

Generalized Engine Inoperative Performance Requirements for Future Aircraft Concepts

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Abstract

Future transport aircraft concepts leveraging advanced propulsion systems enable more efficient flight, but face major certification barriers prior to entering service. One critical barrier is the poor applicability of “one engine inoperative” (OEI) performance requirements defined in FAA Part 25 and EASA CS-25 regulations that only apply to transport aircraft designed with 2-4 homogeneously sized gas turbine engines, making direct application to heterogeneous and distributed propulsion systems ambiguous. This work constructs generalized OEI performance requirements anchored to existing regulations and are parameterized by the specific excess power lost during a propulsion system failure, maintaining broad applicability to any aircraft and propulsion system architecture. Under the same parameterization, an “engine inoperative correction factor” is also constructed to scale the propulsion system size, enabling seamless integration into the aircraft sizing process. Both contributions are utilized to size a notional Elysian E9X, revealing system-level performance penalties for configurations sized under 4-, 5-, and 6-engine inoperative failure modes. The power loading decreases by up to 28%, thus increasing the aircraft’s maximum takeoff weight and battery energy stored onboard by up to 17% and 19%, respectively. Additional operational constraints prevent attaining system-level benefits for less severe propulsion system failure modes.

Keywords: Certification-Based Design, Engine Inoperative Conditions, Distributed Propulsion, Advanced Aircraft Design

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1. Introduction: A Need for Generalized Flight Performance Requirements

Future aircraft concepts leveraging electrified [1, 2] and distributed [3, 4, 5] propulsion systems enable more efficient and quiet flight. Prior to entering service, airframe manufacturers must prove airworthiness by demonstrating that the configuration satisfies certification regulations for either “normal” or “transport” aircraft, codified in Part 23 and 25 regulations, respectively, by the United States’ Federal Aviation Administration (FAA) [6, 7] and European Union Aviation Safety Agency (EASA) [8, 9] ¹.

Aside from establishing airworthiness, the regulations act as constraints on an aircraft’s design. For example, Fig. 1 illustrates the minimum takeoff path prescribed by FAA Part 25 regulations [10], which is divided into multiple segments with specific flight conditions and performance requirements. Airworthiness is ensured by demonstrating that the aircraft can takeoff and climb away from the runway under “one engine inoperative” (OEI) conditions upon reaching V_{EF} (engine failure speed) during the takeoff roll [11]. The FAA defines climb performance by establishing minimum climb gradients for each segment.

The climb gradients defined by the FAA are listed in Tab. 1 for 2-, 3-, and 4-engine aircraft [11], and have remained unchanged since the 1960s [12]. As defined by the Part 25.119 and 25.121d regulations, additional climb performance requirements are imposed during landing for “all engines operative” (AEO) and OEI conditions, respectively, ensuring that the aircraft can safely climb if a missed approach occurs.

The OEI requirements in existing FAA regulations implicitly assume that a 2-, 3-, or 4-engine aircraft powered by homogeneously-sized gas turbine engines is being certified. As a result, advanced concepts and propulsion systems are neglected, particularly distributed propulsion systems that utilize more than four propulsors to generate thrust, such as NASA’s X-57 Maxwell (also known as SCEPTOR) [13] and SUBsonic Single Aft eNginE (SUSAN) [14] concepts. Furthermore, these concepts utilize disparately sized propulsors, contradicting another implicit assumption in the existing

¹Although FAA Part 23 and 25 regulations are broadly harmonized with EASA CS-23 and CS-25 regulations, respectively, differences exist and certification is authority-specific. The present work uses the FAA Part 25 climb gradient structure as a reference baseline for performance requirement generalization, and refers to them as “FAA regulations.”

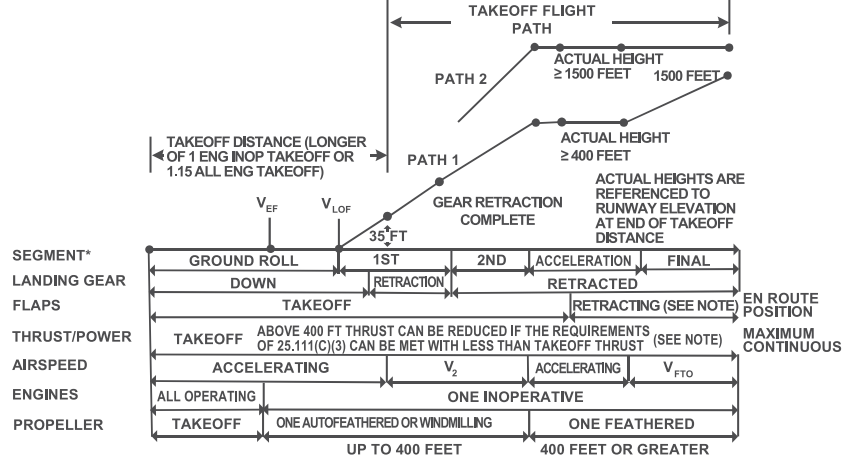


Figure 1: Takeoff segments prescribed by the FAA [10].

Table 1: Climb performance requirements in FAA Part 25 regulations.

Requirement	2-Engine	3-Engine	4-Engine
25.111 (Takeoff Flight Path, OEI)	$\gamma \geq 1.2\%$	$\gamma \geq 1.5\%$	$\gamma \geq 1.7\%$
25.119 (Landing, AEO)	$\gamma \geq 3.2\%$	$\gamma \geq 3.2\%$	$\gamma \geq 3.2\%$
25.121a (First Segment Climb, OEI)	$\gamma \geq 0.0\%$	$\gamma \geq 0.3\%$	$\gamma \geq 0.5\%$
25.121b (Second Segment Climb, OEI)	$\gamma \geq 2.4\%$	$\gamma \geq 2.7\%$	$\gamma \geq 3.0\%$
25.121c (Final Climb, OEI)	$\gamma \geq 1.2\%$	$\gamma \geq 1.5\%$	$\gamma \geq 1.7\%$
25.121d (Landing, OEI)	$\gamma \geq 2.1\%$	$\gamma \geq 2.4\%$	$\gamma \geq 2.7\%$

FAA regulations that all propulsors are homogeneously sized. A regulatory gap analysis [15] has already identified that existing OEI requirements lack applicability for distributed propulsion concepts, suggesting a broader need to redefine OEI scenarios and nominal/off-nominal operating conditions.

Inspired by rendering distributed propulsion concepts airworthy, design and certification of these configurations has become an emergent research area, with a particular focus on engine inoperative conditions.

1.1. Literature Review

Prior research has focused on identifying the critical engine in an advanced propulsion system, modifying existing OEI conditions, and designing advanced concepts with distributed propulsion systems using

current regulations.

Under OEI conditions, the “critical engine” is assumed to fail [11]. Prior researchers recommend identifying the critical engine as the one that: 1) produces the largest thrust asymmetry upon failing; 2) provides the largest amount of power at low altitudes; or 3) powers other propulsors in the propulsion system aside from itself [16]. For distributed propulsion systems, the propulsor closest to the wingtip generates the largest yaw moment, rendering it as the critical engine [17, 18, 19]. However, on NASA’s SUSAN concept [14], minimum propulsive power requirements are defined for failures of the aft turbofan engine *and* distributed propulsion system during takeoff and cruise [20]. Although the aft turbofan engine produces the most power, the most outboard propulsor in the distributed electric propulsion (DEP) system generates the largest yaw moment upon failure. Additional exploration is needed to identify SUSAN’s critical engine for certification.

Aside from selecting the critical engine, there have also been proposals to modify OEI conditions for future aircraft concepts. One approach involved designating larger, discrete propulsors as a “primary” propulsion system while considering the distributed propulsion system as a “secondary” one [21]. However, this fails to accommodate concepts that use distributed propulsion systems as their sole propulsive power source. Another approach assumed that any component failure in the propulsion system represents an engine inoperative condition [22, 23], capturing failure modes that could result in a loss of multiple propulsors. In this case, however, there may not be a “critical engine,” but rather a “critical loss of thrust” instead [24], involving multiple propulsors within one distributed propulsor array failing [25].

