaft torque	3.50E-06	R	Internal failure of Motor #6	2	Internal failure of Motor #6	All	Increased vibrations, decreased efficiency of motor #6	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2R3	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	3	Internal failure of Motor #6	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2R4	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	4	Motor #6 fails Open	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2R5	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	5	Internal jam of Motor #6	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
3A1	Transfer motor torque to rotors	3.83E-12	A	Internal failure of gearbox system Gearbox #1	1	Internal failure of gearbox #1	All	Increased vibrations, decreased efficiency of rotor gearbox #1	Excess drag on gearbox #1 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3A2	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	2	Gearbox #1 Failure, failure to transfer torque from motor #1 or combiner gearbox to Rotor #1	All	Unable to transfer torque from motor #1	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3A3	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	3	Gearbox #1 Failure, gearbox open	All	Unable to transfer torque from motor #1	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3B1	Transfer motor torque to rotors	3.83E-12	B	Internal failure of gearbox system Gearbox #2	1	Internal failure of gearbox #2	All	Increased vibrations, decreased efficiency of rotor gearbox #2	Excess drag on gearbox #2 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3B2	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	2	Gearbox #2 Failure, failure to transfer torque from motor #2 or com-	All	Unable to transfer torque from motor #2	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
						biner gearbox to Rotor #2													
3B3	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	3	Gearbox #2 Failure, gearbox open	All	Unable to transfer torque from motor #2	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3C1	Transfer motor torque to rotors	3.83E-12	C	Internal failure of gearbox system Gearbox #3	1	Internal failure of gearbox #3	All	Increased vibrations, decreased efficiency of rotor gearbox #3	Excess drag on gearbox #3 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3C2	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	2	Gearbox #3 Failure, failure to transfer torque from motor #3 or combiner gearbox to Rotor #3	All	Unable to transfer torque from motor #3	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3C3	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	3	Gearbox #3 Failure, gearbox open	All	Unable to transfer torque from motor #3	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3D1	Transfer motor torque to rotors	3.83E-12	D	Internal failure of gearbox system Gearbox #4	1	Internal failure of gearbox #4	All	Increased vibrations, decreased efficiency of rotor gearbox #4	Excess drag on gearbox #4 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3D2	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	2	Gearbox #4 Failure, failure to transfer torque from motor #4 or combiner gearbox to Rotor #4	All	Unable to transfer torque from motor #4	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3D3	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	3	Gearbox #4 Failure, gearbox open	All	Unable to transfer torque from motor #4	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
3E1	Transfer motor torque to rotors	3.83E-12	E	Internal failure of gearbox system Gearbox #5	1	Internal failure of gearbox #5	All	Increased vibrations, decreased efficiency of rotor gearbox #5	Excess drag on gearbox #5 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3E2	Transfer motor torque to rotors	6.44E-12	E	Internal failure of gearbox system Gearbox #5	2	Gearbox #5 Failure, failure to transfer torque from motor #5 or combiner gearbox to Rotor #5	All	Unable to transfer torque from motor #5	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3E3	Transfer motor torque to rotors	6.44E-12	E	Internal failure of gearbox system Gearbox #5	3	Gearbox #5 Failure, gearbox open	All	Unable to transfer torque from motor #5	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3F1	Transfer motor torque to rotors	3.83E-12	F	Internal failure of gearbox system Gearbox #6	1	Internal failure of gearbox #6	All	Increased vibrations, decreased efficiency of rotor gearbox #6	Excess drag on gearbox #6 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3F2	Transfer motor torque to rotors	6.44E-12	F	Internal failure of gearbox system Gearbox #6	2	Gearbox #6 Failure, failure to transfer torque from motor #6 or combiner gearbox to Rotor #6	All	Unable to transfer torque from motor #6	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3F3	Transfer motor torque to rotors	6.44E-12	F	Internal failure of gearbox system Gearbox #6	3	Gearbox #6 Failure, gearbox open	All	Unable to transfer torque from motor #6	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Pitch control can manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
4A1	Transfer torque from gearbox to rotor	4.45E-11	A	Internal failure of rotor #1 Transmission	1	Internal failure of rotor #1 Transmission	All	Increased vibrations, decreased efficiency of rotor #1	Excess drag on gearbox #1 consumes power from remaining 5 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4A2	Transfer torque from	5.15E-11	A	Internal failure of rotor #1 Transmission	2	Internal failure of rotor #1 Transmission	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from	Remaining rotors copensate for loss of lift, land as soon	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	I	8.00E-01		1.00E+00		4.12E-11

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	gearbox to rotor								Motor #1 unusable.	as practicle; potential loss of aircraft with failure of torque to second rotor		provided to pilot	failure to pilot and crew.						
4A3	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	3	Internal failure of rotor #1 Transmission - Jam	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4A4	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	4	Internal failure of rotor #1 Transmission - Open	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable.	Pitch control required to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4B1	Transfer torque from gearbox to rotor	4.45E-11	B	Internal failure of rotor #2 Transmission	1	Internal failure of rotor #2 Transmission	All	Increased vibrations, decreased efficiency of rotor #2	Excess drag on gearbox #2 consumes power from remaining 5 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4B2	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	2	Internal failure of rotor #2 Transmission	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potential loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4B3	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	3	Internal failure of rotor #2 Transmission - Jam	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4B4	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	4	Internal failure of rotor #2 Transmission - Open	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Pitch control required to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4C1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #3 Transmission	1	Internal failure of rotor #3 Transmission	All	Increased vibrations, decreased efficiency of rotor #3	Excess drag on gearbox #3 consumes power from remaining 5 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4C2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	2	Internal failure of rotor #3 Transmission	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Remaining rotors copensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4C3	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	3	Internal failure of rotor #3 Transmission - Jam	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4C4	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	4	Internal failure of rotor #3 Transmission - Open	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Pitch control rquired to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4D1	Transfer torque from gearbox to rotor	4.45E-11	D	Internal failure of rotor #4 Transmission	1	Internal failure of rotor #4 Transmission	All	Increased vibrations, decreased efficiency of rotor #4	Excess drag on gearbox #4 consumes power from remaining 5 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4D2	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	2	Internal failure of rotor #4 Transmission	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Remaining rotors copensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4D3	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	3	Internal failure of rotor #4 Transmission - Jam	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4D4	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	4	Internal failure of rotor #4 Transmission - Open	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Pitch control rquired to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4E1	Transfer torque from	4.45E-11	E	Internal failure of rotor	1	Internal failure of rotor	All	Increased vibrations, de-	Excess drag on gearbox #5 consumes power	Failure is detected, landed as soon as practical	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	1.00E+00		1.00E+00		4.45E-11

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	gearbox to rotor			#5 Transmission		#5 Transmission		creased efficiency of rotor #5	from remaining 5 motors.			provided to pilot	failure to pilot and crew.						
4E2	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	2	Internal failure of rotor #5 Transmission	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Remaining rotors copensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4E3	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	3	Internal failure of rotor #5 Transmission - Jam	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4E4	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	4	Internal failure of rotor #5 Transmission - Open	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Pitch control required to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4F1	Transfer torque from gearbox to rotor	4.45E-11	F	Internal failure of rotor #6 Transmission	1	Internal failure of rotor #6 Transmission	All	Increased vibrations, decreased efficiency of rotor #6	Excess drag on gearbox #6 consumes power from remaining 5 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4F2	Transfer torque from gearbox to rotor	5.15E-11	F	Internal failure of rotor #6 Transmission	2	Internal failure of rotor #6 Transmission	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from Motor #6 unusable.	Remaining rotors copensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4F3	Transfer torque from gearbox to rotor	5.15E-11	F	Internal failure of rotor #6 Transmission	3	Internal failure of rotor #6 Transmission - Jam	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from Motor #6 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4F4	Transfer torque from gearbox to rotor	5.15E-11	F	Internal failure of rotor #6 Transmission	4	Internal failure of rotor #6 Transmission - Open	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from Motor #6 unusable.	Pitch control required to manage rotor speed, overspeed possible with secondary failure of control system	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
5A1	Provide hydraulic control of Rotor pitch	4.46E-10	A	Failure to transfer commanded input to Rotor #1 pitch	1	Failure of triplex actuator #1	All	Loss of ability for control system to command triplex Actuator #1 position	Loss of ability for control system to command a derived pitch input to Rotor #1	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #1 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5B1	Provide hydraulic control of Rotor pitch	4.46E-10	B	Failure to transfer commanded input to Rotor #2 pitch	1	Failure of triplex actuator #2	All	Loss of ability for control system to command triplex Actuator #2 position	Loss of ability for control system to command a derived pitch input to Rotor #2	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #2 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5C1	Provide hydraulic control of Rotor pitch	4.46E-10	C	Failure to transfer commanded input to Rotor #3 pitch	1	Failure of triplex actuator #3	All	Loss of ability for control system to command triplex Actuator #3 position	Loss of ability for control system to command a derived pitch input to Rotor #3	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #3 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5D1	Provide hydraulic control of Rotor pitch	4.46E-10	D	Failure to transfer commanded input to Rotor #4 pitch	1	Failure of triplex actuator #4	All	Loss of ability for control system to command triplex Actuator #4 position	Loss of ability for control system to command a derived pitch input to Rotor #4	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #4 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5E1	Provide hydraulic control of Rotor pitch	4.46E-10	E	Failure to transfer commanded input to Rotor #5 pitch	1	Failure of triplex actuator #5	All	Loss of ability for control system to command triplex	Loss of ability for control system to command a derived pitch input to Rotor #5	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #5 and adjusts controls to other 3 ro-	I	1.00E+00		1.00E+00	Beta will be highly dependent upon de-	4.46E-10

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								Actuator #5 position					tors to maintain limited control and soft landing					tailed design and controllability of aircraft with limited power	
5F1	Provide hydraulic control of Rotor pitch	4.46E-10	F	Failure to transfer commanded input to Rotor #6 pitch	1	Failure of triplex actuator #6	All	Loss of ability for control system to command triplex Actuator #6 position	Loss of ability for control system to command a derived pitch input to Rotor #6	Loss of flight-path control. Loss of aircraft and crew	B	Visual and audible alert provided to pilot	Flight control system removes power to Rotor #6 and adjusts controls to other 3 rotors to maintain limited control and soft landing	I	1.00E+00		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.46E-10
5G1	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 1	1	Failure of hydraulic pump #1 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5H2	Provide hydraulic control of Rotor pitch	4.98E-07	H	Failure of hydraulic pump 1	2	Failure of hydraulic pump #1 resulting in loss of sufficient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #1 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5I1	Provide hydraulic control of Rotor pitch	4.98E-07	I	Failure of hydraulic pump #2	1	Failure of hydraulic pump #2 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft	4.56E-07

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		with limited power	
5J2	Provide hydraulic control of Rotor pitch	4.98E-07	J	Failure of hydraulic pump #2	2	Failure of hydraulic pump #2 resulting in loss of suffecient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #2 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5K1	Provide hydraulic control of Rotor pitch	4.98E-07	K	Failure of hydraulic pump #3	1	Failure of hydraulic pump #3 resulting in minor performance degradation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5L2	Provide hydraulic control of Rotor pitch	4.98E-07	L	Failure of hydraulic pump #3	2	Failure of hydraulic pump #3 resulting in loss of suffecient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #3 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
6A1	Transfer heat generated during operation of motor gearbox	3.62E-07	A	TMS #1 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6B1	Transfer heat generated during operation of motor gearbox	2.73E-07	B	Degraded oil flow through TMS #1	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07

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6C1	Transfer heat generated during operation of motor gearbox	1.49E-05	C	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6D1	Transfer heat generated during operation of motor gearbox	2.73E-07	D	Loss of excess oil return path preventing flow to motor #1	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6E1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #1/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6F1	Transfer heat generated during operation of motor gearbox	3.62E-07	F	TMS #2 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6G1	Transfer heat generated during operation of motor gearbox	2.73E-07	G	Degraded oil flow through TMS #2	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6H1	Transfer heat generated during operation of motor gearbox	1.49E-05	H	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6I1	Transfer heat generated during operation of motor gearbox	2.73E-07	I	Loss of excess oil return path preventing flow to motor #2	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6J1	Transfer heat generated during operation of motor gearbox	2.73E-07	J	Severely Degraded Flow of oil through TMS #2/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6K1	Transfer heat generated during operation of motor gearbox	3.62E-07	K	TMS #3 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6L1	Transfer heat generated during operation of motor gearbox	2.73E-07	L	Degraded oil flow through TMS #3	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6M1	Transfer heat generated during operation of motor gearbox	1.49E-05	M	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6N1	Transfer heat generated during operation of motor gearbox	2.73E-07	N	Loss of excess oil return path preventing flow to motor #3	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6O1	Transfer heat generated during operation of motor gearbox	2.73E-07	O	Severely Degraded Flow of oil through TMS #3/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6P1	Transfer heat generated during operation	3.62E-07	P	TMS #4 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	of motor gearbox																		
6Q1	Transfer heat generated during operation of motor gearbox	2.73E-07	Q	Degraded oil flow through TMS #4	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6R1	Transfer heat generated during operation of motor gearbox	1.49E-05	R	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6S1	Transfer heat generated during operation of motor gearbox	2.73E-07	S	Loss of excess oil return path preventing flow to motor #4	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6T1	Transfer heat generated during operation of motor gearbox	2.73E-07	T	Severely Degraded Flow of oil through TMS #4/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6U1	Transfer heat generated during operation of motor gearbox	3.62E-07	U	TMS #5 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6V1	Transfer heat generated during operation of motor gearbox	2.73E-07	V	Degraded oil flow through TMS #5	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6W1	Transfer heat generated during operation	1.49E-05	W	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	of motor gearbox									seal damage/structural damage/leak									
6X1	Transfer heat generated during operation of motor gearbox	2.73E-07	X	Loss of excess oil return path preventing flow to motor #5	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6Y1	Transfer heat generated during operation of motor gearbox	2.73E-07	Y	Severely Degraded Flow of oil through TMS #5/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6Z1	Transfer heat generated during operation of motor gearbox	3.62E-07	Z	TMS #6 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6AA1	Transfer heat generated during operation of motor gearbox	2.73E-07	AA	Degraded oil flow through TMS #6	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6AB1	Transfer heat generated during operation of motor gearbox	1.49E-05	AB	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6AC1	Transfer heat generated during operation of motor gearbox	2.73E-07	AC	Loss of excess oil return path preventing flow to motor #6	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07

Table D4: Collective Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6AD1	Transfer heat generated during operation of motor gearbox	2.73E-07	AD	Severely Degraded Flow of oil through TMS #6/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
1A1	Provide HVDC power to electric motors	1.00E-06	A	Failure to provide HVDC electrical energy to ESC #1	1	HV Battery output failure or associated wiring. Loss of output to ESC #1 only.	All	No power to ESC #1; Motor #1 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #1 through combiner gearbox. Available power reduced	Possible loos of ocupants and aircraft with secondary failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1B1	Provide HVDC power to electric motors	1.00E-06	B	Failure to provide HVDC electrical energy to ESC #2	1	HV Battery output failure or associated wiring. Loss of output to ESC #2 only.	All	No power to ESC #2; Motor #2 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #2 through combiner gearbox. Available power reduced	Possible loos of ocupants and aircraft with secondary failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1C1	Provide HVDC power to electric motors	1.00E-06	C	Failure to provide HVDC electrical energy to ESC #3	1	HV Battery output failure or associated wiring. Loss of output to ESC #3 only.	All	No power to ESC #3; Motor #3 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #3 through combiner gearbox. Available power reduced	Possible loos of ocupants and aircraft with secondary failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1D1	Provide HVDC power to electric motors	1.00E-06	D	Failure to provide HVDC electrical energy to ESC #4	1	HV Battery output failure or associated wiring. Loss of output to ESC #4 only.	All	No power to ESC #4; Motor #4 fails to provide output torque	Torque from other 3 motors is transferred to gearbox #4 through combiner gearbox. Available power reduced	Possible loos of ocupants and aircraft with secondary failure	HB	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	2.50E-01	1 of 4 outputs	2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	5.00E-08
1E1	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	1	Battery cell failure - no runaway.	All	Loss of output from single branch within battery network. Battery output voltage	Output voltage slightly reduced to one or more motors.	Reduced range and/or slight degradation of motor performance.	B	Visual and audible warning provided to pilot.	Battery monitoring system must detect and isolate the fault. Continued operation with failed cell may put additional stress	IV	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								slightly reduced. Increased current draw from remaining battery cells					on other battery cells which must be managed to prevent catastrophic failure						
1E2	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	2	Battery cell failure - Thermal runaway. - contained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery monitoring system detects failure, disconnects and isolates the defective battery cell.	Reduced battery system capacity, slight degradation of battery output voltage provided to one or more electric motors. Battery may catch fire. Excess heat generated may affect adjacent battery cells.	Reduced range and/or degradation electric motor performance.	B	Visual and audible warning provided to pilot.	Battery cooling and fire protection system must contain battery temperature to prevent loss of aircraft	III	2.50E-01	1 of 4 internal battery failure modes	1.00E+00	Beta = 1 for Cat III & Cat 4 FM's	2.50E-07
1E3	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	3	Battery cell failure internal short - thermal runaway - uncontained	All	Battery cell temperature rises rapidly, causing thermal runaway. Battery catches fire. Loss of all HVDC output.	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes	1.00E-02	It is assumed that in most cases, battery failure will occur gradually giving the pilot time to land safely.	2.50E-09
1E4	Provide HVDC power to electric motors	1.00E-06	E	Internal battery failure	4	Complete HV battery failure; low voltage or no voltage output. (Battery discharged)	All	Loss of all HVDC output	No power provided to electric motors. Loss of torque output to rotors	Aircraft descends to ground. Autorotation employed to provide soft landing	HB	Visual and audible warning provided to pilot. Pilot detects loss of power	Flight control system and rotor pitch control actuators are powered by a low voltage battery which is still operational. Controlled landing possible through autorotation	II	2.50E-01	1 of 4 internal battery failure modes. Battery system is assumed to have redundancy where multiple internal failures must occur for complete loss of HVDC to all 4 motors.	1.00E-02	It is assumed that in most cases, battery voltage would decrease gradually, giving the pilot time to land safely.	2.50E-09