Since modified OEI regulations have not yet been adopted, two approaches were previously utilized to define climb requirements for distributed propulsion concepts. One option is to assume that the required climb gradients would match those for a 4-engine conventional aircraft [23, 26, 27], reflecting the most stringent requirements in FAA Part 25 regulations. Another option pursued involved developing an extrapolation function dependent on the number of propulsors [28]. However, the extrapolation function developed requires a steeper climb gradient as more engines are added, and exceeds those defined within the FAA regulations. Additionally, climb performance in an aircraft with a distributed propulsion system degrades more gracefully as a result of many, smaller propulsors installed on the aircraft [29], possibly resulting in better off-nominal performance than their conventional counterparts.

OEI conditions are also considered during aircraft sizing to ensure that the propulsion system provides sufficient power during an engine failure. Traditionally, each engine’s size is scaled by a correction factor of $n/(n - 1)$, where n is the number of engines installed on the aircraft [30]. Examples of these scale factors for 2-, 3-, and 4-engine configurations are listed in Tab. 2, requiring each propulsor to be oversized by 33 to 100%. However, the correction factors listed in Tab. 2 again assume that the engines are homogeneously sized, rendering them invalid for advanced propulsion system architectures with disparately sized propulsors [14, 31].

Table 2: Engine inoperative correction factors for conventional aircraft configurations.

Number of Engines	Correction Factor
2	2.00
3	1.50
4	1.33

1.2. Gaps in the Literature

The literature review exposed three critical gaps for designing and certifying advanced aircraft concepts with distributed propulsion systems. First, it is uncertain how OEI scenarios may be defined for distributed propulsion concepts, and may encompass multiple propulsor failures. Flexible design methods are needed to explore multiple propulsion system failure modes and assess their implications on the resultant aircraft design.

Second, existing OEI performance requirements have not changed in over 60 years and fail to accommodate future aircraft concepts leveraging distributed propulsion systems. These requirements are defined as a function of the number of engines installed (a discrete variable) and implicitly assume that all propulsors are homogeneously sized, prohibiting their broad application to future aircraft concepts with possibly heterogeneously-sized propulsion systems. Prior studies designing aircraft with distributed propulsion systems utilize the published climb gradients from FAA Part 25 regulations for 4-engine aircraft or develop extrapolation functions based on the number of propulsors installed. The former approach utilizes the climb performance requirements defined within the bounds of existing FAA requirements by imposing the most stringent performance requirements

available, but neglects the failure mode severity. The latter approach prescribes a larger climb gradient as the number of propulsors installed on the aircraft increases, expanding existing parameterizations for an aircraft with n propulsors, but requires the aircraft to perform more aggressively relative to conventional aircraft previously certified under FAA Part 25 regulations. The strengths and shortcomings of both approaches could be combined to develop engine inoperative performance requirements in a configuration-agnostic manner while anchoring it within the bounds of the FAA’s published climb performance requirements, enabling all transport aircraft to be certified under a unified set of regulations.

Third, novel approaches are needed to scale a propulsion system’s size for engine inoperative conditions. Previous methods assume each propulsor is equally sized, resulting in a correction factor that depends on the number of engines installed. However, such an approach is not applicable to future aircraft concepts utilizing heterogeneously-sized propulsors. Again, configuration-agnostic methods are needed to size future aircraft concepts and their propulsion systems, independent of the number of propulsors installed on the airframe.

2. Objective: Generalized Engine Inoperative Performance Requirements

This research proposes generalizing existing OEI performance requirements by parameterizing them as a function of the specific excess power lost during a propulsion system failure (a continuous variable), overcoming limitations on their current definition as a function of the number of propulsors installed (a discrete variable). To support sizing conventional and advanced propulsion systems under OEI conditions, an “engine inoperative correction factor” is also proposed under the same parameterization, scaling the propulsion system size to satisfy the necessary flight performance requirements during such conditions. The proposed method enhances the state-of-the-art by enabling the consistent sizing of conventional and electrified transport concepts with different propulsion system architectures during conceptual design while accounting for certification-relevant constraints, reducing the need for ad-hoc OEI requirement adaptations.

Three key contributions are made in this paper:

1. A fleet-calibrated re-parameterization of certification requirements by the specific excess power loss, mapping the propulsion system failure severity to an FAA flight performance requirement.
2. Construction of generalized climb performance requirements utilizing the mapping between the specific excess power loss and FAA flight performance requirements.
3. Development of a generalized engine inoperative correction factor utilizing a mapping between the specific excess power loss and traditional $n/(n - 1)$ engine inoperative correction factors.

The contributions produced in this paper are intended to be used during conceptual aircraft design, primarily for predicting system-level performance. More detailed component sizing studies, such as the impacts of OEI conditions on control surface sizing, are rendered out-of-scope, as well as other engine design and maintenance certification gaps [32, 33].

The remainder of the paper is structured as follows. First, Section 3 utilizes flight simulations to estimate the specific excess power losses under OEI conditions among previously certified aircraft, providing a basis for re-parameterizing the certification requirements. Then, Section 4 constructs the generalized OEI performance requirements and an expression defining the engine inoperative correction factor, both as functions of the specific excess power lost under OEI conditions. All constructed expressions are applied to an aircraft sizing study in Section 5, revealing the system-level impacts of sizing an aircraft under the proposed generalized requirements relative to those sized under the existing FAA Part 25 regulations. Lastly, Section 6 concludes the paper and provides areas for future study.

3. Aircraft Operational Analysis

This section describes the aircraft operations simulations performed to demonstrate the validity of parameterizing the generalized climb performance requirements by the specific excess power lost during an engine failure. First, Section 3.1 describes the mission profile flown and operational conditions examined. Then, Section 3.2 reveals the relationship between the specific excess power lost during an engine failure and the flight performance degradation observed. Lastly, Section 3.3 establishes first-order

approximations from the data in Section 3.2 to mathematically demonstrate that an aircraft’s specific excess power losses exhibit weak sensitivity with respect to its operational strategy, further emphasizing the broad applicability of the estimations established in this work.

3.1. Aircraft Operations Simulations

Aircraft operations simulations were performed on 156 conventional aircraft concepts previously certified under the FAA’s Part 25 (transport aircraft) regulations. The aircraft examined span regional, narrowbody, and widebody configurations from major airframe manufacturers that serve regional, domestic, and international markets. Table 3 lists the aircraft configurations and number of variants accounted for during the analysis ². Across the 156 aircraft analyzed, 118 aircraft had two engines installed, 4 aircraft had three engines installed, and 34 aircraft had four engines installed.

The aircraft were modeled in the Future Aircraft Sizing Tool (FAST, described next)[34], leveraging data from its Aerobase [35], a publicly-available database with over 400 aircraft previously certified by the FAA or EASA. The Aerobase primarily contains system-level performance parameters and weights such as maximum takeoff weight, fuel weight, and cruise speed. All data to model the 156 aircraft in this work are also available in a separate publicly-available data repository [36] to facilitate result replication and further testing.

3.1.1. FAST: Aircraft Analysis Framework

The aircraft operations simulations were performed in FAST [34], an open-source, MATLAB-based aircraft design tool for early-phase conceptual design. FAST utilizes an energy-based mission analysis and assumes that the aircraft is a point mass, enabling a rapid analysis to accommodate the 156 aircraft considered in this study. Its graph-based propulsion analysis framework [37] and segment-based power management strategy definition allow individual engines to be powered on/off, thus enabling the engine inoperative simulations necessary for this work. Additionally, its detailed aerodynamic model leverages empirical relations from an aircraft’s geometry to predict the parasite and induced drag coefficients, and is an adaptation of an empirical drag polar model from Aviary, an open-source aircraft sizing

²The row for the Airbus A320 includes Airbus A318, A319, and A321 variants, as well as other “new engine option” (NEO) variants.

Table 3: Aircraft configurations and variants considered during the operational analysis.

Aircraft	Number of Variants
Airbus A300	15
Airbus A310	10
Airbus A320	46
Airbus A380	3
Boeing 707	1
Boeing 717	2
Boeing 737	14
Boeing 747	30
Boeing 757	3
Boeing 767	7
Boeing 777	4
Boeing 787	3
Embraer E135	2
Embraer E145	2
Embraer E170	4
Embraer E190	6
Lockheed Martin L-1011	4

tool for preliminary design developed by NASA [38]. Such capabilities were augmented in this paper by adding windmilling and trim drag contributions to the parasite drag coefficient (discussed in Section 3.1.3 and Appendix C).

3.1.2. Mission Profile

Each aircraft was flown twice on a 370 km diversion mission illustrated in Fig. 2 – once under AEO conditions and once under OEI conditions, failing one engine after takeoff. During an engine failure, the aircraft will likely not continue to its destination airport and divert to a nearby one, thus justifying the shorter mission profile.

The takeoff and climb speeds, V_{Tko} and V_{Cib} , respectively, were varied to represent multiple operational strategies during departure. The climb speed was computed by incrementing the takeoff speed with a fixed value, ΔV . The takeoff speed and increments are listed in Tab. 4. Since the maximum allowable climb speed under 3,048 m is approximately 128.6 m/s in the United States [39], the maximum takeoff speed and speed increment

were set to 102.89 and 25.72 m/s, respectively, obeying the operational constraints imposed on any aircraft flying in the National Airspace System. Additional boundary conditions for each mission profile segment are provided in Appendix A.

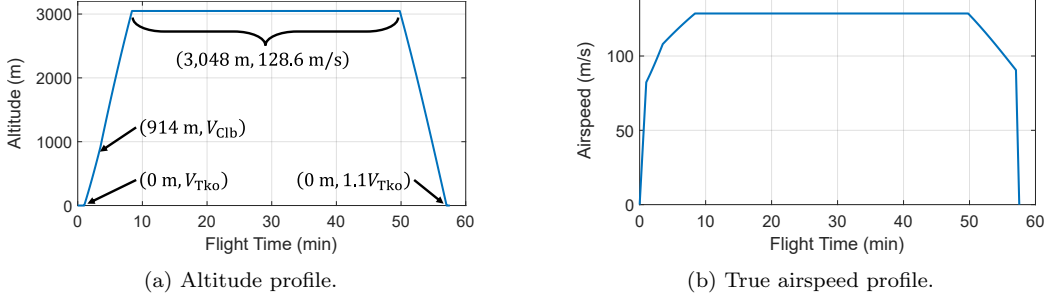


Figure 2: Mission profile for fleet-level OEI condition simulations.

Table 4: Takeoff speeds and increments (both in m/s) simulated under OEI conditions. Note that $V_{Clb} = V_{Tko} + \Delta V$.

Speed	Lower Bound	Upper Bound	Step Size
V_{Tko}	66.88	102.89	5.14
ΔV	0.00	25.72	5.14

3.1.3. Aerodynamic Correction for Engine Inoperative Conditions

Due to the lack of geometric data available for all 156 aircraft analyzed, FAST’s simpler aerodynamic model – assuming a constant lift-to-drag ratio during each flight segment – was instead utilized. To scale the constant lift-to-drag ratios for OEI conditions, the AEO lift-to-drag ratio was multiplied by a correction factor, and is dependent upon the flight conditions simulated and number of engines installed on the aircraft.

The correction factors were obtained by performing a detailed aerodynamic analysis on six notional aircraft models – Airbus A300, Airbus A310, Airbus A320, Airbus A380, Boeing 777, and Embraer E175 – in FAST under both AEO and OEI conditions. For these six configurations only, FAST’s detailed aerodynamic model was utilized. The geometric data necessary to utilize the drag polar model was extracted from publicly

available CAD files [40, 41], and have also been included in an open-source dataset enabling readers to reproduce the results obtained in this work [36].

As previously introduced, windmilling and trim drag were accounted for by penalizing the parasite drag coefficient. Windmilling drag was modeled from Walsh and Fletcher’s *Gas Turbine Performance* text [42] for windmilling turbofan engines (Chart 10.5, free windmilling curve), and parameterizes the drag increment by the engine’s specific thrust. Trim drag was predicted using an elementary lateral stability model, as detailed in Appendix C.

After simulating the models under AEO and OEI conditions, the lift-to-drag fraction, defined as $(L/D_{\text{OEI}}) / (L/D_{\text{AEO}})$, was computed for both configurations. At each flight condition, the correction factor was estimated as the following:

1. 2-Engine Aircraft: the mean lift-to-drag fraction across the Airbus A300, Airbus A310, Airbus A320, Boeing 777, and Embraer E175 models analyzed.
2. 3-Engine Aircraft: the mean lift-to-drag fraction estimated for both 2- and 4-engine aircraft.
3. 4-Engine Aircraft: the lift-to-drag fraction from the Airbus A380 model analyzed.

All correction factors are listed in Appendix B, nominally reflecting a 20% decrease in the lift-to-drag ratio.

3.2. Predicting Flight Performance Degradation for Transport Aircraft

Flight performance degradation is quantified by recording the specific excess power, defined in Eq. 1, from both the AEO and OEI simulations. The specific excess power represents the excess power normalized by the aircraft’s weight, and can be utilized to either climb or accelerate, as indicated in Eq. 2 [43].

$$P_s = \frac{TV - DV}{W} \quad (1)$$

$$P_s = \dot{h} + \frac{V_\infty \dot{V}_\infty}{g} \quad (2)$$

While conceptualizing this study, the thrust power was also considered as a possible parameter to predict flight performance degradation. However,

the specific excess power was ultimately selected because of its ability to better quantify the system-level impacts of an engine failure, incorporating both aerodynamic (drag power) and propulsive (thrust power) considerations within one quantity.

The specific excess power lost during an engine failure was accounted for by defining the “specific excess power loss fraction” (abbreviated as the “SEPLF”) in Eq. 3. The SEPLF represents the relative difference in specific excess power quantities recorded from the AEO and OEI simulations. Since the specific excess power under OEI conditions is always less than that of AEO conditions, the SEPLF remains bounded between 0 and 1 for all aircraft configurations and operational strategies.

$$P_{s_{\text{loss}}} = \frac{P_{s_{\text{AEO}}} - P_{s_{\text{OEI}}}}{P_{s_{\text{AEO}}}} \quad (3a)$$

$$= 1 - \frac{P_{s_{\text{OEI}}}}{P_{s_{\text{AEO}}}} \quad (3b)$$

For each simulation, the minimum, median, and maximum SEPLFs were computed to identify the lower, nominal, and upper bounds of such losses. The median SEPLFs are primarily utilized for predicting flight performance degradation during OEI conditions, while the minimum/maximum ones are employed to assess the broader implications of the estimations made within this paper. For the remainder of this paper, assume that the SEPLF refers to the median SEPLF unless otherwise specified.

To map the propulsion system failure severity to the FAA climb performance requirements, the climb gradient is computed by substituting the rate of climb computed from Eq. 2 into Eq. 4 to compute the maximum achievable climb gradient (γ) for the given flight conditions and operational strategy [43].

$$\gamma = \tan \left[\arcsin \left(\dot{h}/V_{\infty} \right) \right] \quad (4)$$

Similar to the SEPLF, a climb gradient reduction fraction (abbreviated as the “CGRF”) was also estimated, as defined in Eq. 5, representing the climb performance degradation under OEI conditions relative to AEO conditions.