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2A1	Convert HV electrical energy to shaft torque	2.70E-04	A	ESC #1 A failure	1	ESC #1 A fail low- fail off	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2A2	Convert HV electrical energy to shaft torque	2.70E-04	A	ESC #1 A failure	2	ESC #1 A fail high	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2B1	Convert HV electrical energy to shaft torque	2.70E-04	B	ESC #1 B failure	1	ESC #1 A fail low- fail off	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2B2	Convert HV electrical energy to shaft torque	2.70E-04	B	ESC #1 B failure	2	ESC #2 A fail high	All	No output from ESC #1 to Motor #1. Motor #1 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes	2.70E-10

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		1e-5 flight control/sensor failure	
2C1	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #1	1	Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of Engine #1	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2C2	Convert HV electrical energy to shaft torque	3.50E-06	C	Internal failure of Motor #1	2	Minor Internal failure of Motor #1	All	Increased vibrations, decreased efficiency of motor #1	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.50E-06
2C3	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	3	Internal failure of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2C4	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	4	Motor #1 fails Open	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2C5	Convert HV electrical energy to shaft torque	3.80E-06	C	Internal failure of Motor #1	5	Internal jam of Motor #1	All	Unable to transfer torque gearbox #1.	No torque from Motor #1. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2D1	Convert HV electrical energy to shaft torque	2.70E-04	D	ESC #2 A failure	1	ESC #2 A fail low- fail off	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft	2.70E-05

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		with limited power	
2D2	Convert HV electrical energy to shaft torque	2.70E-04	D	ESC #2 A failure	2	ESC #2 A fail high	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2E1	Convert HV electrical energy to shaft torque	2.70E-04	E	ESC #2 B failure	1	ESC #2 A fail low- fail off	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2E2	Convert HV electrical energy to shaft torque	2.70E-04	E	ESC #2 B failure	2	ESC #2 A fail high	All	No output from ESC #2 to Motor #2. Motor #2 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2F1	Convert HV electrical energy to	3.50E-06	F	Internal failure of Motor #2	1	Minor Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of Engine #2	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	shaft torque																		
2F2	Convert HV electrical energy to shaft torque	3.50E-06	F	Internal failure of Motor #2	2	Internal failure of Motor #2	All	Increased vibrations, decreased efficiency of motor #2	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2F3	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	3	Internal failure of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2F4	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	4	Motor #2 fails Open	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2F5	Convert HV electrical energy to shaft torque	3.80E-06	F	Internal failure of Motor #2	5	Internal jam of Motor #2	All	Unable to transfer torque gearbox #2.	No torque from Motor #2. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of occupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2G1	Convert HV electrical energy to shaft torque	2.70E-04	G	ESC #3 A failure	1	ESC #3 A fail low- fail off	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2G2	Convert HV electrical energy to shaft torque	2.70E-04	G	ESC #3 A failure	2	ESC #3 A fail high	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and	2.70E-10

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	
2H1	Convert HV electrical energy to shaft torque	2.70E-04	H	ESC #3 B failure	1	ESC #3 A fail low- fail off	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2H2	Convert HV electrical energy to shaft torque	2.70E-04	H	ESC #3 B failure	2	ESC #3 A fail high	All	No output from ESC #3 to Motor #3. Motor #3 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2I1	Convert HV electrical energy to shaft torque	3.50E-06	I	Internal failure of Motor #3	1	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of Engine #3	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2I2	Convert HV electrical energy to shaft torque	3.50E-06	I	Internal failure of Motor #3	2	Internal failure of Motor #3	All	Increased vibrations, decreased efficiency of motor #3	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02	Beta will be highly dependent upon detailed design and	3.50E-08

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		controllability of aircraft with limited power	
2I3	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	3	Internal failure of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.75E-06
2I4	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	4	Motor #3 fails Open	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.80E-09
2I5	Convert HV electrical energy to shaft torque	3.80E-06	I	Internal failure of Motor #3	5	Internal jam of Motor #3	All	Unable to transfer torque gearbox #3.	No torque from Motor #3. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	3.80E-08
2J1	Convert HV electrical energy to shaft torque	2.70E-04	J	ESC #4 A failure	1	ESC #4 A fail low- fail off	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2J2	Convert HV electrical energy to shaft torque	2.70E-04	J	ESC #4 A failure	2	ESC #4 A fail high	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2K1	Convert HV electrical energy to shaft torque	2.70E-04	K	ESC #4 B failure	1	ESC #4 A fail low- fail off	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-05
2K2	Convert HV electrical energy to shaft torque	2.70E-04	K	ESC #4 B failure	2	ESC #4 A fail high	All	No output from ESC #4 to Motor #4. Motor #4 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2L1	Convert HV electrical energy to shaft torque	3.50E-06	L	Internal failure of Motor #4	1	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of Engine #4	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2L2	Convert HV electrical energy to shaft torque	3.50E-06	L	Internal failure of Motor #4	2	Internal failure of Motor #4	All	Increased vibrations, decreased efficiency of motor #4	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2L3	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	3	Internal failure of Motor #4	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2L4	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	4	Motor #4 fails Open	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2L5	Convert HV electrical energy to shaft torque	3.80E-06	L	Internal failure of Motor #4	5	Internal jam of Motor #4	All	Unable to transfer torque gearbox #4.	No torque from Motor #4. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2M1	Convert HV electrical energy to shaft torque	2.70E-04	M	ESC #5 A failure	1	ESC #5 A fail low- fail off	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2M2	Convert HV elec-	2.70E-04	M	ESC #5 A failure	2	ESC #5 A fail high	All	No output from ESC #5 to Motor #5.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or	B	Visual and audi-	Pilot can avoid flight conditions which require maximum	II	5.00E-01		2.00E-06	Beta will be highly dependent	2.70E-10

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	trical energy to shaft torque							Motor #5 fails to provide output torque.		FCC/sensor failure; pitch control can manage rotor speed		ble warning provided to pilot.	torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)					upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	
2N1	Convert HV electrical energy to shaft torque	2.70E-04	N	ESC #5 B failure	1	ESC #5 A fail low- fail off	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2N2	Convert HV electrical energy to shaft torque	2.70E-04	N	ESC #5 B failure	2	ESC #5 A fail high	All	No output from ESC #5 to Motor #5. Motor #5 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2O1	Convert HV electrical energy to shaft torque	3.50E-06	O	Internal failure of Motor #5	1	Internal failure of Motor #5	All	Increased vibrations, decreased efficiency of Engine #5	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2O2	Convert HV electrical energy to	3.50E-06	O	Internal failure of Motor #5	2	Internal failure of Motor #5	All	Increased vibrations, de-	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	1.00E+00		1.00E-02		3.50E-08

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	shaft torque							creased efficiency of motor #5				provided to pilot	failure to pilot and crew.						
2O3	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	3	Internal failure of Motor #5	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06
2O4	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	4	Motor #5 fails Open	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2O5	Convert HV electrical energy to shaft torque	3.80E-06	O	Internal failure of Motor #5	5	Internal jam of Motor #5	All	Unable to transfer torque gearbox #5.	No torque from Motor #5. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
2P1	Convert HV electrical energy to shaft torque	2.70E-04	P	ESC #6 A failure	1	ESC #6 A fail low- fail off	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2P2	Convert HV electrical energy to shaft torque	2.70E-04	P	ESC #6 A failure	2	ESC #6 A fail high	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes	2.70E-10

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
																		1e-5 flight control/sensor failure	
2Q1	Convert HV electrical energy to shaft torque	2.70E-04	Q	ESC #6 B failure	1	ESC #6 A fail low- fail off	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd Esc compensates for loss of function	Loss of air craft/hard landing possible with second ESC failure of FCC/sensor failure	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-01	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	2.70E-05
2Q2	Convert HV electrical energy to shaft torque	2.70E-04	Q	ESC #6 B failure	2	ESC #6 A fail high	All	No output from ESC #6 to Motor #6. Motor #6 fails to provide output torque.	2nd ESC compensates for loss of function	Rotor Overspeed possible with 2nd esc failure or FCC/sensor failure; pitch control can manage rotor speed	B	Visual and audible warning provided to pilot.	Pilot can avoid flight conditions which require maximum torque. Pilot can use energy stored in rotors to provide soft landing (autorotate)	II	5.00E-01		2.00E-06	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power. Includes 1e-5 flight control/sensor failure	2.70E-10
2R1	Convert HV electrical energy to shaft torque	3.50E-06	R	Internal failure of Motor #6	1	Internal failure of Motor #6	All	Increased vibrations, decreased efficiency of Engine #6	None	Unscheduled maintenance	B	Functional checks and inspections	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	III	1.00E+00		1.00E+00		3.50E-06
2R2	Convert HV electrical energy to shaft torque	3.50E-06	R	Internal failure of Motor #6	2	Internal failure of Motor #6	All	Increased vibrations, decreased efficiency of motor #6	Excess drag on remaining 3 motors	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E-02		3.50E-08
2R3	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	3	Internal failure of Motor #6	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Torque from the remaining rotors compensate.	Flying qualified unaffected. Pilot alerted, land as soon as practical. Continued safe flight and landing possible.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	9.89E-01		1.00E+00		3.75E-06

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
2R4	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	4	Motor #6 fails Open	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Potential undesirable rotor-speed condition. Possible loss of rotating machinery parts due to over-speed condition	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-03		1.00E+00		3.80E-09
2R5	Convert HV electrical energy to shaft torque	3.80E-06	R	Internal failure of Motor #6	5	Internal jam of Motor #6	All	Unable to transfer torque gearbox #6.	No torque from Motor #6. Power continually applied to failed gearbox rasing internal temperature.	Motor Fire; Worst case possible outcome: loss of ocupents/aircraft.	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-02		1.00E+00		3.80E-08
3A1	Transfer motor torque to rotors	3.83E-12	A	Internal failure of gearbox system Gearbox #1	1	Internal failure of gearbox #1	All	Increased vibrations, decreased efficiency of rotor gearbox #1	Excess drag on gearbox #1 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3A2	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	2	Gearbox #1 Failure, failure to transfer torque from motor #1 or combiner gearbox to Rotor #1	All	Unable to transfer torque from motor #1	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12
3A3	Transfer motor torque to rotors	6.44E-12	A	Internal failure of gearbox system Gearbox #1	3	Gearbox #1 Failure, gearbox open	All	Unable to transfer torque from motor #1	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3B1	Transfer motor torque to rotors	3.83E-12	B	Internal failure of gearbox system Gearbox #2	1	Internal failure of gearbox #2	All	Increased vibrations, decreased efficiency of rotor gearbox #2	Excess drag on gearbox #2 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3B2	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	2	Gearbox #2 Failure, failure to transfer torque from motor #2 or com-	All	Unable to transfer torque from motor #2	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
						biner gearbox to Rotor #2													
3B3	Transfer motor torque to rotors	6.44E-12	B	Internal failure of gearbox system Gearbox #2	3	Gearbox #2 Failure, gearbox open	All	Unable to transfer torque from motor #2	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3C1	Transfer motor torque to rotors	3.83E-12	C	Internal failure of gearbox system Gearbox #3	1	Internal failure of gearbox #3	All	Increased vibrations, decreased efficiency of rotor gearbox #3	Excess drag on gearbox #3 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3C2	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	2	Gearbox #3 Failure, failure to transfer torque from motor #3 or combiner gearbox to Rotor #3	All	Unable to transfer torque from motor #3	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12
3C3	Transfer motor torque to rotors	6.44E-12	C	Internal failure of gearbox system Gearbox #3	3	Gearbox #3 Failure, gearbox open	All	Unable to transfer torque from motor #3	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3D1	Transfer motor torque to rotors	3.83E-12	D	Internal failure of gearbox system Gearbox #4	1	Internal failure of gearbox #4	All	Increased vibrations, decreased efficiency of rotor gearbox #4	Excess drag on gearbox #4 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3D2	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	2	Gearbox #4 Failure, failure to transfer torque from motor #4 or combiner gearbox to Rotor #4	All	Unable to transfer torque from motor #4	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12
3D3	Transfer motor torque to rotors	6.44E-12	D	Internal failure of gearbox system Gearbox #4	3	Gearbox #4 Failure, gearbox open	All	Unable to transfer torque from motor #4	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
3E1	Transfer motor torque to rotors	3.83E-12	E	Internal failure of gearbox system Gearbox #5	1	Internal failure of gearbox #5	All	Increased vibrations, decreased efficiency of rotor gearbox #5	Excess drag on gearbox #5 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3E2	Transfer motor torque to rotors	6.44E-12	E	Internal failure of gearbox system Gearbox #5	2	Gearbox #5 Failure, failure to transfer torque from motor #5 or combiner gearbox to Rotor #5	All	Unable to transfer torque from motor #5	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12
3E3	Transfer motor torque to rotors	6.44E-12	E	Internal failure of gearbox system Gearbox #5	3	Gearbox #5 Failure, gearbox open	All	Unable to transfer torque from motor #5	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
3F1	Transfer motor torque to rotors	3.83E-12	F	Internal failure of gearbox system Gearbox #6	1	Internal failure of gearbox #6	All	Increased vibrations, decreased efficiency of rotor gearbox #6	Excess drag on gearbox #6 consumes power from remaining motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		3.83E-12
3F2	Transfer motor torque to rotors	6.44E-12	F	Internal failure of gearbox system Gearbox #6	2	Gearbox #6 Failure, failure to transfer torque from motor #6 or combiner gearbox to Rotor #6	All	Unable to transfer torque from motor #6	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Flying qualified unaffected. Pilot alerted, land as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	5.00E-01		1.00E+00		3.22E-12
3F3	Transfer motor torque to rotors	6.44E-12	F	Internal failure of gearbox system Gearbox #6	3	Gearbox #6 Failure, gearbox open	All	Unable to transfer torque from motor #6	Remainign rotors copenstate with thrust necissary for continued safe fligth and landing	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	5.00E-01		1.00E+00		3.22E-12
4A1	Transfer torque from gearbox to rotor	4.45E-11	A	Internal failure of rotor #1 Transmission	1	Internal failure of rotor #1 Transmission	All	Increased vibrations, decreased efficiency of rotor #1	Excess drag on gearbox #1 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4A2	Transfer torque from	5.15E-11	A	Internal failure of rotor #1 Transmission	2	Internal failure of rotor #1 Transmission	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from	Remaining rotors copensate for loss of lift, land as soon	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	II	8.00E-01		1.00E+00		4.12E-11

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	gearbox to rotor								Motor #1 unusable.	as practicle; potentail loss of aircraft with failure of torque to second rotor		provided to pilot	failure to pilot and crew.						
4A3	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	3	Internal failure of rotor #1 Transmission - Jam	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4A4	Transfer torque from gearbox to rotor	5.15E-11	A	Internal failure of rotor #1 Transmission	4	Internal failure of rotor #1 Transmission - Open	All	Unable to transfer torque Rotor #1.	No lift available to Rotor #1. Torque from Motor #1 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4B1	Transfer torque from gearbox to rotor	4.45E-11	B	Internal failure of rotor #2 Transmission	1	Internal failure of rotor #2 Transmission	All	Increased vibrations, decreased efficiency of rotor #2	Excess drag on gearbox #2 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4B2	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	2	Internal failure of rotor #2 Transmission	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	8.00E-01		1.00E+00		4.12E-11
4B3	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	3	Internal failure of rotor #2 Transmission - Jam	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4B4	Transfer torque from gearbox to rotor	5.15E-11	B	Internal failure of rotor #2 Transmission	4	Internal failure of rotor #2 Transmission - Open	All	Unable to transfer torque Rotor #2.	No lift available to Rotor #2. Torque from Motor #2 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E-01		1.00E+00		5.15E-12