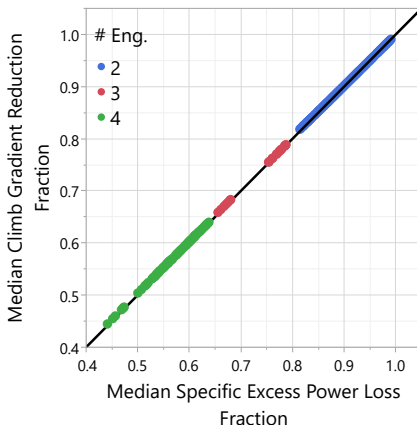
$$\gamma_{\text{loss}} = \frac{\gamma_{\text{AEO}} - \gamma_{\text{OEI}}}{\gamma_{\text{AEO}}} \quad (5a)$$

$$= 1 - \frac{\gamma_{\text{OEI}}}{\gamma_{\text{AEO}}} \quad (5b)$$

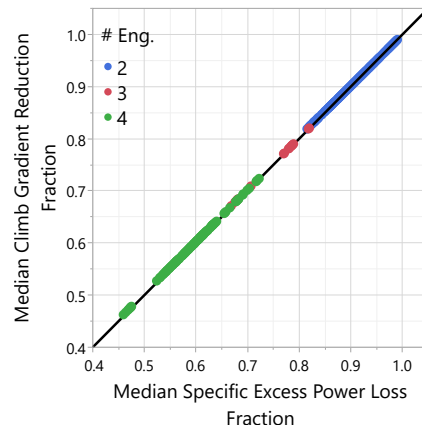
The minimum, median, and maximum CGRFs were also extracted from the operational analysis previously introduced in Section 3.1. For the remainder of this paper, assume that the CGRF refers to the median CGRF unless otherwise specified.

Together, the SEPLF and CGRF are utilized to explore the relationship between the propulsion system failure severity and its impact on climb performance. Figure 3 plots the CGRF against the SEPLF for all aircraft operations simulations, with each plot spanning all takeoff speeds and one ΔV value.

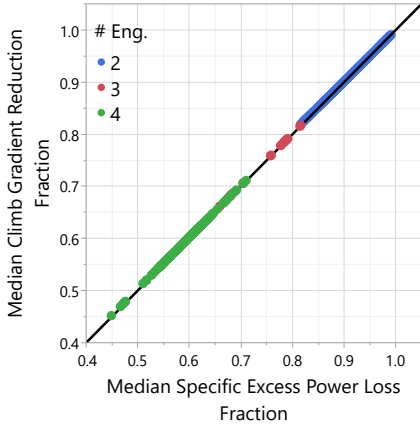
As observed in Fig. 3, the SEPLF closely approximates the CGRF for a given propulsion system failure. For a propulsion system failure on a 2-, 3-, and 4-engine aircraft, the SEPLF nominally spans between 0.80 to 1.00, 0.65 to 0.85, and 0.45 to 0.75, respectively, also reflecting a similar degradation in climb performance. At smaller ΔV values (0.00 and 5.14 m/s), the points deviate the most from the reference line (indicating that $\text{CGRF} = \text{SEPLF}$), but still demonstrate a strong direct relationship. As ΔV increases, the deviation from the reference line decreases, indicating that the SEPLF better approximates the CGRF.



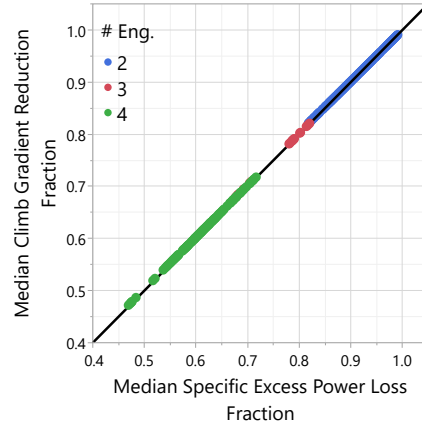
(a) $\Delta V = 0.00$ m/s.



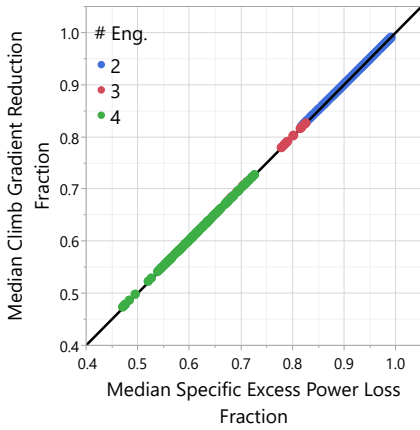
(b) $\Delta V = 5.14$ m/s.



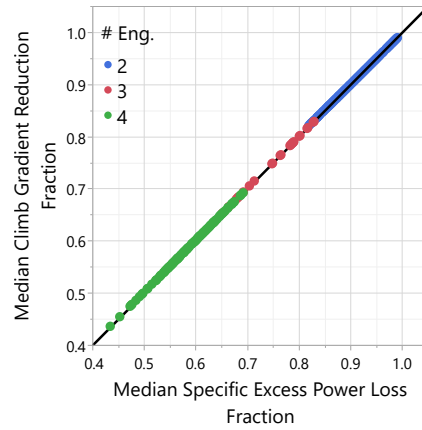
(c) $\Delta V = 10.29$ m/s.



(d) $\Delta V = 15.43$ m/s.



(e) $\Delta V = 20.58$ m/s.

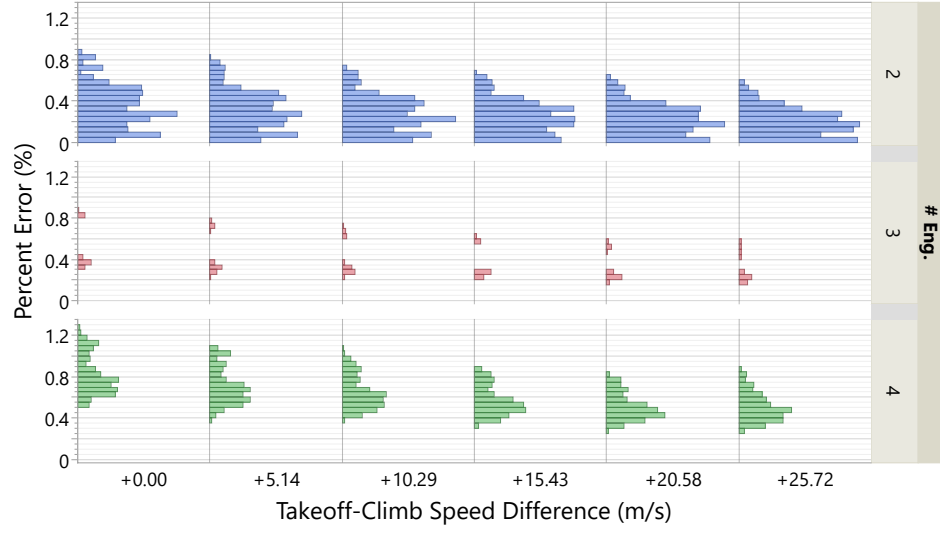


(f) $\Delta V = 25.72$ m/s.

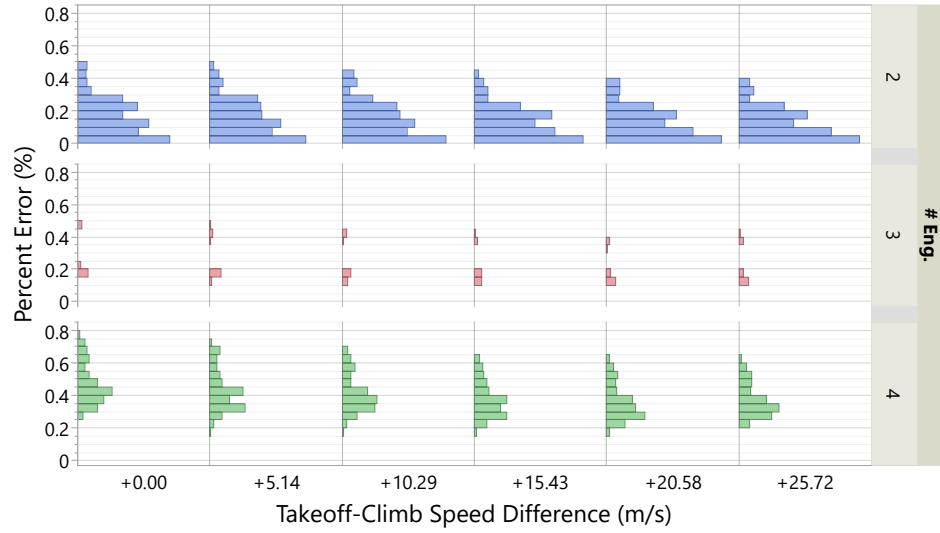
Figure 3: Climb performance impacts due to a propulsion system failure. The black diagonal line is a reference line indicating $\text{CGRF} = \text{SEPLF}$.