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4C1	Transfer torque from gearbox to rotor	4.45E-11	C	Internal failure of rotor #3 Transmission	1	Internal failure of rotor #3 Transmission	All	Increased vibrations, decreased efficiency of rotor #3	Excess drag on gearbox #3 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4C2	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	2	Internal failure of rotor #3 Transmission	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potential loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	8.00E-01		1.00E+00		4.12E-11
4C3	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	3	Internal failure of rotor #3 Transmission - Jam	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4C4	Transfer torque from gearbox to rotor	5.15E-11	C	Internal failure of rotor #3 Transmission	4	Internal failure of rotor #3 Transmission - Open	All	Unable to transfer torque Rotor #3.	No lift available to Rotor #3. Torque from Motor #3 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potential loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4D1	Transfer torque from gearbox to rotor	4.45E-11	D	Internal failure of rotor #4 Transmission	1	Internal failure of rotor #4 Transmission	All	Increased vibrations, decreased efficiency of rotor #4	Excess drag on gearbox #4 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4D2	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	2	Internal failure of rotor #4 Transmission	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potential loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	8.00E-01		1.00E+00		4.12E-11
4D3	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	3	Internal failure of rotor #4 Transmission - Jam	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
4D4	Transfer torque from gearbox to rotor	5.15E-11	D	Internal failure of rotor #4 Transmission	4	Internal failure of rotor #4 Transmission - Open	All	Unable to transfer torque Rotor #4.	No lift available to Rotor #4. Torque from Motor #4 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4E1	Transfer torque from gearbox to rotor	4.45E-11	E	Internal failure of rotor #5 Transmission	1	Internal failure of rotor #5 Transmission	All	Increased vibrations, decreased efficiency of rotor #5	Excess drag on gearbox #5 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4E2	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	2	Internal failure of rotor #5 Transmission	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	8.00E-01		1.00E+00		4.12E-11
4E3	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	3	Internal failure of rotor #5 Transmission - Jam	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Potential for jam to causes cascading failures of rotor leading to rotor destruction; worst case outcome: loss of aircraft/ocupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4E4	Transfer torque from gearbox to rotor	5.15E-11	E	Internal failure of rotor #5 Transmission	4	Internal failure of rotor #5 Transmission - Open	All	Unable to transfer torque Rotor #5.	No lift available to Rotor #5. Torque from Motor #5 unusable.	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
4F1	Transfer torque from gearbox to rotor	4.45E-11	F	Internal failure of rotor #6 Transmission	1	Internal failure of rotor #6 Transmission	All	Increased vibrations, decreased efficiency of rotor #6	Excess drag on gearbox #6 consumes power from remaining 3 motors.	Failure is detected, landed as soon as practical	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	1.00E+00		1.00E+00		4.45E-11
4F2	Transfer torque from gearbox to rotor	5.15E-11	F	Internal failure of rotor #6 Transmission	2	Internal failure of rotor #6 Transmission	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from Motor #6 unusable.	Remaining rotors compensate for loss of lift, land as soon as practicle; potentail loss of aircraft with failure of torque to second rotor	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	II	8.00E-01		1.00E+00		4.12E-11
4F3	Transfer torque from	5.15E-11	F	Internal failure of rotor	3	Internal failure of rotor	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from	Potential for jam to causes cascading failures of rotor	B	Visual and audible alert	Chip detector, electrified debris screen, or VHMS annunciate	I	1.00E-01		1.00E+00		5.15E-12

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	gearbox to rotor			#6 Transmission		#6 Transmission - Jam			Motor #6 unusable.	leading to rotor destruction; worst case outcome: loss of aircraft/ocupants		provided to pilot	failure to pilot and crew.						
4F4	Transfer torque from gearbox to rotor	5.15E-11	F	Internal failure of rotor #6 Transmission	4	Internal failure of rotor #6 Transmission - Open	All	Unable to transfer torque Rotor #6.	No lift available to Rotor #6. Torque from Motor #6 unusable.	Open rotor shaft may result in rotor overspeed. Worstcase outcome loss of aircraft and occupants	B	Visual and audible alert provided to pilot	Chip detector, electrified debris screen, or VHMS annunciate failure to pilot and crew.	I	1.00E-01		1.00E+00		5.15E-12
5G1	Provide hydraulic control of Rotor pitch	4.98E-07	G	Failure of hydraulic pump 1	1	Failure of hydraulic pump #1 resulting in minor performance degredation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5H2	Provide hydraulic control of Rotor pitch	4.98E-07	H	Failure of hydraulic pump 1	2	Failure of hydraulic pump #1 resulting in loss of suffecient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #1 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
5I1	Provide hydraulic control of Rotor pitch	4.98E-07	I	Failure of hydraulic pump #2	1	Failure of hydraulic pump #2 resulting in minor performance degredation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5J2	Provide hydraulic control of Rotor pitch	4.98E-07	J	Failure of hydraulic pump #2	2	Failure of hydraulic pump #2 resulting in loss of suffecient flow	All	Internal failure of pump results in severely de-	Loss of Flow trhough Pump #2 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and	4.24E-08

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FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								graded output/ no output					be maintained with single pump					controllability of aircraft with limited power	
5K1	Provide hydraulic control of Rotor pitch	4.98E-07	K	Failure of hydraulic pump #3	1	Failure of hydraulic pump #3 resulting in minor performance degredation	All	Internal failure of pump results	No effect on system performance; unscheduled maintenaince required	None	B	Visual and audible alert provided to pilot	None	IV	9.15E-01	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.56E-07
5L2	Provide hydraulic control of Rotor pitch	4.98E-07	L	Failure of hydraulic pump #3	2	Failure of hydraulic pump #3 resulting in loss of sufficient flow	All	Internal failure of pump results in severely degraded output/ no output	Loss of Flow trhough Pump #3 compensated by two reamining pumps	None	B	Visual and audible alert provided to pilot	Flight control system compensates for lack of flow with two reamining pumps. System designed such that control can be maintained with single pump	II	8.50E-02	FM distribuiton based on FDM -91 pumpp hydraulic	1.00E+00	Beta will be highly dependent upon detailed design and controllability of aircraft with limited power	4.24E-08
6A1	Transfer heat generated during operation of motor gearbox	3.62E-07	A	TMS #1 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6B1	Transfer heat generated during operation of motor gearbox	2.73E-07	B	Degraded oil flow through TMS #1	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6C1	Transfer heat generated during operation of motor gearbox	1.49E-05	C	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
6D1	Transfer heat generated during operation of motor gearbox	2.73E-07	D	Loss of excess oil return path preventing flow to motor #1	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6E1	Transfer heat generated during operation of motor gearbox	2.73E-07	E	Severely Degraded Flow of oil through TMS #1/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6F1	Transfer heat generated during operation of motor gearbox	3.62E-07	F	TMS #2 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6G1	Transfer heat generated during operation of motor gearbox	2.73E-07	G	Degraded oil flow through TMS #2	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6H1	Transfer heat generated during operation of motor gearbox	1.49E-05	H	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6I1	Transfer heat generated during operation of motor gearbox	2.73E-07	I	Loss of excess oil return path preventing flow to motor #2	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6J1	Transfer heat generated during operation of motor gearbox	2.73E-07	J	Severely Degraded Flow of oil through TMS #2/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely de-	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period;	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
								graded output/ no output		effected motor shut down									
6K1	Transfer heat generated during operation of motor gearbox	3.62E-07	K	TMS #3 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6L1	Transfer heat generated during operation of motor gearbox	2.73E-07	L	Degraded oil flow through TMS #3	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6M1	Transfer heat generated during operation of motor gearbox	1.49E-05	M	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6N1	Transfer heat generated during operation of motor gearbox	2.73E-07	N	Loss of excess oil return path preventing flow to motor #3	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6O1	Transfer heat generated during operation of motor gearbox	2.73E-07	O	Severely Degraded Flow of oil through TMS #3/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6P1	Transfer heat generated during operation of motor gearbox	3.62E-07	P	TMS #4 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6Q1	Transfer heat generated during	2.73E-07	Q	Degraded oil flow through TMS #4	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures;	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	operation of motor gearbox									potential degraded component life									
6R1	Transfer heat generated during operation of motor gearbox	1.49E-05	R	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6S1	Transfer heat generated during operation of motor gearbox	2.73E-07	S	Loss of excess oil return path preventing flow to motor #4	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6T1	Transfer heat generated during operation of motor gearbox	2.73E-07	T	Severely Degraded Flow of oil through TMS #4/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6U1	Transfer heat generated during operation of motor gearbox	3.62E-07	U	TMS #5 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6V1	Transfer heat generated during operation of motor gearbox	2.73E-07	V	Degraded oil flow through TMS #5	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6W1	Transfer heat generated during operation of motor gearbox	1.49E-05	W	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6X1	Transfer heat generated during	2.73E-07	X	Loss of excess oil return path preventing	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert	Interconnected Gearboxes allow	II	1.00E+00		1.00E+00		2.73E-07

Table D5: RPM-Controlled eHex FMECA Table

FMECA ID Code	Function	Failure Rate (λ)	FMI - Mode	Failure Mode	FMI - Cause	Failure Cause	Mission Phase	Local Failure Effect	Next Higher Effect	End Effect	Detection Code	Detection Method	Compensating Provisions	Severity Code	Alpha (Mode Ratio)	Alpha Mode Ratio Explanation	Beta	Beta Mode Ratio Explanation	Failure Mode Criticality No.
	operation of motor gearbox			flow to motor #5					pressure down stream			provided to pilot	rotor to continue operation						
6Y1	Transfer heat generated during operation of motor gearbox	2.73E-07	Y	Severely Degraded Flow of oil through TMS #5/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07
6Z1	Transfer heat generated during operation of motor gearbox	3.62E-07	Z	TMS #6 valve stuck open	1	Mechanical jam	All	Valve mechanically stuck open	None	None	N	None	Redundant Systems	IV	1.00E+00		1.00E+00		3.62E-07
6AA1	Transfer heat generated during operation of motor gearbox	2.73E-07	AA	Degraded oil flow through TMS #6	1	Leaking	All	Oil leak	minor loss of flow down stream of leak	Increased system temperatures; potential degraded component life	HB	None	Redundant Systems	III	1.00E+00		1.00E+00		2.73E-07
6AB1	Transfer heat generated during operation of motor gearbox	1.49E-05	AB	Loss of excess oil return path	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	150% flow downstream of jam	Over pressurized system, potential seal damage/structural damage/leak	B	Visual and audible alert provided to pilot	Down stream design rated for overflow	II	1.00E+00		1.00E+00		1.49E-05
6AC1	Transfer heat generated during operation of motor gearbox	2.73E-07	AC	Loss of excess oil return path preventing flow to motor #6	1	Mechanical Jam/debris	All	Valve mechanically stuck closed	High temperature and low pressure down stream	Motor shutdown; oil flow to gearbox continues	B	Visual and audible alert provided to pilot	Interconnected Gearboxes allow rotor to continue operation	II	1.00E+00		1.00E+00		2.73E-07
6AD1	Transfer heat generated during operation of motor gearbox	2.73E-07	AD	Severely Degraded Flow of oil through TMS #6/ No Flow	1	Mechanical Jam/debris	All	Internal failure of pump results in severely degraded output/ no output	Insufficient flow through cooling system	No thermal management system; transmission enters 30min oil out operating period; effected motor shut down	B	Visual and audible alert provided to pilot	Thermal management system pressure/temperature sensors	II	1.00E+00		1.00E+00		2.73E-07

Appendix E Fault Tree Diagrams

The aircraft level fault tree models for the eQuad ‘Loss of Propulsion’ and ‘Loss of Collective Control’ are included in Table E1. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page numbering. All of the failure rates are come from the FMECA failure rates associated with Table D1.

The aircraft level fault tree models for the hQuad ‘Loss of Propulsion’ and ‘Loss of Collective Control’ are included in Table E2. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are come from the FMECA failure rates associated with Table D2.

The aircraft level fault tree models for the eQuad ‘Loss of Propulsion’ and ‘Loss of Collective Control’ are included in Table E3. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are come from the FMECA failure rates associated with Table D3.

The aircraft level fault tree models for the non-cross-shafted eQuad ‘Loss of Propulsion’ and ‘Loss of Collective Control’ are included in Table E4. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are based on the cross-shafted eQuad FMECA failure rates associated with Table D1, but the FMECA IDs may not be directed correlated.

The aircraft level fault tree model for the non-cross-shafted eQuad ‘Loss of Propulsion’ is included in Table E5. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are based on the cross-shafted eQuad FMECA failure rates associated with Table D1, but the FMECA IDs may not be directed correlated.

The aircraft level fault tree models for the Pitch-Hex ‘Loss of Propulsion’ and ‘Loss of Collective Control’ are included in Table E6. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are come from the FMECA failure rates associated with Table D4.

The aircraft level fault tree model for the RPM-Hex ‘Loss of Propulsion’ is included in Table E7. Subgates of the Loss of Propulsion fault tree for visibility and page numbering links under cutsets are relative to this page as Page 1. All of the failure rates are come from the FMECA failure rates associated with Table D5.

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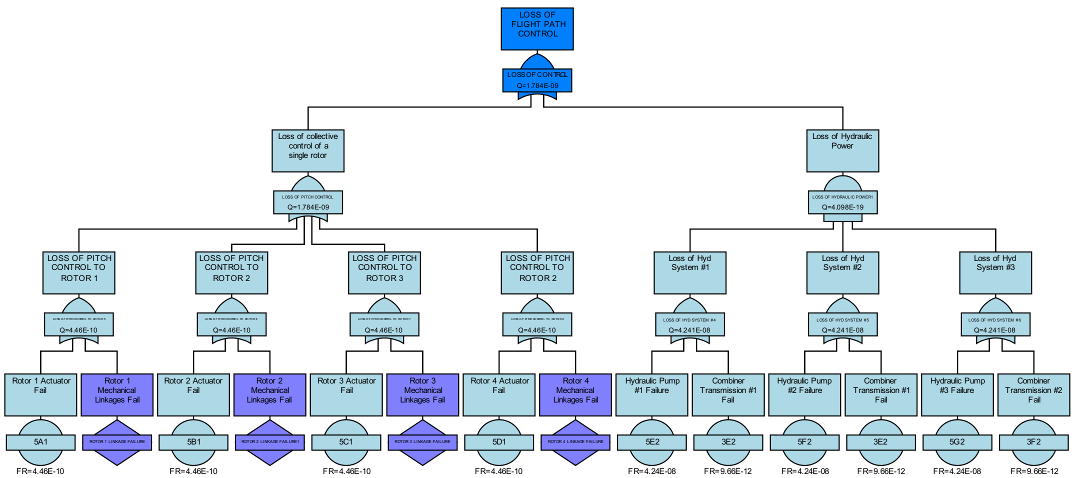
Project Diagrams

Table E1: eQuad Fault Tree Diagram



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Project Diagrams

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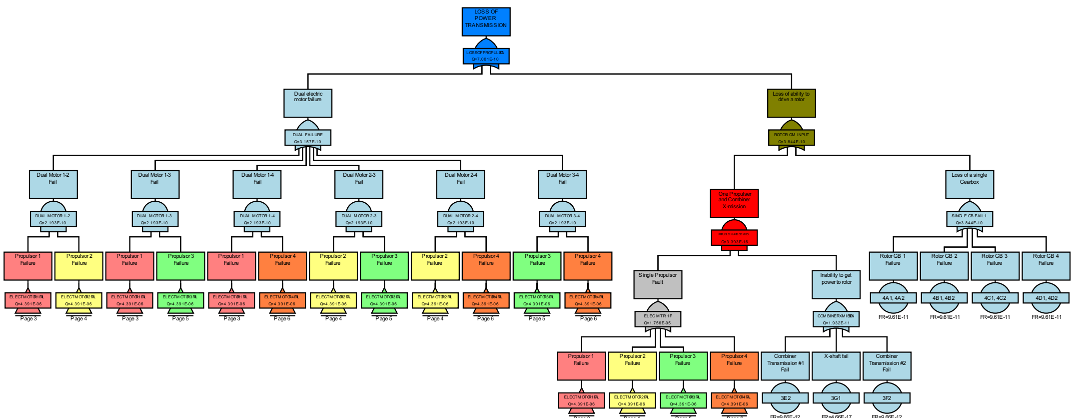

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Project Diagrams

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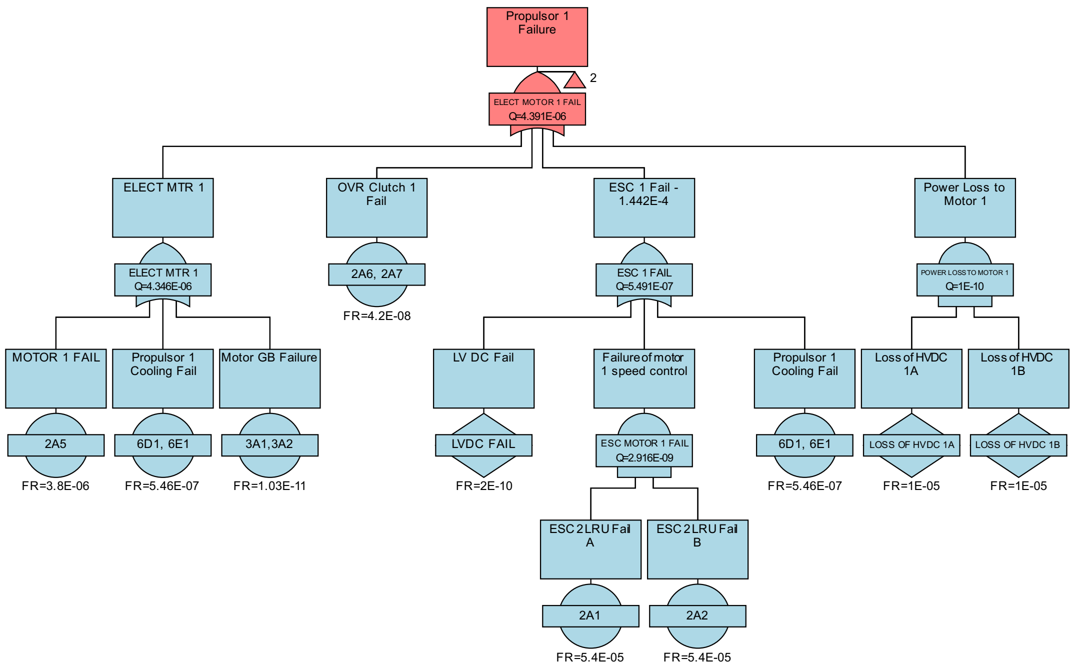