Based on the data observed, the SEPLF is approximately equal to the CGRF, and suggests weak sensitivity with respect to the operational strategy. Further investigation is needed to assess if this claim spans all SEPLF and CGRF recorded from the operational analysis. To test this, the percent error between the minimum, median, and maximum SEPLFs and CGRFs was computed via Eq. 6 and plotted in Fig. 4.

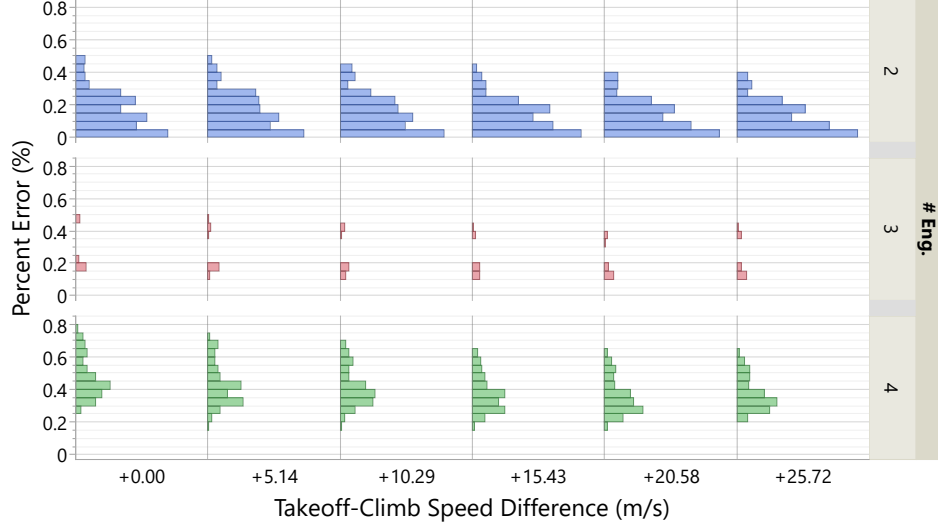
$$\text{Err} = 100 \times \frac{|\text{CGRF} - \text{SEPLF}|}{\text{SEPLF}} \quad (6)$$



(a) Minimum SEPLF and CGRF.



(b) Median SEPLF and CGRF.



(c) Maximum SEPLF and CGRF.

Figure 4: CGRF prediction error for a given SEPLF.

For the all SEPLF and CGRF examined, the percent error is no larger than 1.30%, with the largest error exhibited by 4-engine configurations at the minimum SEPLF and CGRF values recorded during the operational analysis. As ΔV increases, the percent error decreases, particularly for 4-engine aircraft, corroborating the previous observation that the approximation between the SEPLF and CGRF approaches an equality for larger ΔV . Therefore, for the given operational strategies reported – takeoff speeds between 66.88 and 102.89 m/s and climb speeds up to 25.72 m/s faster than the takeoff speed – the SEPLF is approximately equal to the CGRF.

To verify that the SEPLF and CGRF are approximately equal in all operational simulations, the percent error was computed across *all* operating points in the mission profile during both climb segments. Figure 5 plots a cumulative distribution function of the percent error across these operating points, spanning 84,919 samples; additional statistics are presented in Tab. 5. A percent error less than or equal to 1% is demonstrated for over 95% of the data samples considered, with a maximum percent error of 1.53%. The vast majority of samples considered demonstrate less than 0.5% percent error, indicating a strong relationship between the SEPLF and CGRF across all takeoff and climb speeds evaluated. Both observations confirm that the

SEPLF and CGRF are approximately equal within the operational envelope prescribed in Tab. 4, demonstrating the SEPLFs versatility to capture both propulsion system performance and climb performance degradations across a variety of operational strategies.

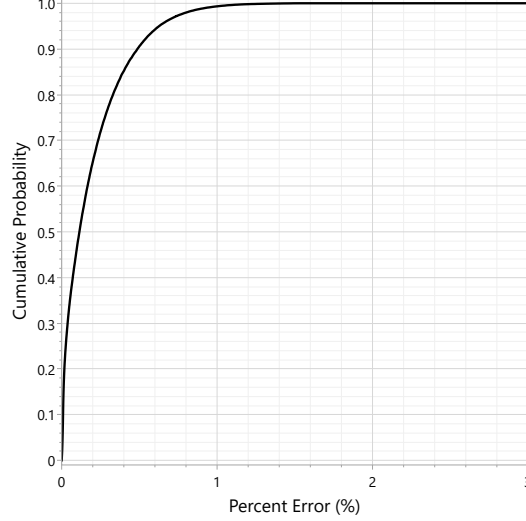


Figure 5: Cumulative distribution function of percent error between the SEPLF and CGRF for *all* operating points during the climb segments modeled.

Table 5: SEPLF-CGRF error across *all* operating points during climb.

Quantity	Percent Error (%)
Mean	0.1889
5 th Percentile	0.0058
10 th Percentile	0.0088
25 th Percentile	0.0262
Median (50 th Percentile)	0.1142
75 th Percentile	0.2780
90 th Percentile	0.4873
95 th Percentile	0.6271

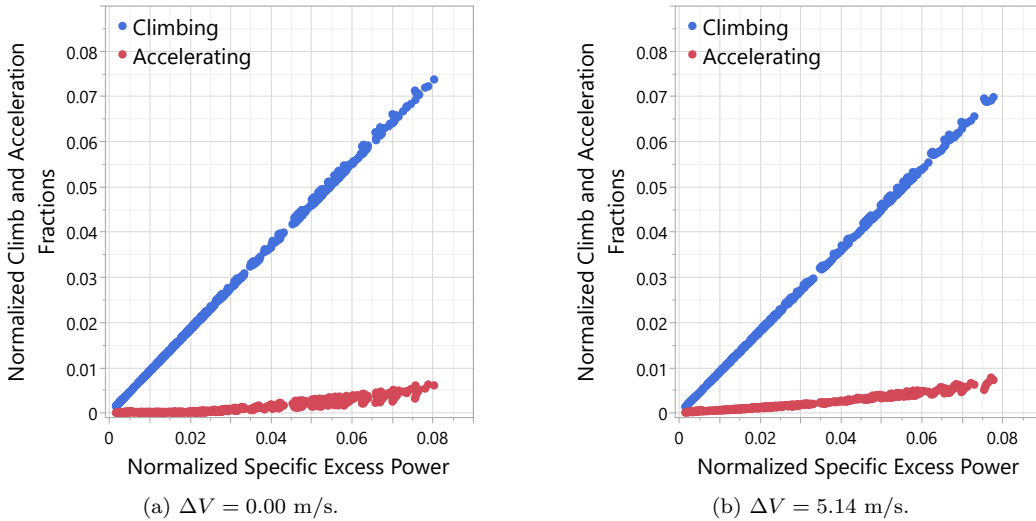
Although approximate equivalence between the SEPLF and CGRF was demonstrated using empirical data from the aircraft operational analysis, it can also be demonstrated mathematically.

3.3. Mathematical Demonstration: Approximate Equivalence between the SEPLF and CGRF

The specific excess power, defined in Eq. 2, is normalized by the airspeed in Eq. 7. The terms on the right hand side of Eq. 7 represent the normalized fraction of specific excess power allocated to climbing and accelerating, respectively. Additionally, the first term on the right-hand side matches the argument required to evaluate Eq. 4, enabling a relation between the specific excess power and climb gradient to be derived.

$$P_s/V_\infty = \dot{h}/V_\infty + \dot{V}_\infty/g \quad (7)$$

The normalized quantities in Eq. 7 are plotted in Fig. 6 to indicate the fractions of specific excess power allocated for climbing and accelerating. Each subplot in Fig. 6 corresponds to a different ΔV . In some cases, the median SEPLF and CGRF occurred during the second climb segment between 914 and 3,048 m while the aircraft was accelerating to its diversion cruise speed of 128.6 m/s, explaining why some of the specific excess power was allocated to accelerating, even in the case that $\Delta V = 0.00$ m/s.