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Project Diagrams

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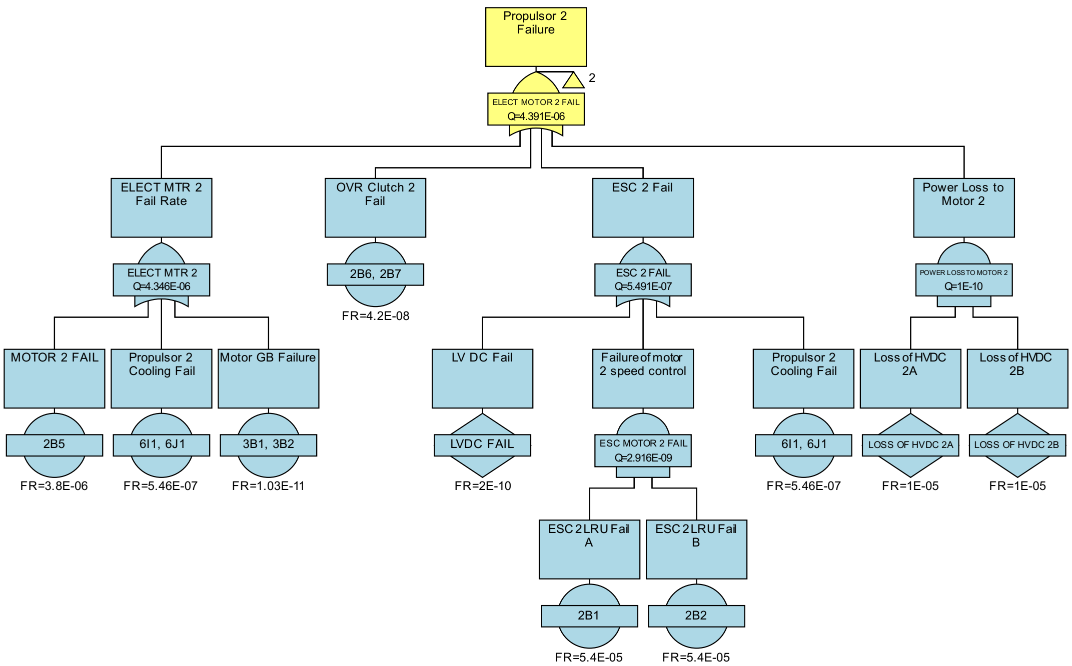

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Project Diagrams

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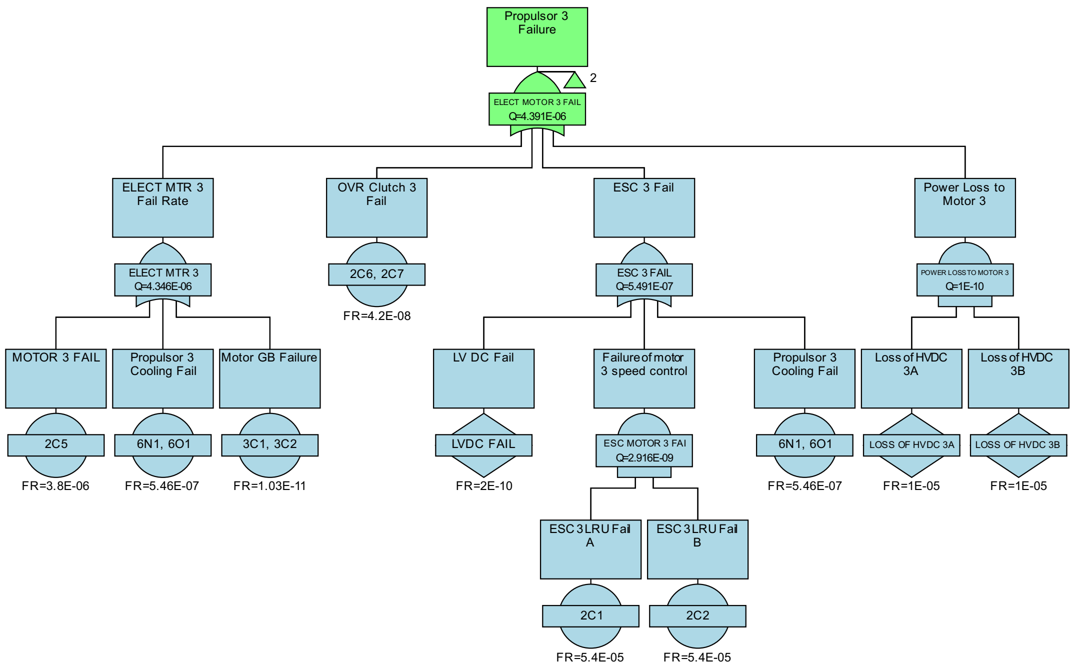

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Project Diagrams

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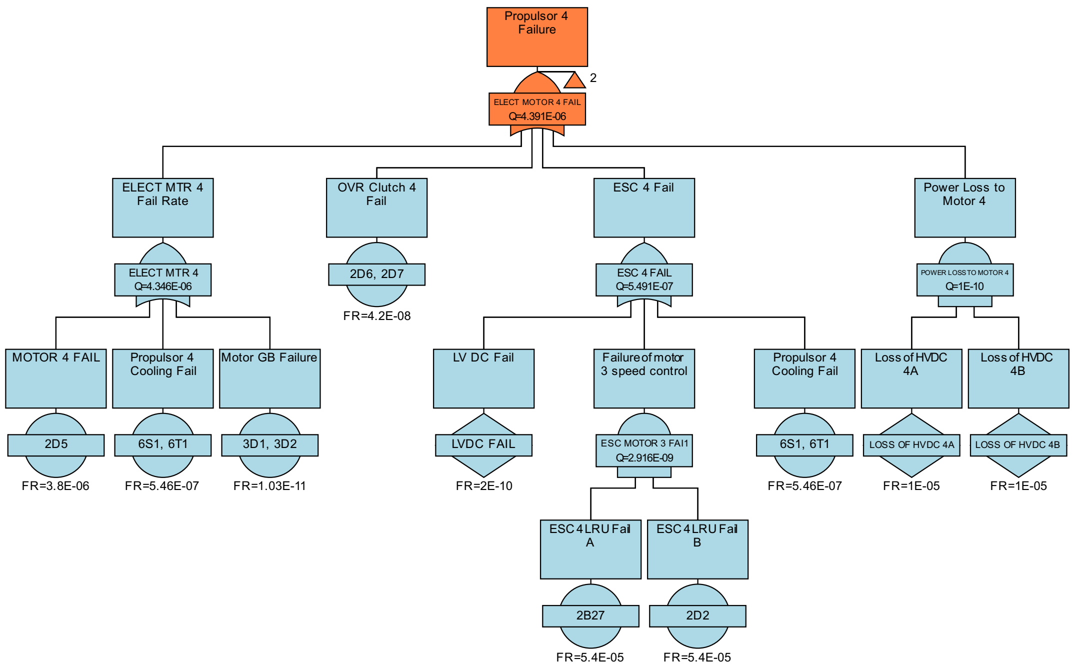

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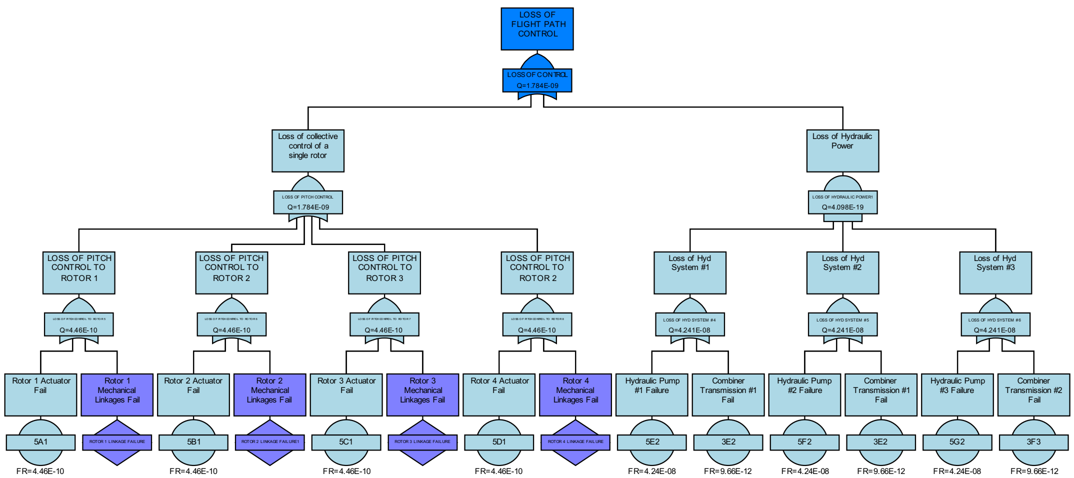
Project Diagrams

Table E2: hQuad Fault Tree Diagram



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Project Diagrams

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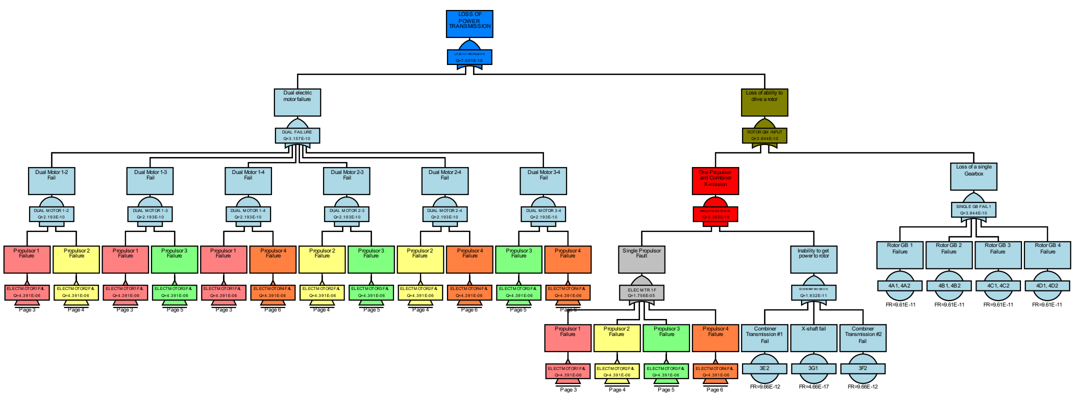

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Project Diagrams

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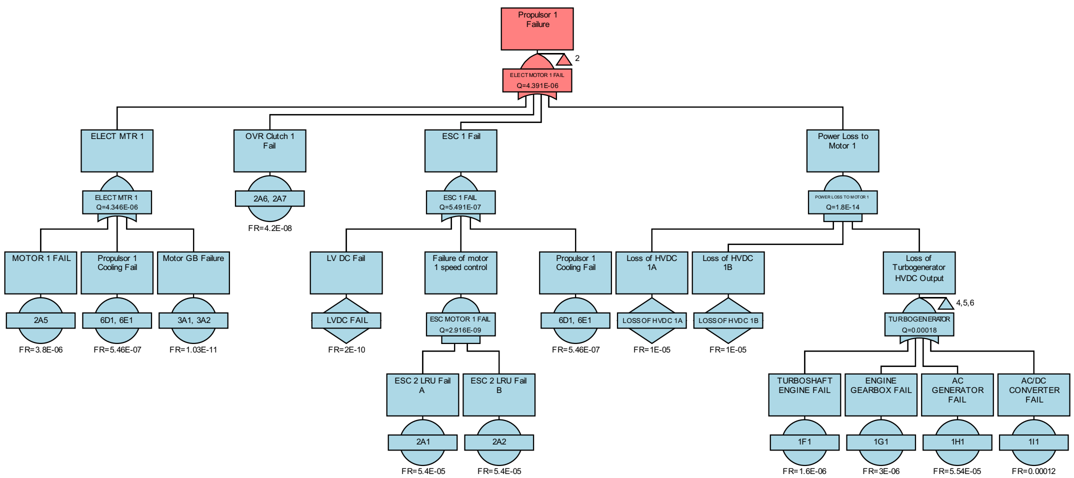

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Project Diagrams

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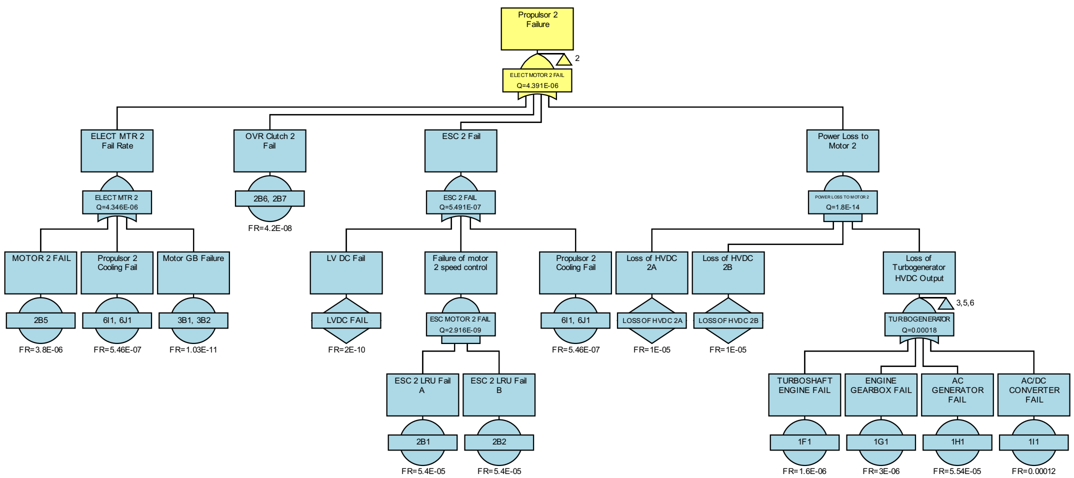

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Project Diagrams

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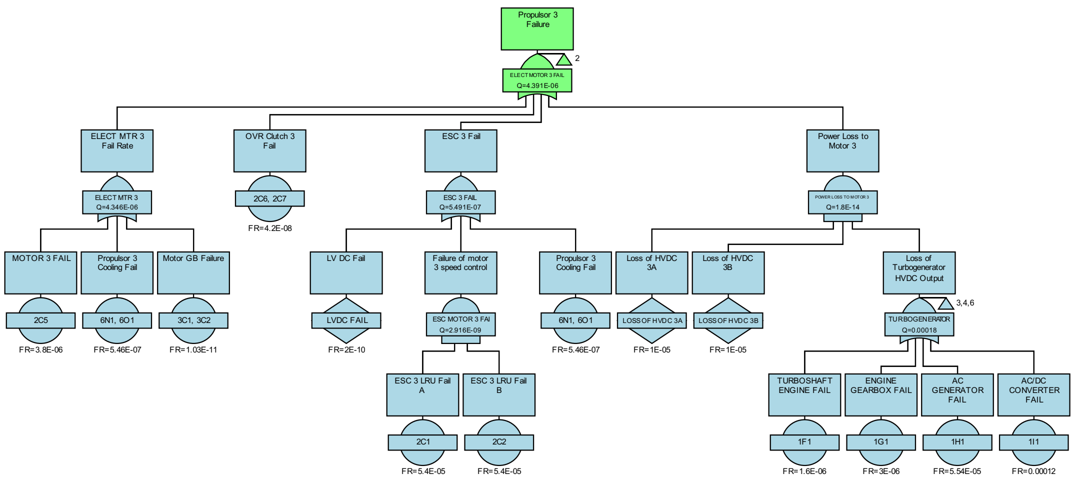

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Project Diagrams

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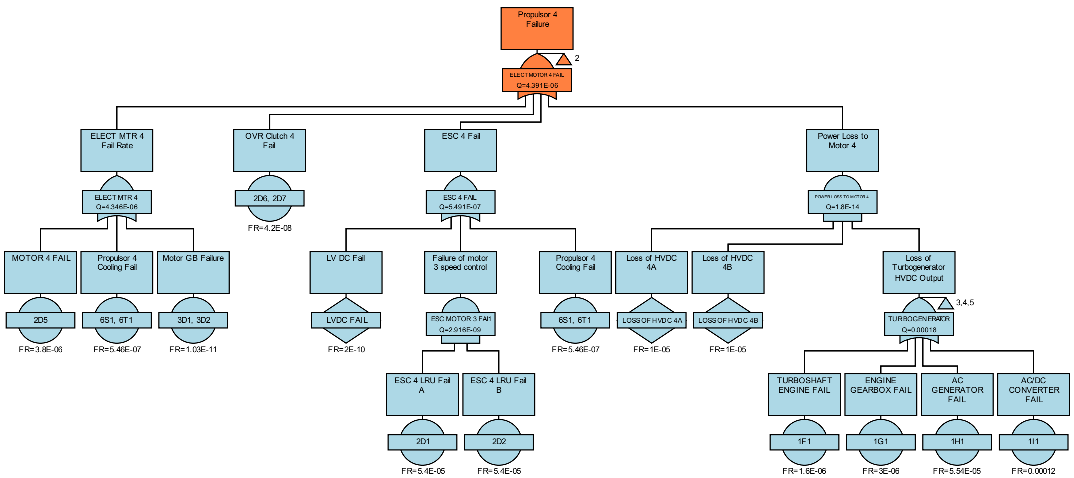

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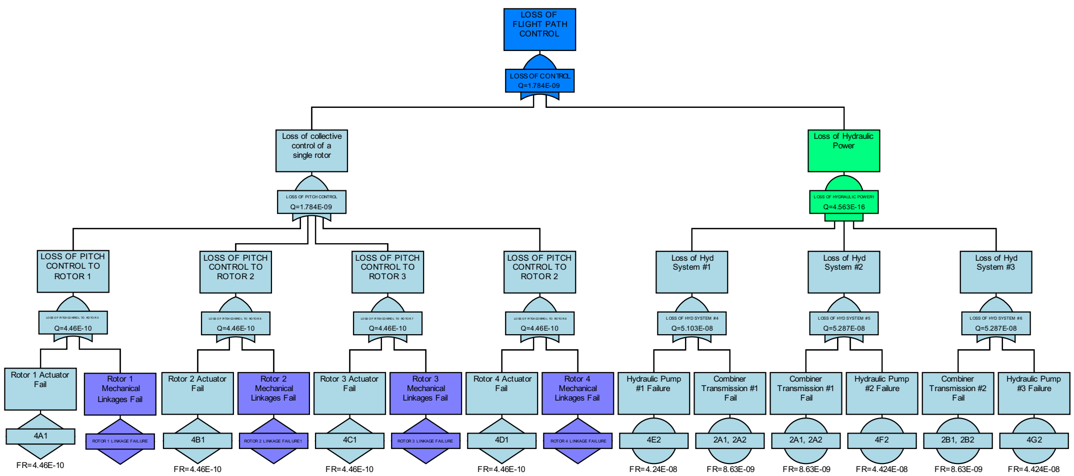
Project Diagrams

Table E3: tQuad Fault Tree Diagram



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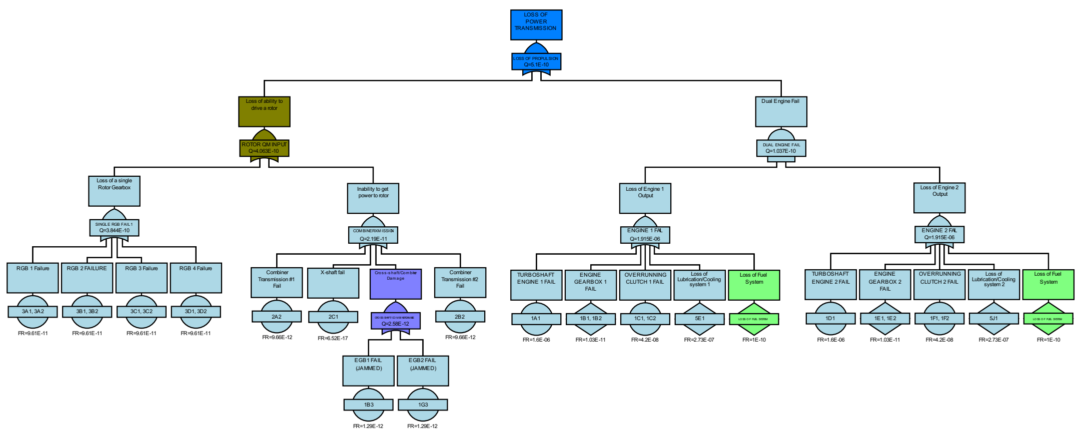
Project Diagrams

Table E3: tQuad Fault Tree Diagram



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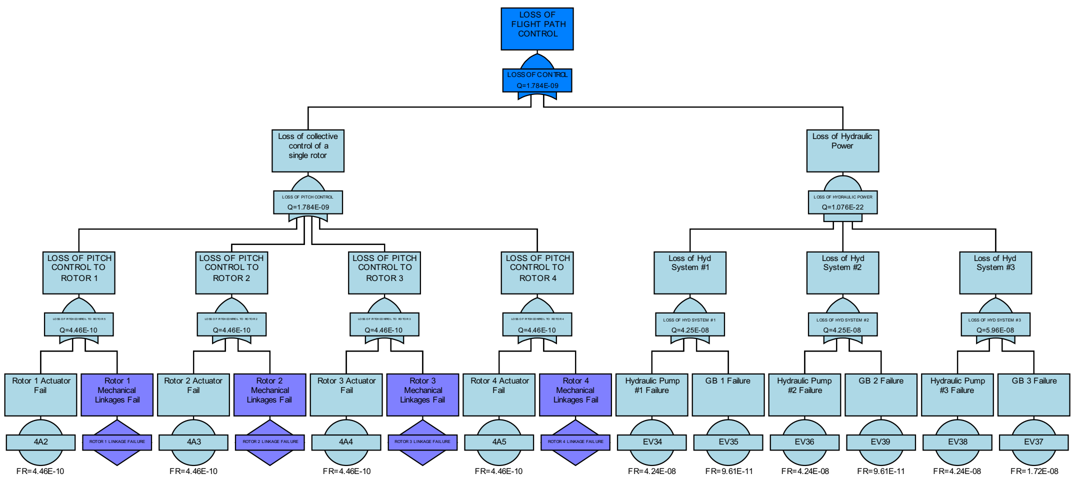
Project Diagrams

Table E4: eQuad without Cross-Shafts Fault Tree Diagram



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Project Diagrams

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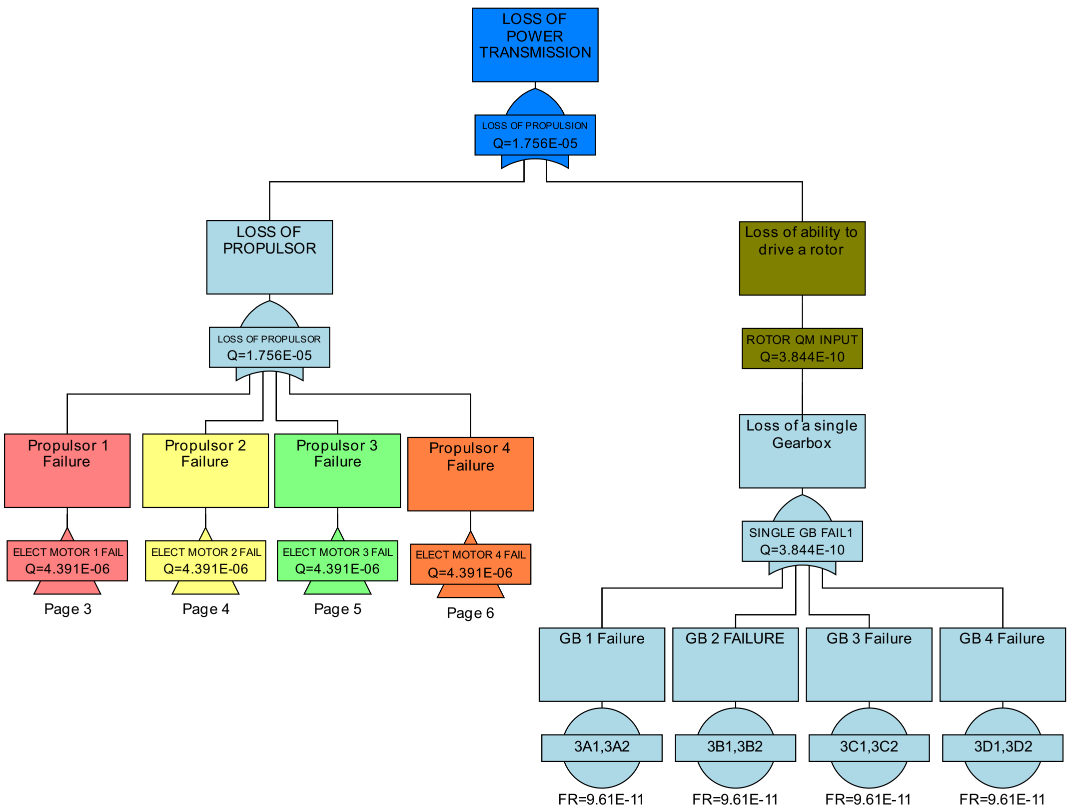

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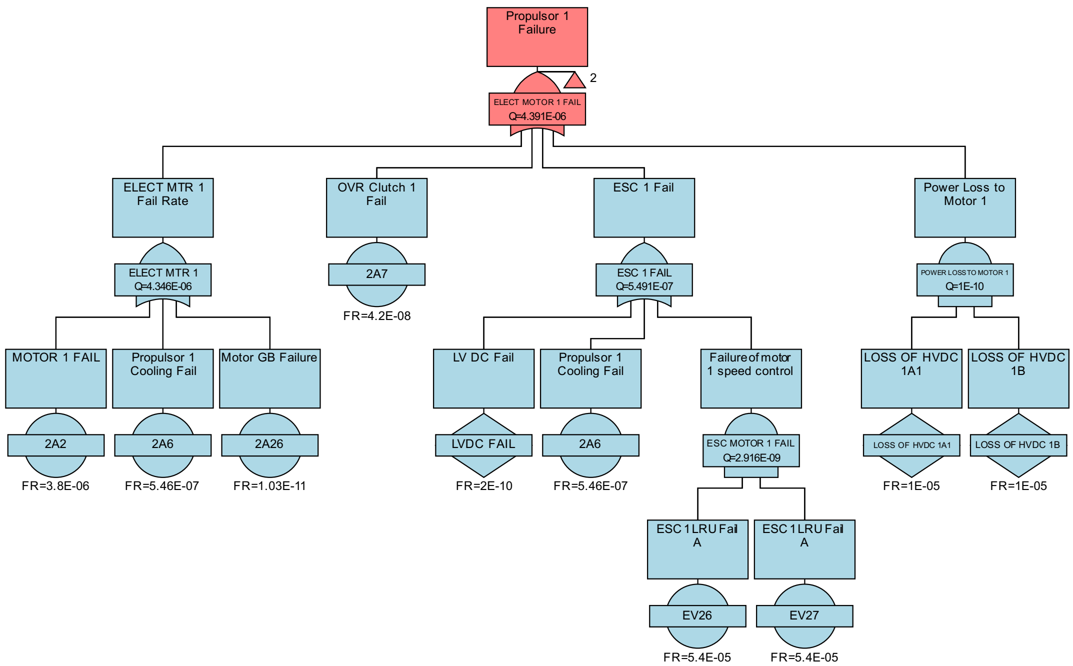
Project Diagrams

Table E4: eQuad without Cross-Shafts Fault Tree Diagram



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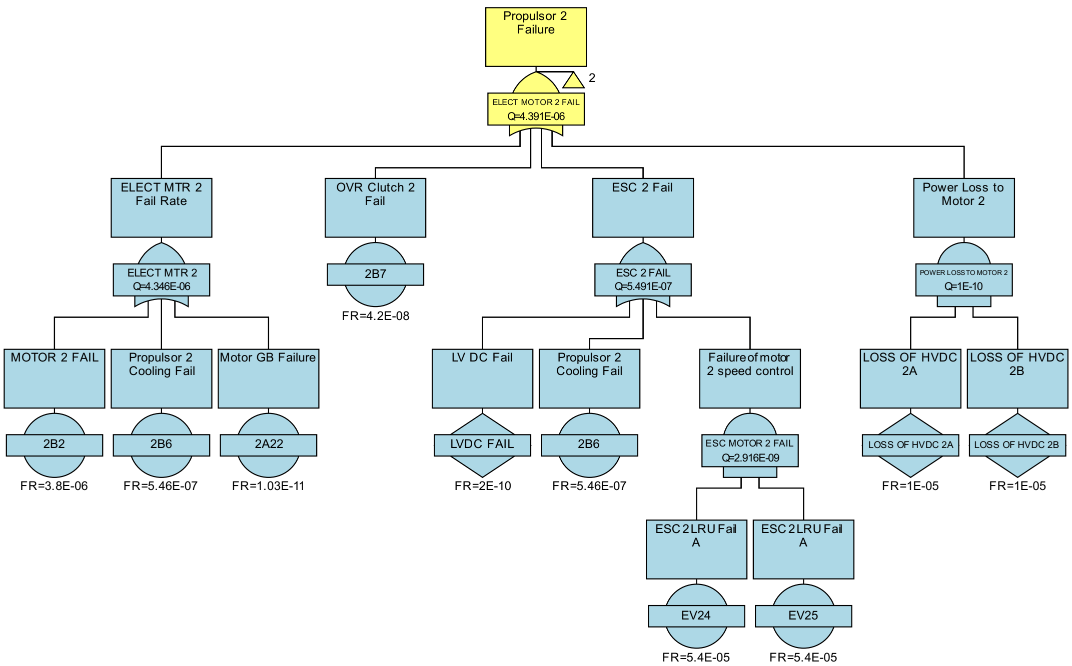
Project Diagrams

Table E4: eQuad without Cross-Shafts Fault Tree Diagram



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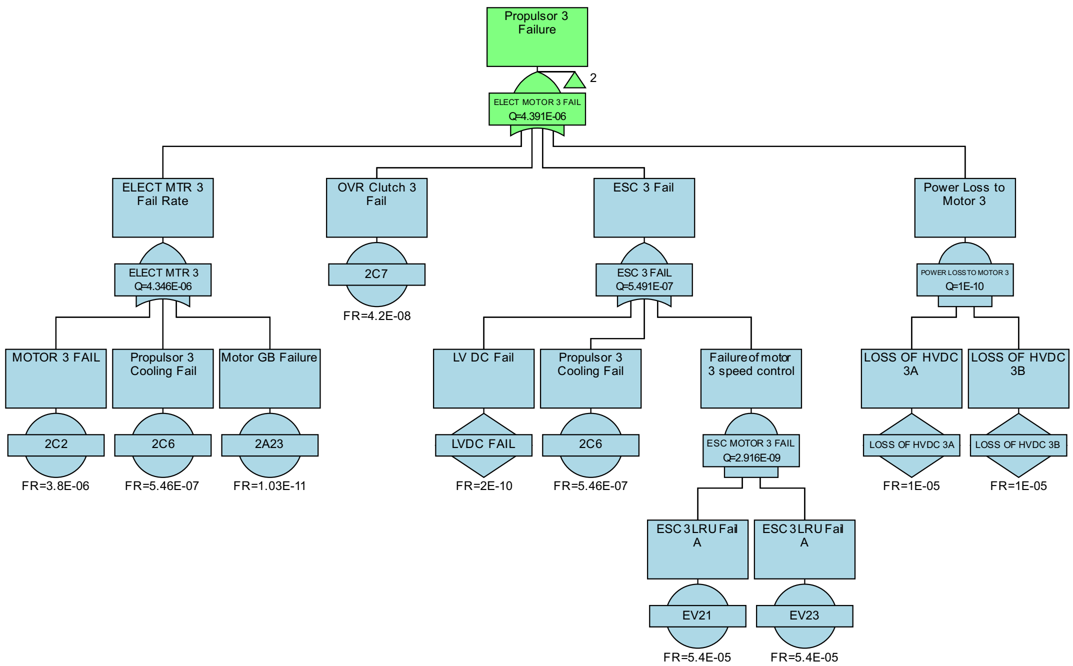
Project Diagrams

Table E4: eQuad without Cross-Shafts Fault Tree Diagram



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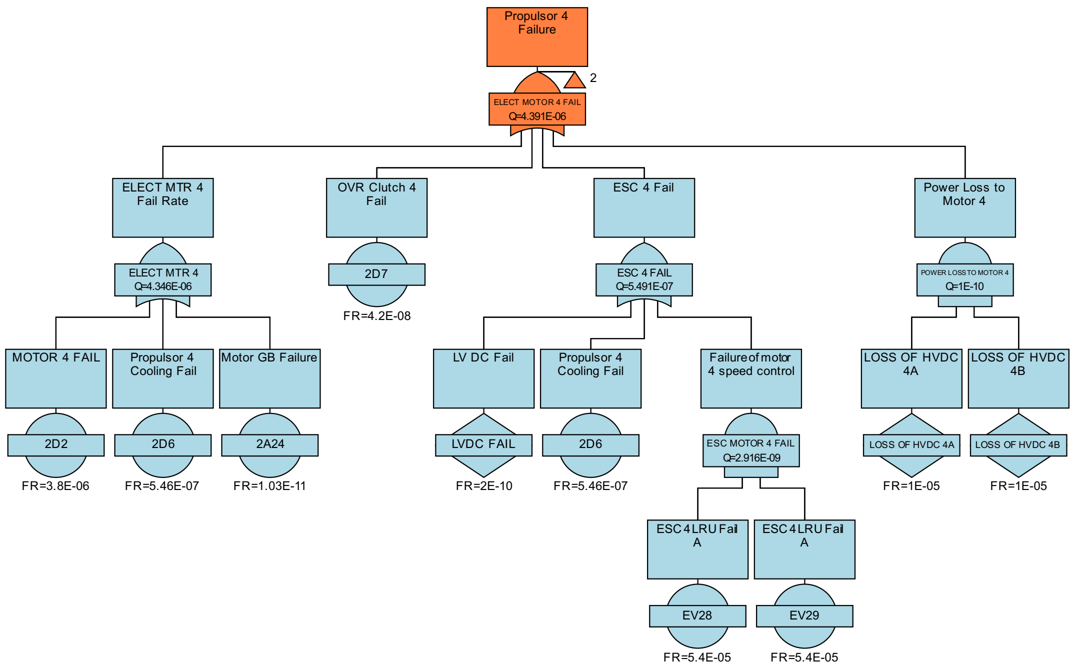
Project Diagrams