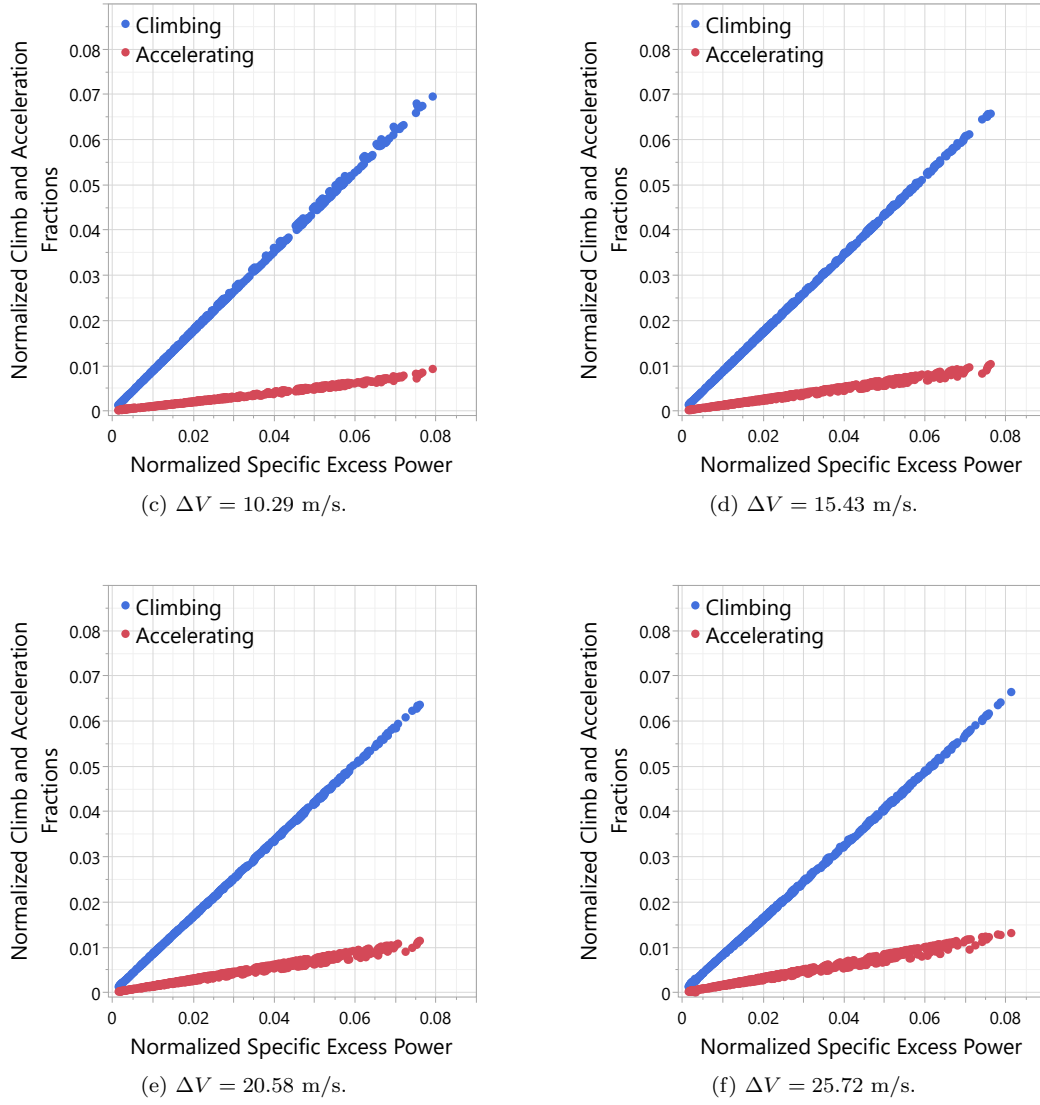


Figure 6: Normalized specific excess power components allocated for climbing and accelerating for varying ΔV .

As ΔV increases, the normalized climb fraction decreases while the normalized acceleration fraction increases, indicating that more specific excess power is utilized to accelerate. However, regardless of the ΔV examined, the specific excess power allocated to climbing is always larger

than that allocated to accelerating. The relative impact of the two contributions is analyzed in Tab. 6 and lists the minimum/maximum normalized climb/acceleration fractions as well as the ratio between the climb and acceleration fractions for the smallest/largest normalized specific excess power values (representative of the varied operational strategies considered).

Table 6: Relative climb/acceleration fraction contributions.

ΔV (m/s)	Climb Fraction		Acceleration Fraction		Fraction Ratio	
	Min.	Max.	Min.	Max.	Min.	Max.
0.00	0.0016	0.0738	0.0001	0.0061	16.00	12.10
5.14	0.0014	0.0698	0.0001	0.0073	14.00	9.56
10.29	0.0013	0.0695	0.0002	0.0093	6.50	7.47
15.43	0.0014	0.0657	0.0002	0.0104	7.00	6.32
20.58	0.0013	0.0636	0.0002	0.0114	6.50	5.58
25.72	0.0012	0.0664	0.0002	0.0131	6.00	5.07

Across the operational strategies sampled, the climb fraction remains at least 5- to 6-times larger than the acceleration fraction. The observations suggest that the vast majority of specific excess power available is utilized for climbing throughout the operational strategies examined. Therefore, it is assumed that the specific excess power allocated for accelerating is negligible relative to that for climbing, and implies that the normalized specific excess power is approximately equal to the normalized climb fraction, as shown in Eq. 8. Since the climb-acceleration fraction ratios decrease as ΔV increases, it is possible that the assumption may not hold for larger ΔV values. Additionally, the constant lift-to-drag ratios utilized during the analysis could also impact the validity of this assumption, requiring additional testing if more detailed geometric and aerodynamic data were publicly available for the aircraft analyzed in the operational simulations.

$$P_s/V_\infty \approx \dot{h}/V_\infty \quad (8)$$

Substituting this approximation into Eq. 4 reveals an approximate relationship between the specific excess power and climb gradient in Eq. 9:

$$\gamma \approx \tan [\arcsin (P_s/V_\infty)] \quad (9)$$

As observed in Fig. 6, the normalized specific excess power for climbing is no larger than 0.08, a value much less than 1. As a result of its small value, Eq. 9 can be linearized about zero to further approximate the relationship between the climb gradient and specific excess power, as illustrated in Eq. 10.

$$\gamma \approx P_s/V_\infty \quad (10)$$

This approximation directly relates the specific excess power to the maximum achievable climb gradient. More broadly, a relationship between the propulsion system performance and overall flight performance has been established.

To connect this approximation to the propulsion system failure severity and flight performance degradation, the CGRF defined in Eq. 5 is rewritten utilizing the approximation established in Eq. 10.

$$\begin{aligned} \gamma_{\text{loss}} &\approx 1 - \frac{P_{s_{\text{OEI}}}/V_\infty}{P_{s_{\text{AEO}}}/V_\infty} \\ &\approx 1 - \frac{P_{s_{\text{OEI}}}}{P_{s_{\text{AEO}}}} \\ &\approx P_{s_{\text{loss}}} \end{aligned} \quad (11)$$

The right-hand side of Eq. 11 represents the SEPLF, yielding an approximate relationship between the flight performance degradation and propulsion system failure severity. Thus, the aircraft's maximum achievable climb gradient can be nominally predicted by the specific excess power lost during a propulsion system failure.

The resulting approximate relationship between the SEPLF and CGRF suggests that it is reasonable to relate the climb requirements established in FAA Part 25 regulations to the SEPLF, enabling climb gradients to be universally defined for any propulsion system failure scenario under the operational conditions simulated in Section 3.1. To this end, expressions parameterized by the SEPLF are constructed next in Section 4 to define generalized requirements and bridge the gap between existing FAA Part 25 regulations and those for future aircraft concepts with novel propulsion systems. The generalized requirements constructed are intended to be used as a tool for predicting expected flight performance requirements for future aircraft concepts; compliance under FAA Part 25 regulations (or future relevant certification criteria) still must be demonstrated by the airframe manufacturer.