Table E4: eQuad without Cross-Shafts Fault Tree Diagram



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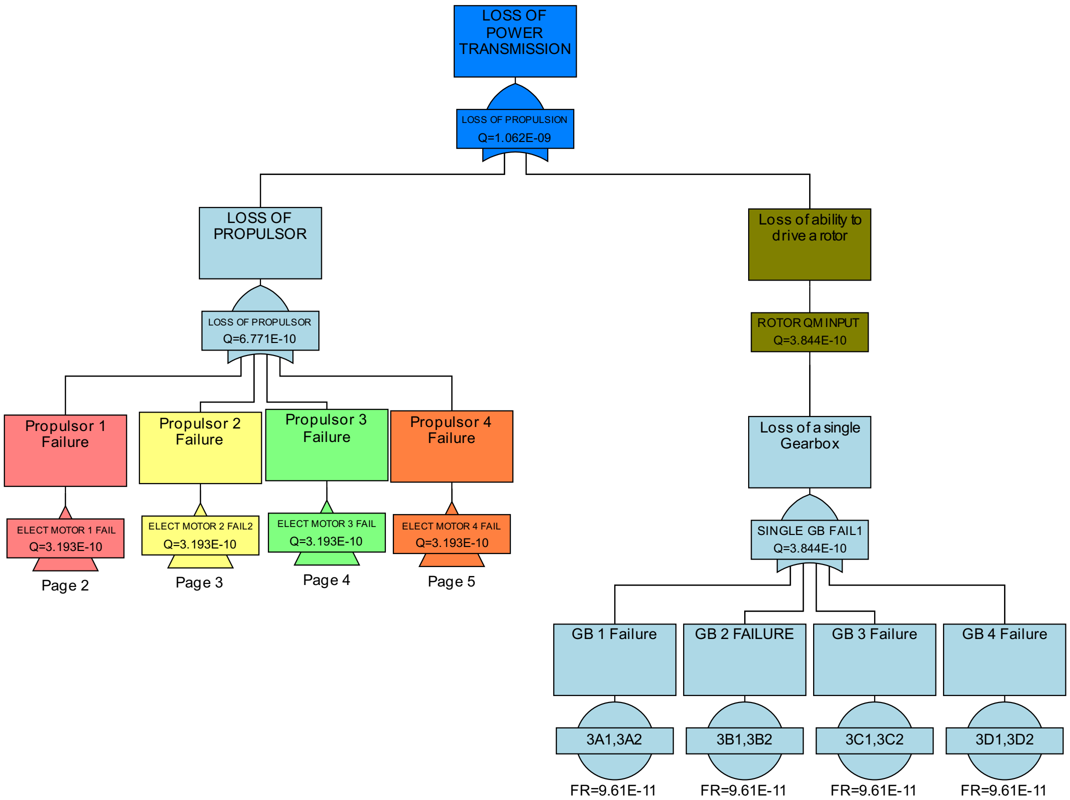
Project Diagrams

Table E5: eQuad without Cross-Shafts and Dual Redundant Propulsors Fault Tree Diagram



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Project Diagrams

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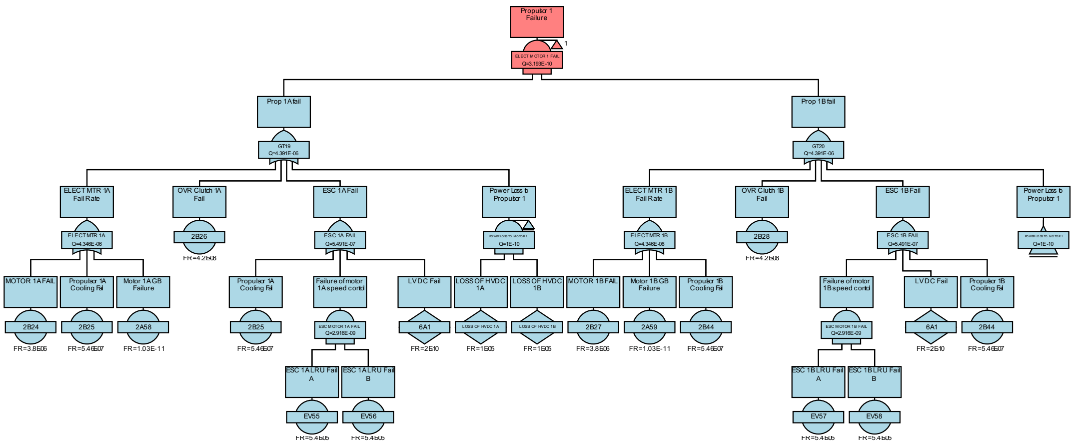

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Project Diagrams

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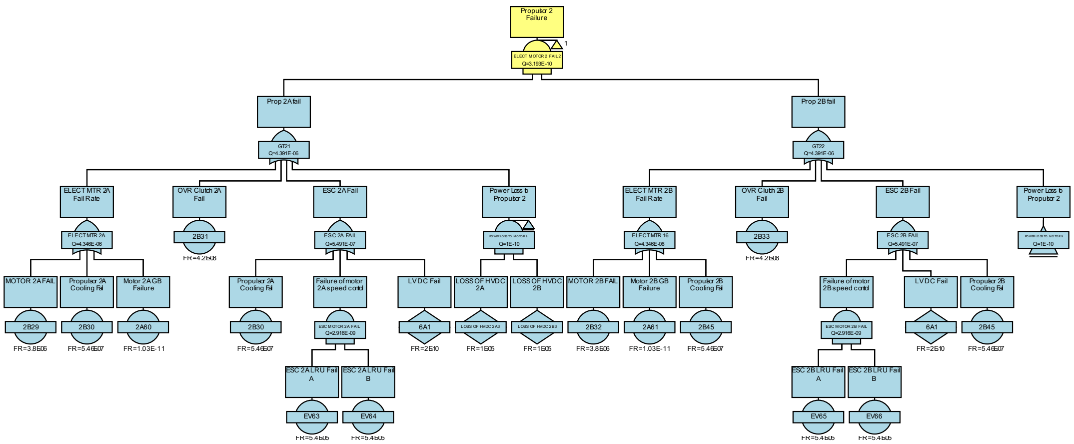

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Project Diagrams

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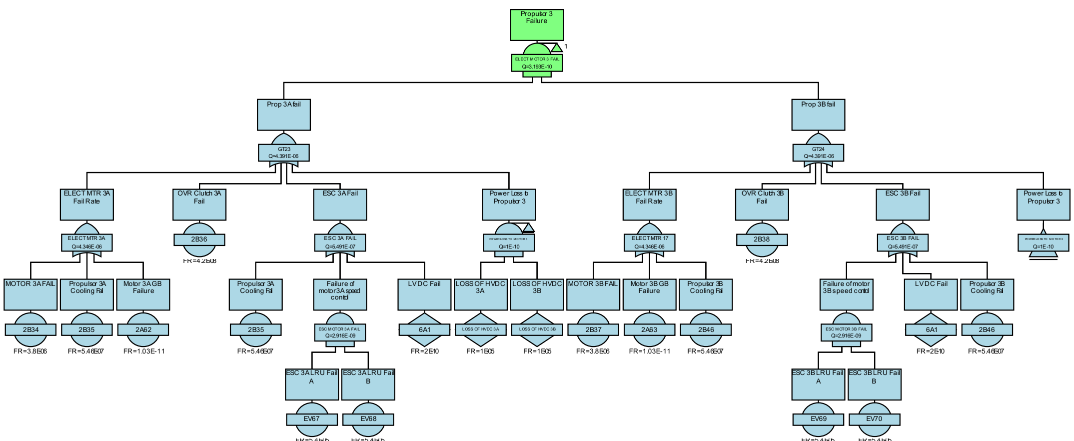

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Project Diagrams

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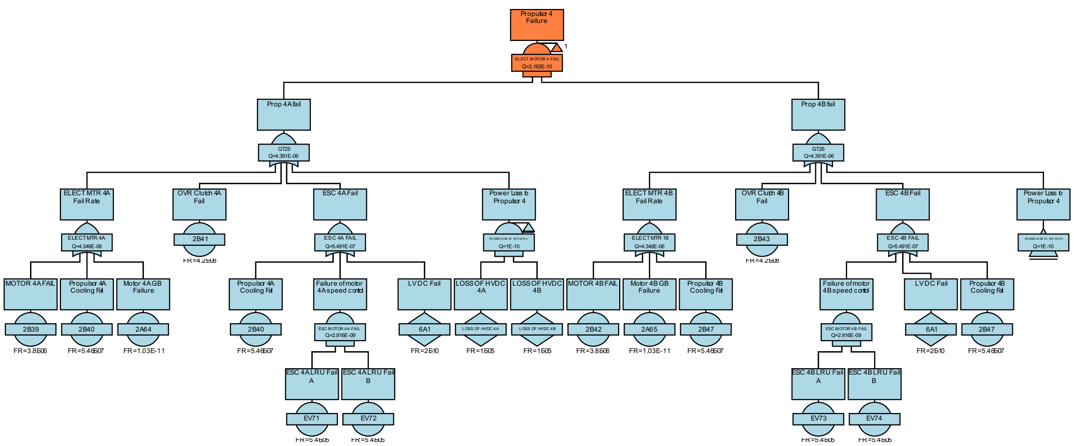

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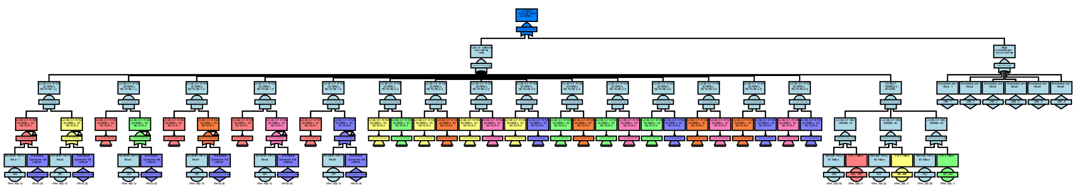
Project Diagrams

Table E6: Pitch-Controlled eHex Fault Tree Diagram



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Project Diagrams

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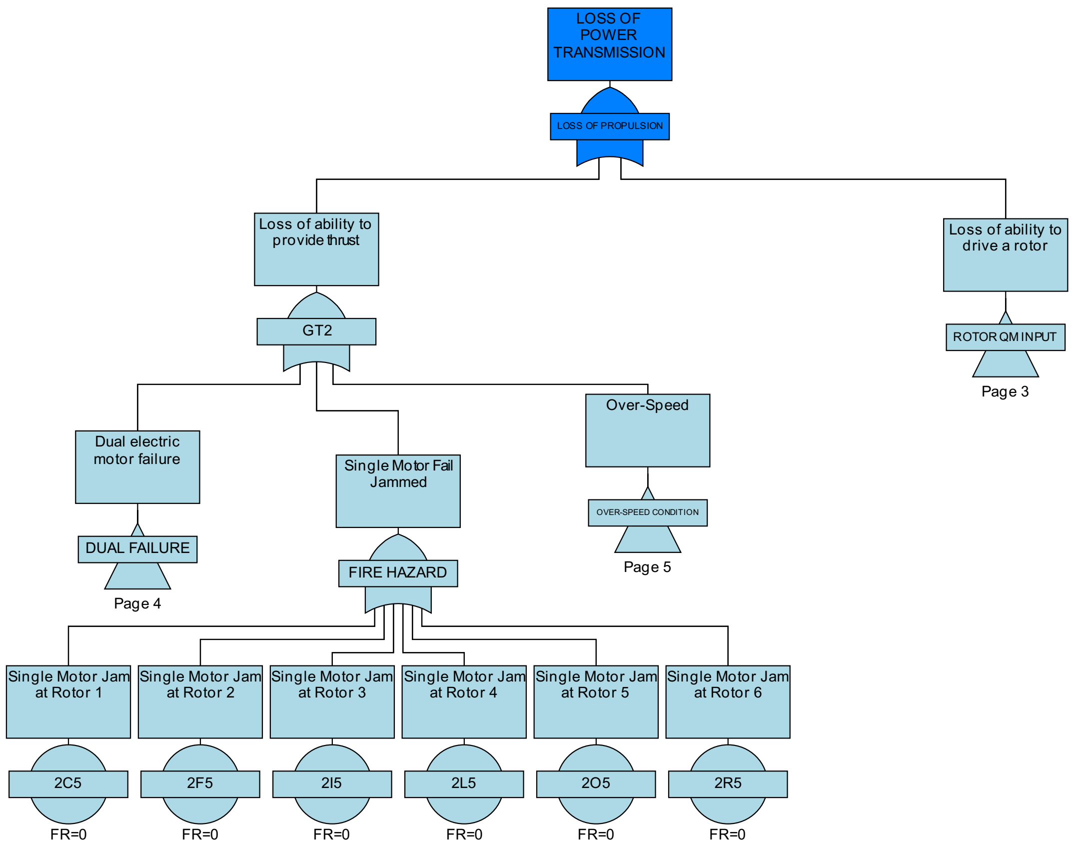

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Project Diagrams

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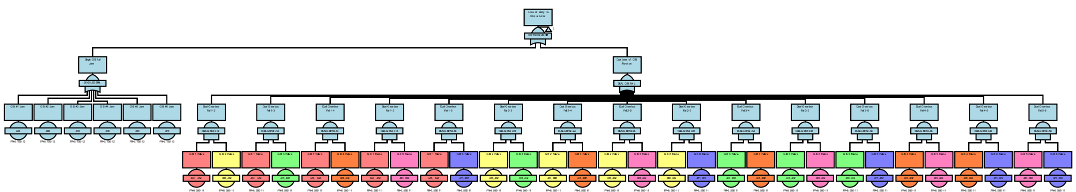

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Project Diagrams

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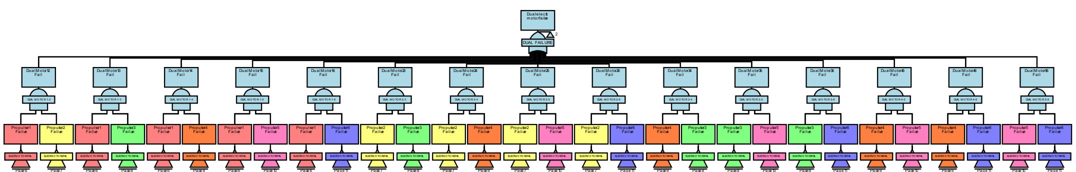

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Project Diagrams

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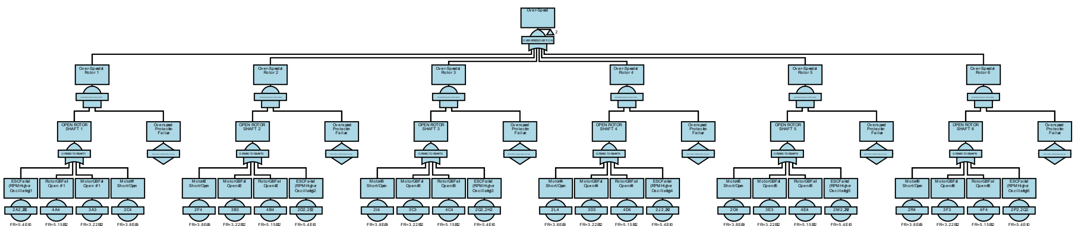