4. Requirement Construction

Before constructing the generalized requirements, historical SEPLF trends are first examined in Section 4.1 to map the existing FAA climb requirements to a nominal SEPLF for 2-, 3-, and 4-engine aircraft previously certified by the FAA. Then, the generalized requirements are constructed in Section 4.2 by fitting a curve to the previous mapping generated. Lastly, Section 4.3 constructs an expression defining an engine inoperative correction factor for any propulsion system failure (also parameterized by the SEPLF). Since the SEPLF exhibited weak sensitivity with respect to the operational strategy, all analyses here utilize the SEPLF corresponding to a ΔV of 25.72 m/s. Such a value was chosen because the error between the SEPLF and CGRF was the lowest among all ΔV values considered in Section 3.

4.1. Specific Excess Power Loss Trends for Conventional Aircraft

The SEPLF distribution for all 156 aircraft at a ΔV of 25.72 m/s is plotted in Fig. 7. Additional SEPLF plots for the other ΔV are provided in Appendix D.

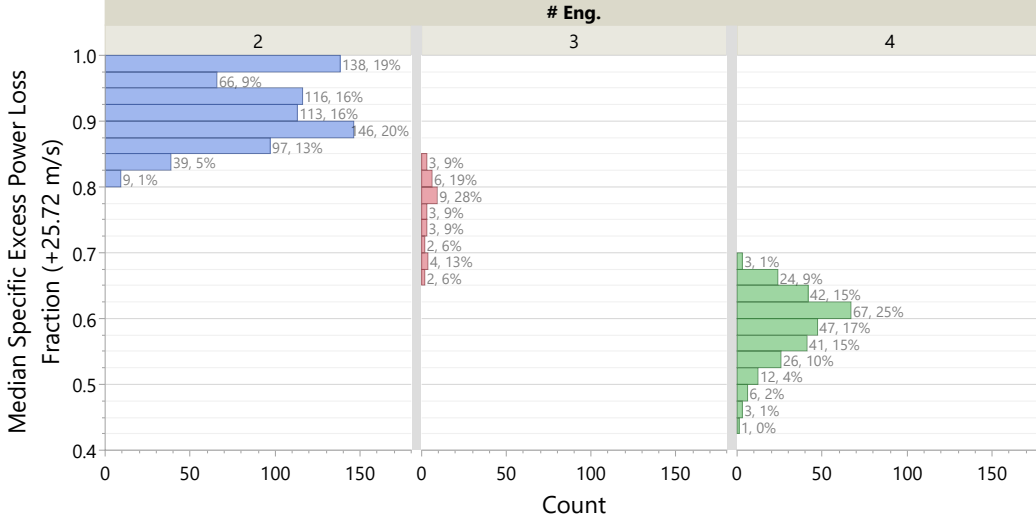


Figure 7: SEP loss fraction distribution, $\Delta V = 25.72$ m/s.

As observed in Fig. 7, all SEPLFs for 2-engine aircraft are between 0.80 and 1.00, with the majority between 0.85 and 0.95. Due to the limited data

available for 3-engine aircraft, the SEPLFs are scattered between 0.65 and 0.80, though the majority are closer to its upper bound. The SEPLFs for 4-engine aircraft vary between 0.40 and 0.70, with the majority between 0.55 and 0.65.

The data also reveals that the SEPLF decreases as more engines are installed on the airframe, and is a result of both propulsive, aerodynamic, and stability/control considerations. First, configurations with more engines installed have a larger fraction of their thrust power available remaining after an engine failure. For example, after an engine failure, 50% of the thrust power available remains on a 2-engine configuration, while 66% and 75% of the thrust power remains on 3- and 4-engine configurations, respectively. Second, configurations with fewer engines have larger fan diameters, resulting in more windmilling drag when the engine fails. The yaw moment induced by the increased windmilling drag requires a larger rudder deflection to maintain directional stability, further increasing the parasite drag and the SEPLF.

Percentiles from each distribution, provided in Tab. 7 for $\Delta V = 25.72$ m/s, highlight the range of SEPLFs exhibited under engine inoperative conditions for aircraft previously certified under FAA Part 25 regulations. Additional tables listing the distributional anchors for the remaining ΔV are available in Appendix D.

Table 7: Selected quantities across the SEPLF distributions for $\Delta V = 25.72$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8587	0.6788	0.5344
25	0.8794	0.7213	0.5634
50	0.9171	0.7828	0.6000
75	0.9604	0.8001	0.6265
90	0.9828	0.8240	0.6501

Using the data from Tab. 7, the median (50th percentile) SEPLF for 2-, 3-, and 4-engine aircraft are mapped to their corresponding FAA climb performance requirements, as listed in Tab. 8. The mapping re-parameterizes the FAA climb performance requirements by the SEPLF (an analogue for propulsion system failure severity) rather than the number of engines installed on the aircraft, and serves as a basis for constructing the generalized climb performance requirements.

Table 8: Median SEPLFs from the operational analysis mapped to the corresponding FAA Part 25 regulations.

Requirement	Median SEPLF		
	0.9171	0.7828	0.6000
25.111 (Takeoff Flight Path, OEI)	$\gamma \geq 1.2\%$	$\gamma \geq 1.5\%$	$\gamma \geq 1.7\%$
25.119 (Landing, AEO)	$\gamma \geq 3.2\%$	$\gamma \geq 3.2\%$	$\gamma \geq 3.2\%$
25.121a (First Segment Climb, OEI)	$\gamma \geq 0.0\%$	$\gamma \geq 0.3\%$	$\gamma \geq 0.5\%$
25.121b (Second Segment Climb, OEI)	$\gamma \geq 2.4\%$	$\gamma \geq 2.7\%$	$\gamma \geq 3.0\%$
25.121c (Final Climb, OEI)	$\gamma \geq 1.2\%$	$\gamma \geq 1.5\%$	$\gamma \geq 1.7\%$
25.121d (Landing, OEI)	$\gamma \geq 2.1\%$	$\gamma \geq 2.4\%$	$\gamma \geq 2.7\%$

4.2. Constructing Generalized Engine Inoperative Performance Requirements

Using the mapping outlined in Tab. 8, a Sigmoid Curve was fit to the data for each FAA Part 25 climb requirement. A general Sigmoid Curve equation is defined in Eq. 12 and parameterized by the SEPLF. The coefficients a and d affect the curve’s upper and lower asymptotes while b and c affect the curve’s steepness and inflection point.

$$\hat{\gamma}(P_{s_{\text{loss}}}) = \frac{a}{1 + \exp[-b(P_{s_{\text{loss}}} - c)]} + d \quad (12)$$

The Sigmoid Curve was selected because of its ability to model asymptotic behavior, maintaining consistency with existing FAA climb requirements at the upper/lower SEPLF bounds. Such behavior is critical for guaranteeing that the performance requirements for future aircraft concepts are similar to those for conventional aircraft, particularly towards SEPLFs of 0.00 to 0.40 (reflective of regimes for distributed propulsion system failures). Additionally, the Sigmoid Curve is both value- and slope-continuous, making it easy to implement in a multidisciplinary design optimization environment, particularly if derivatives are computed analytically.