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Project Diagrams

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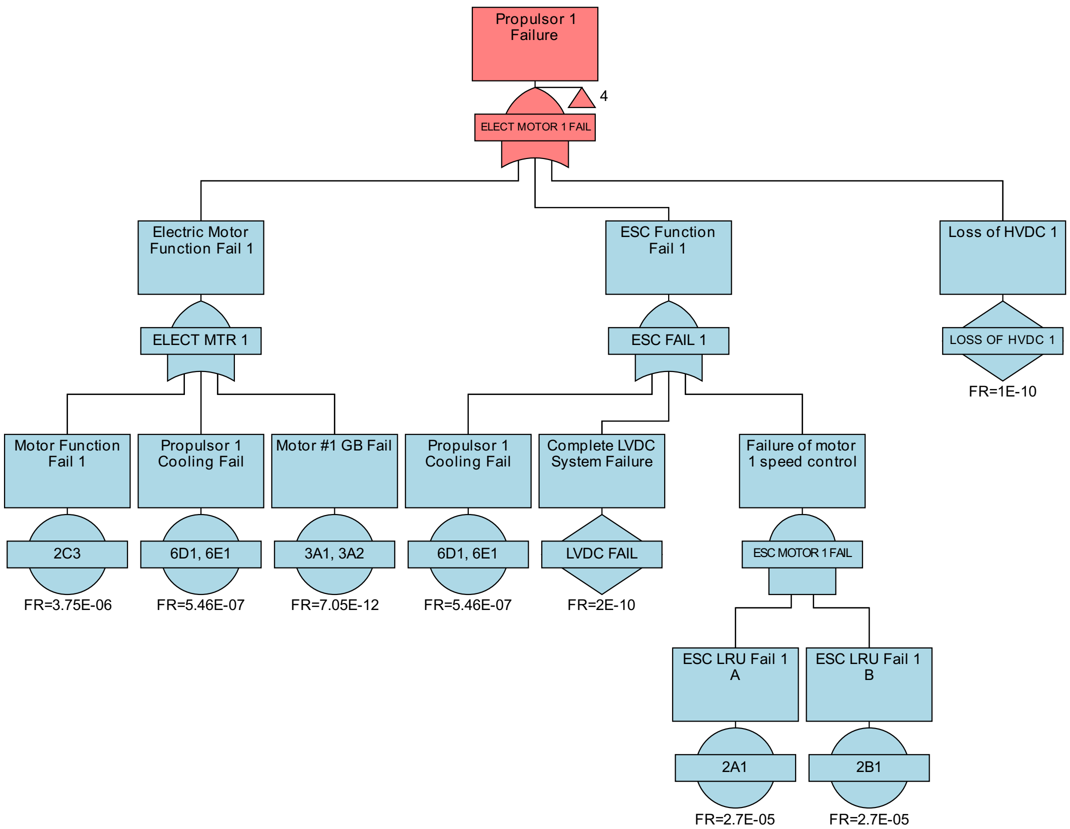

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Project Diagrams

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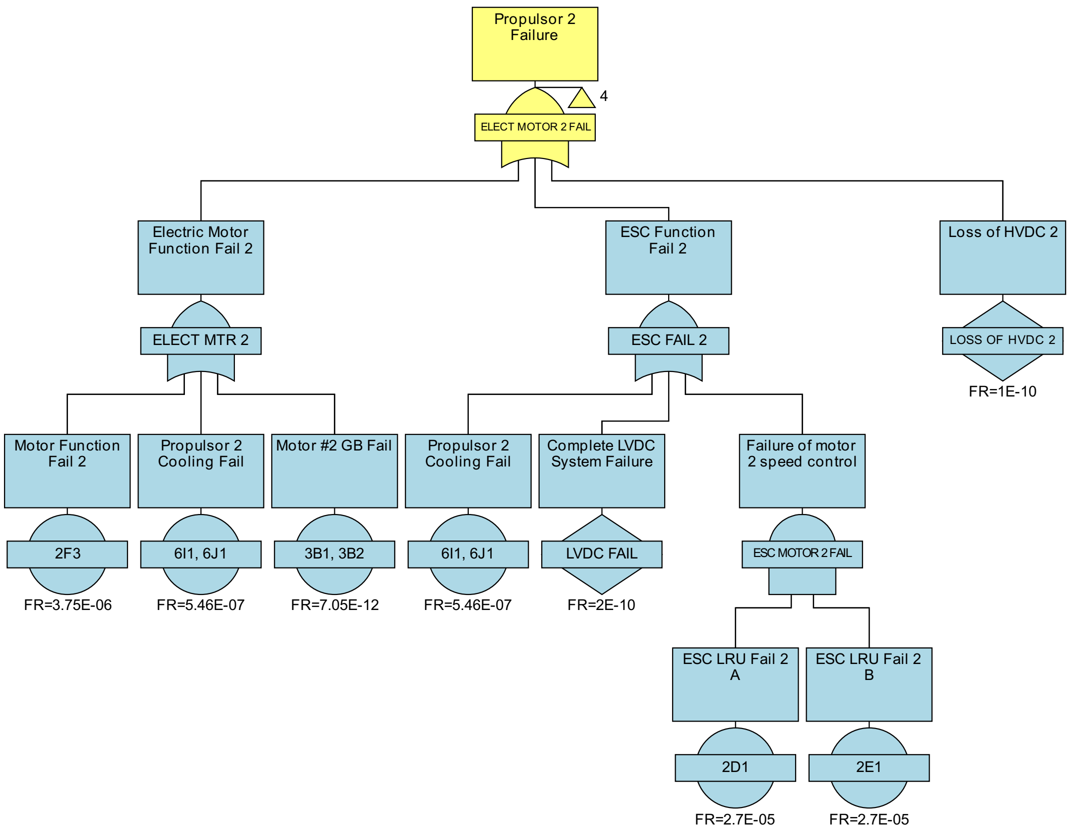

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Project Diagrams

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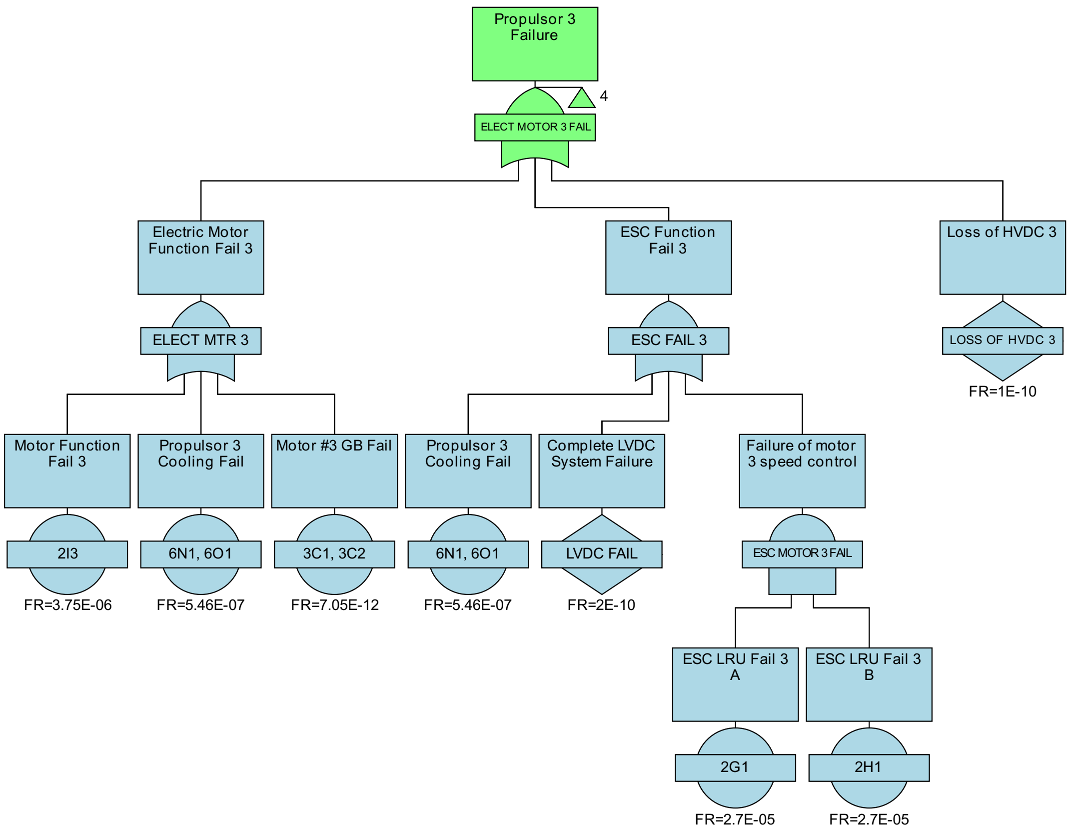

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Project Diagrams

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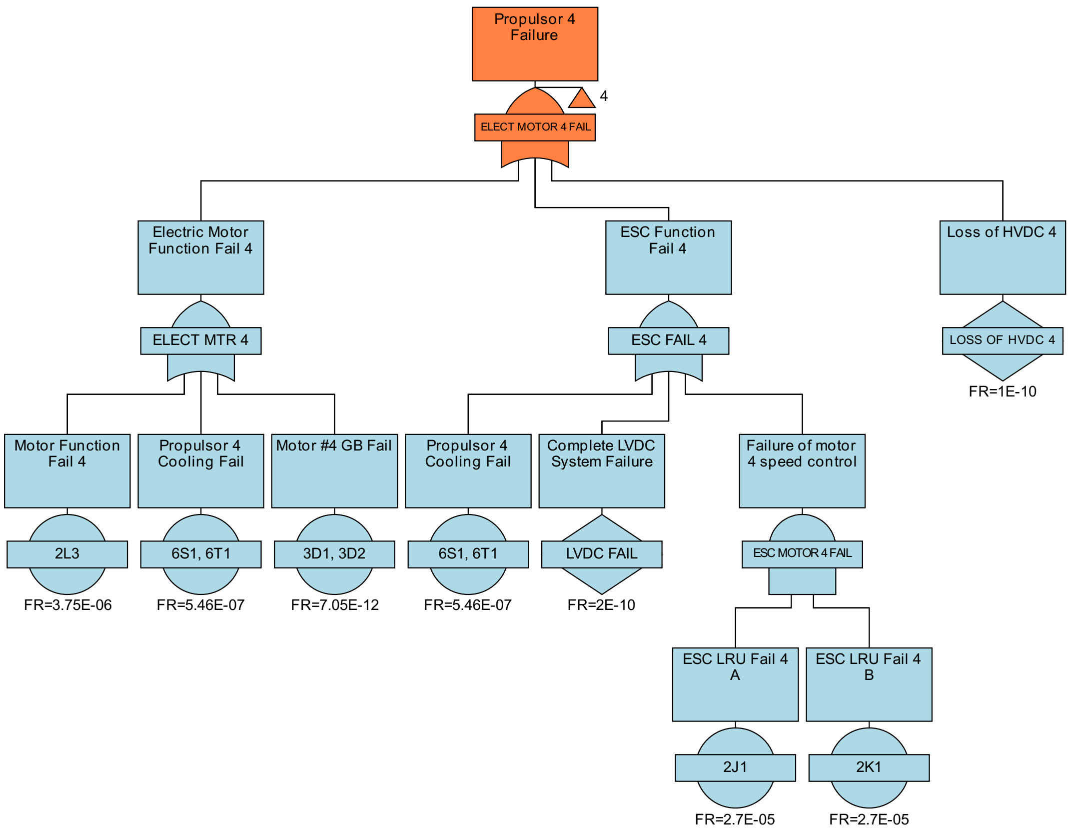

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Project Diagrams

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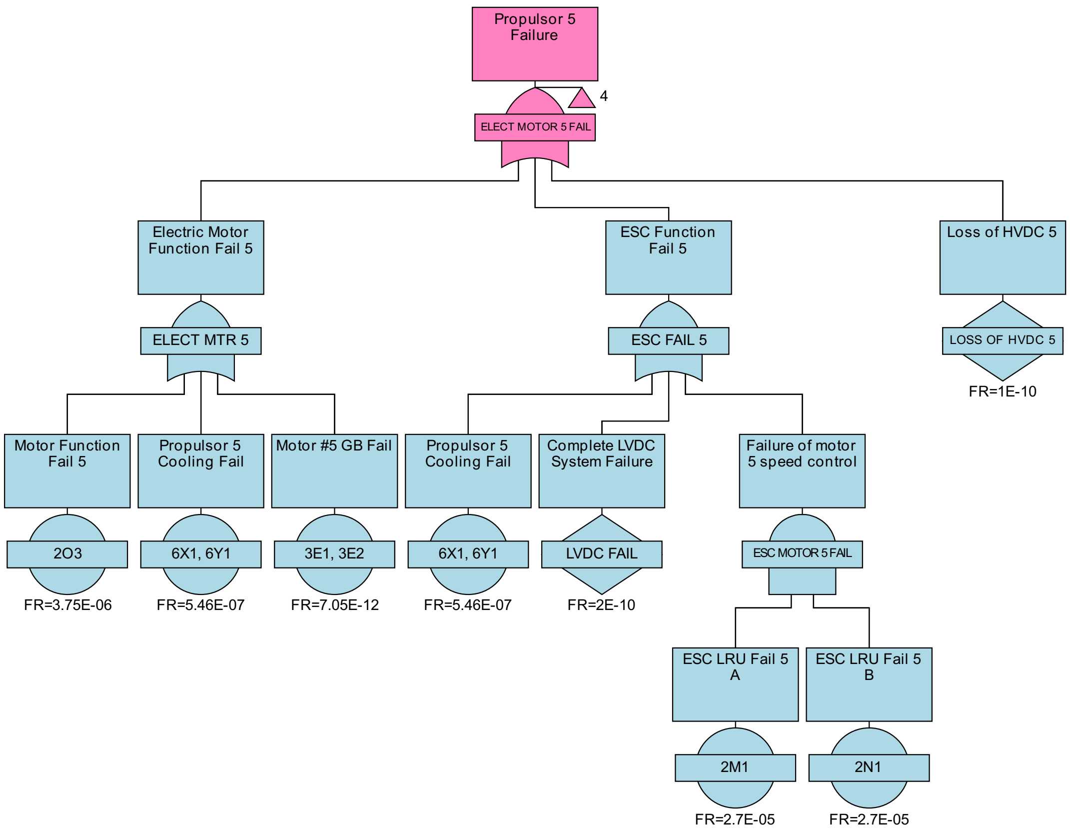

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Project Diagrams

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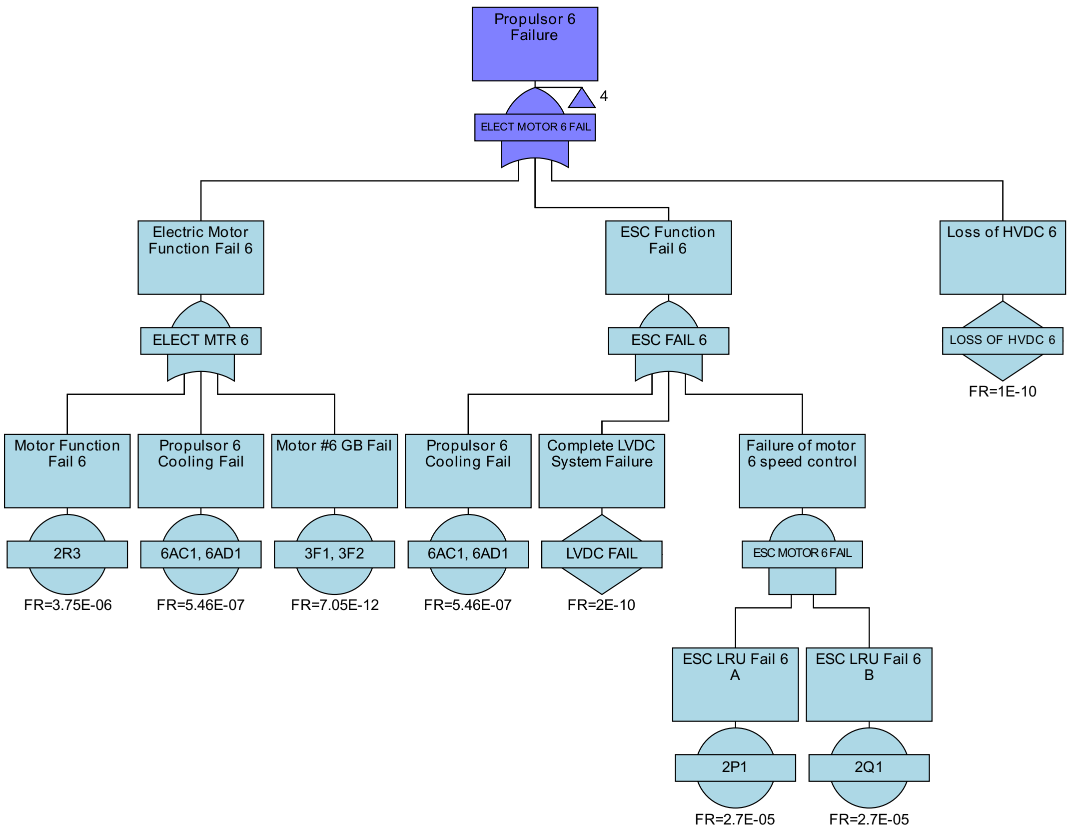

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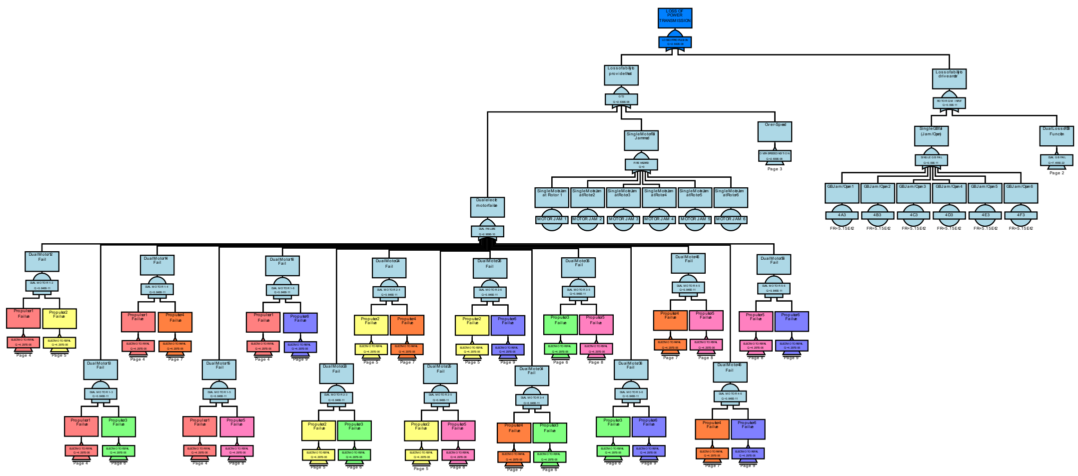
Project Diagrams

Table E7: RPM-Controlled eHex Fault Tree Diagram



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Project Diagrams

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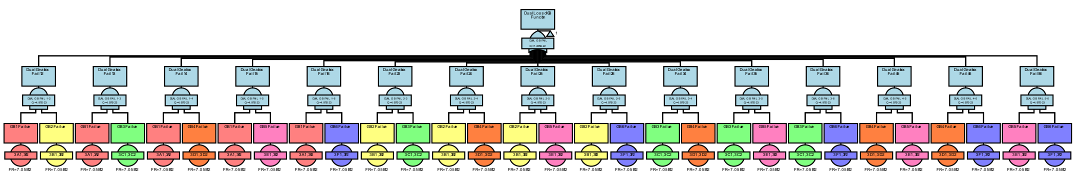

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Project Diagrams

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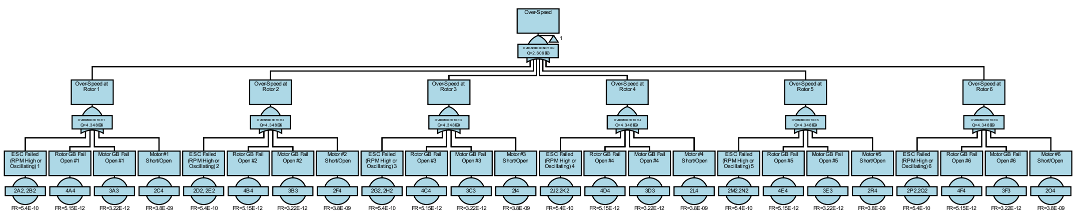

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Project Diagrams

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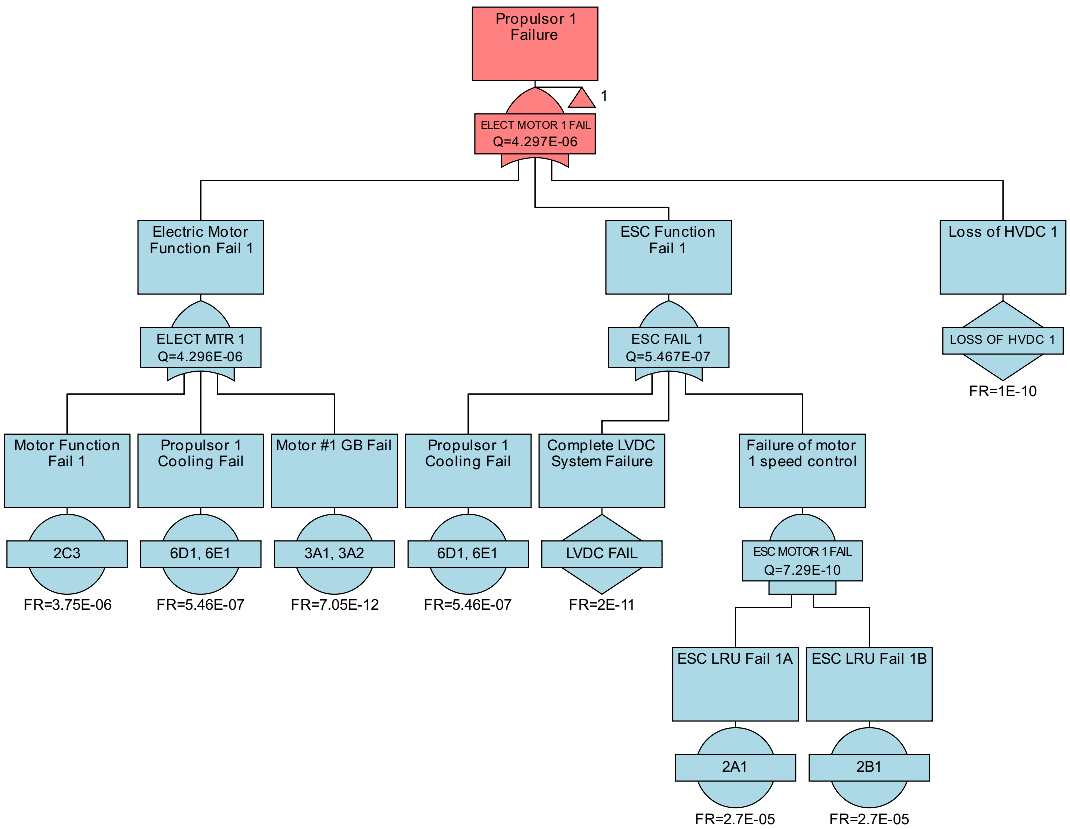

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Project Diagrams

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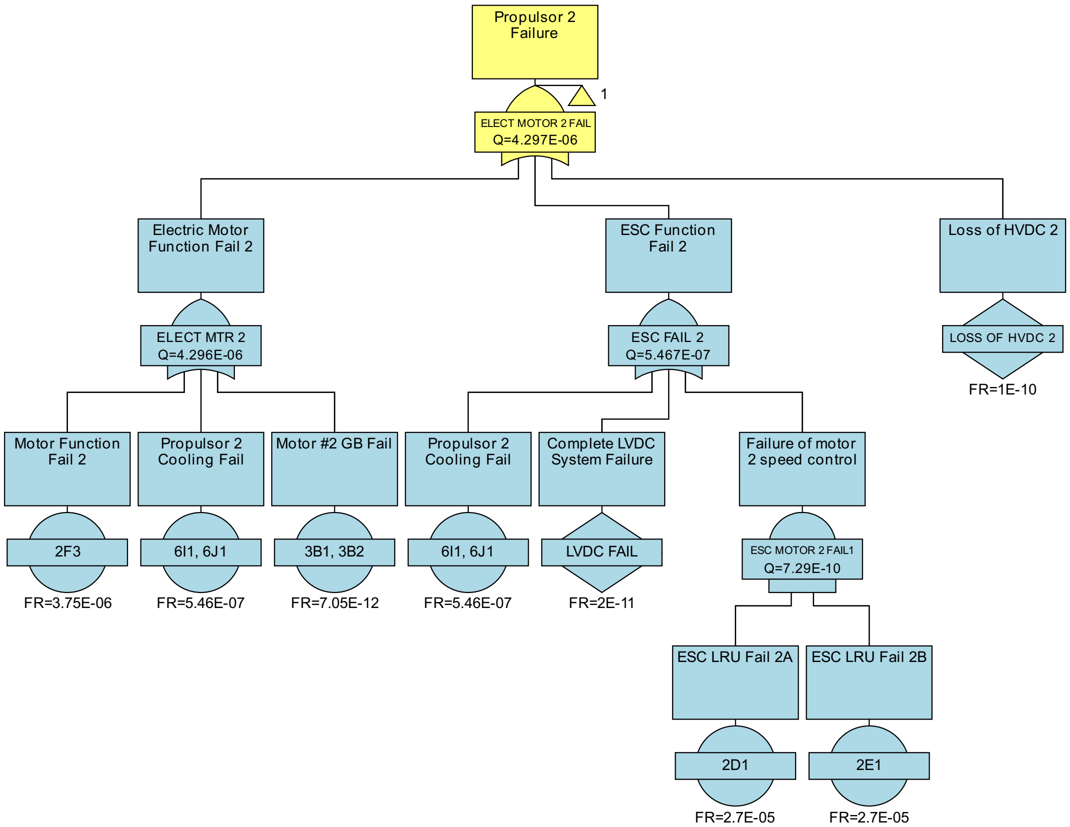

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Project Diagrams

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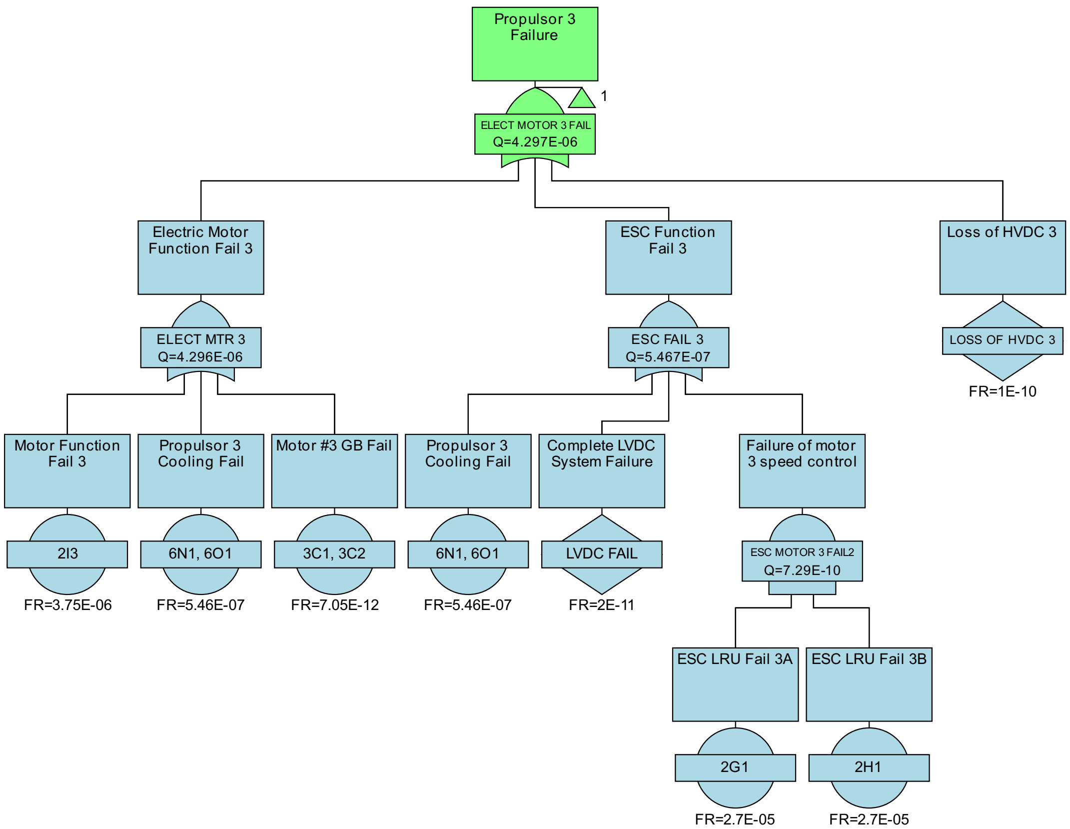

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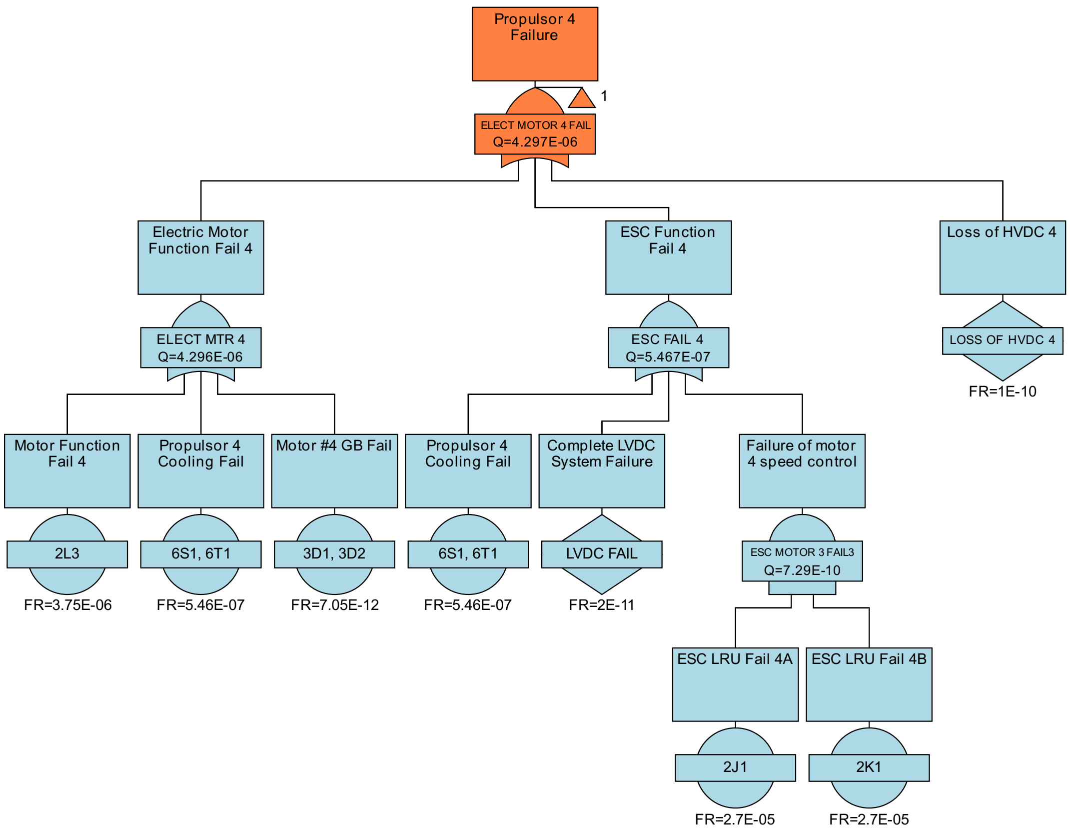
Project Diagrams

Table E7: RPM-Controlled eHex Fault Tree Diagram



RWB V12.1

Project Diagrams

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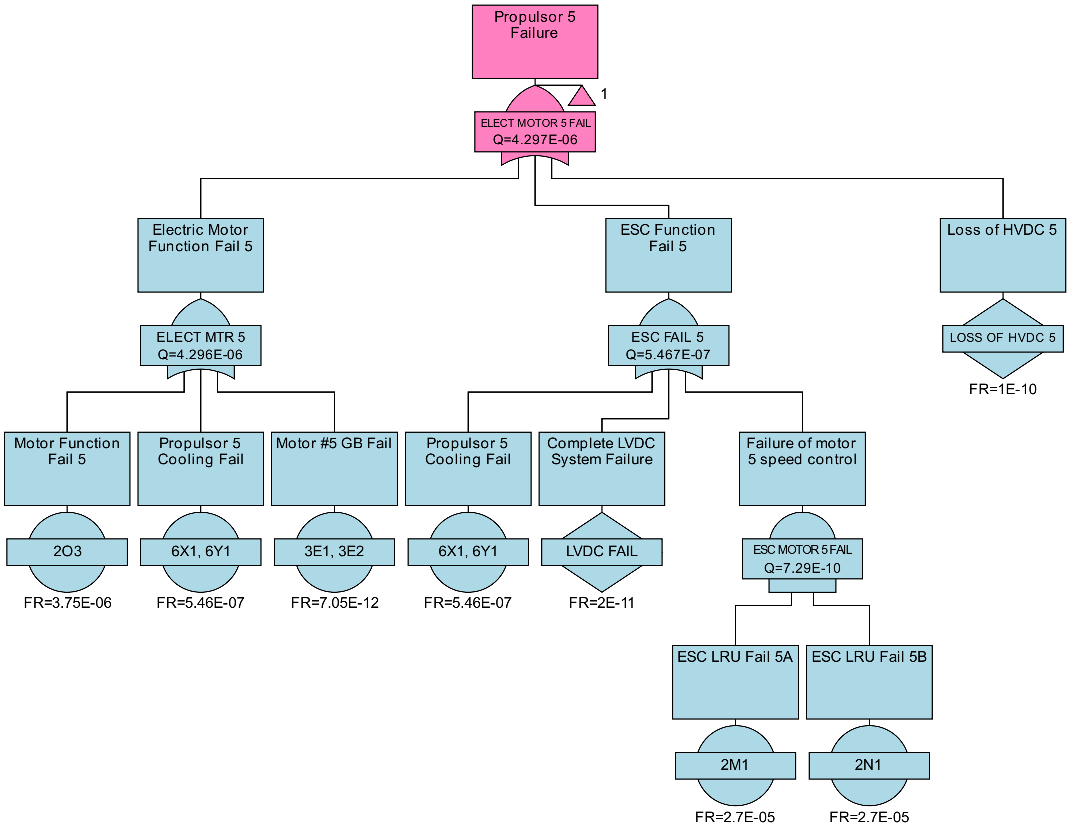

RWB V12.1

Project Diagrams

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