The Sigmoid Curve was fit to the data by solving the optimization problem in Eq. 13; the root mean square error (RMSE) is minimized with respect to the mapped climb performance requirements listed in Tab. 8 by varying the coefficients governing the Sigmoid Curve’s shape. To ensure that the Sigmoid Curves exhibit asymptotic behavior at extreme SEPLF

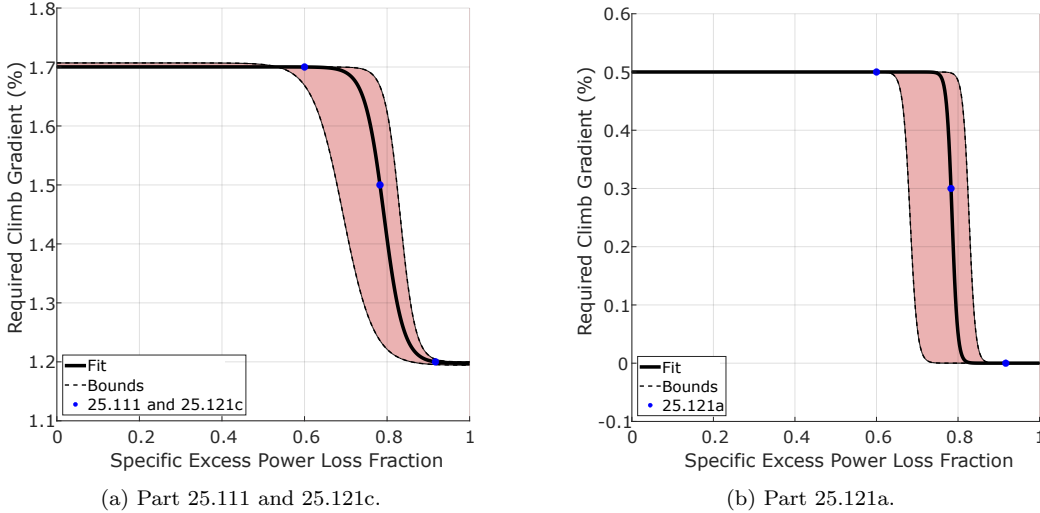
values, the beginning and ending slopes ($\hat{\gamma}'(0)$ and $\hat{\gamma}'(1)$, respectively) were constrained to be within $\pm 10^{-2}$.

$$\begin{aligned} \text{Minimize:} \quad & \text{RMSE} = \sqrt{\frac{\sum_{i=1}^n [\gamma_i - \hat{\gamma}_i]^2}{n}} \\ \text{With Respect to:} \quad & \text{Sigmoid Curve Parameters } a, b, c, \text{ and } d \\ \text{Subject to:} \quad & -10^{-2} \leq \hat{\gamma}'(0) \leq +10^{-2} \\ & -10^{-2} \leq \hat{\gamma}'(1) \leq +10^{-2} \end{aligned} \quad (13)$$

Additionally, the climb performance requirement under FAA Part 25.121a necessitates the climb gradient to be greater than 0% for 2-engine aircraft. To ensure that a nonnegative climb gradient is prescribed across all SEPLFs, an additional constraint was imposed, requiring that $\hat{\gamma}(1) \geq 0$.

The optimized Sigmoid Curves are plotted in Fig. 8; their coefficients and corresponding RMSE values are listed in Tab. 9. Uncertainty bounds are also plotted by constructing Sigmoid Curves using data from the 10th and 90th SEPLF percentiles listed in Tab. 7, as indicated by the red shaded region. No Sigmoid Curve was constructed for FAA Part 25.119 requirements because the minimum climb gradient for all configurations is 3.2% and can already be universally applied to current and future aircraft concepts.

Overall, the Sigmoid Curves illustrated in Fig. 8 capture the FAA Part 25 regulations while spanning all propulsion system failure severities. The constrained derivatives at the endpoints prevent the climb gradients



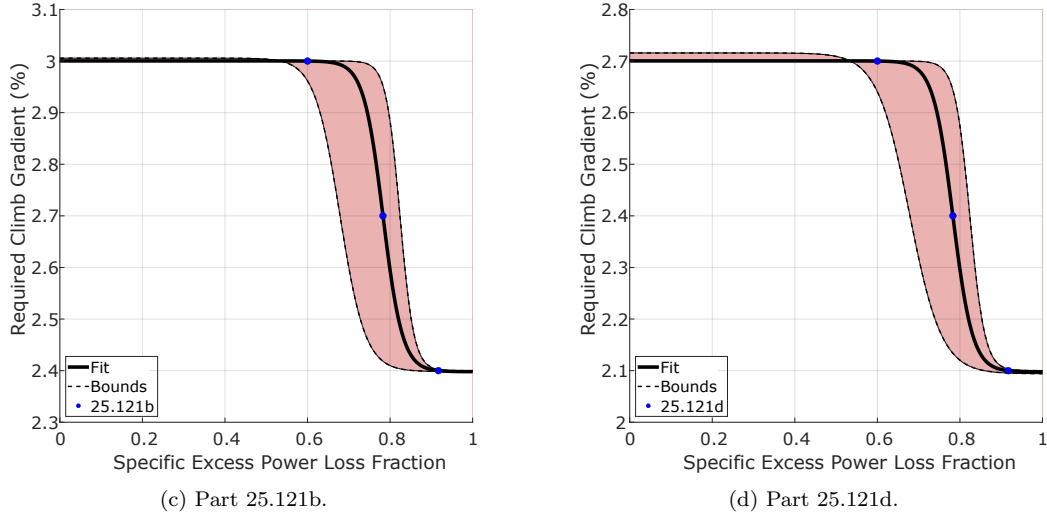


Figure 8: Sigmoid Curves constructed to represent FAA Part 25 climb performance requirements.

Table 9: Optimized Sigmoid Curve parameters for all FAA Part 25 performance requirements.

FAA Regulation	RMSE	a	b	c	d
25.111 and 25.121c	1.68×10^{-8}	0.5026	-42.54	0.7925	1.198
25.121a	4.00×10^{-5}	0.4999	-151.22	0.7855	0.001
25.121b	1.80×10^{-8}	0.6024	-42.00	0.7829	2.398
25.121d	3.50×10^{-8}	0.6025	-41.75	0.7830	2.098

from becoming either too aggressive or conservative relative to the existing requirements, and guarantees that conventional and future aircraft concepts experiencing propulsion system failures with similar SEPLFs will be sized for similar climb performance requirements.

Although most of the curves smoothly transition between data points, the curve modeling FAA Part 25.121a in Fig. 8b is steeper due to the additional constraint requiring $\hat{\gamma}(1) \geq 0$, resulting in a stiffer optimization problem and an RMSE that is three-orders of magnitude larger than the other curve fits. Despite the poorer RMSE, the curve fit still reasonably and accurately models the climb performance requirements.

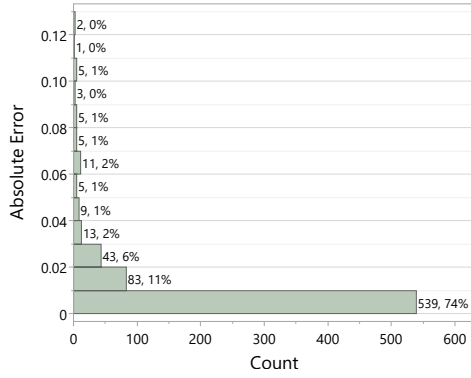
All Sigmoid Curves are constructed with asymptotic regions spanning SEPLFs from 0.00 to 0.60 (upper asymptote) and 0.9171 to 1.00 (lower asymptote), aligning with the climb performance requirements for 4- and 2-engine aircraft, respectively, revealing that such requirements may be utilized for a majority of the aircraft designs considered. In other words, the Sigmoid Curve constructions nearly act as step-functions, quickly transitioning between the required climb gradients for 2- and 4-engine aircraft. More broadly, the generalized requirement constructions suggest that each climb requirement could be defined with a larger climb requirement for SEPLFs from 0.00 to 0.7828 and a smaller climb requirement for larger SEPLFs from 0.7828 to 1.00.

Additionally, as observed in the literature review, the FAA climb performance requirements for 4-engine aircraft were previously employed to design aircraft with distributed propulsion systems [23, 26, 27]. The Sigmoid Curves constructed reveal that this is a reasonable method to size such aircraft in the absence of generalized climb performance regulations, validating the assumptions made in prior research. Assuming that a distributed propulsion system failure involves one or two propulsors, the SEPLF is likely to be less than 0.60, aligning with the upper asymptote on the Sigmoid Curves, corresponding to the 4-engine climb requirements prescribed by the FAA. Thus, for future aircraft conceptual design studies, it may be reasonable to utilize the FAA’s 4-engine climb performance requirements as a general “rule-of-thumb” until improved regulations are released.

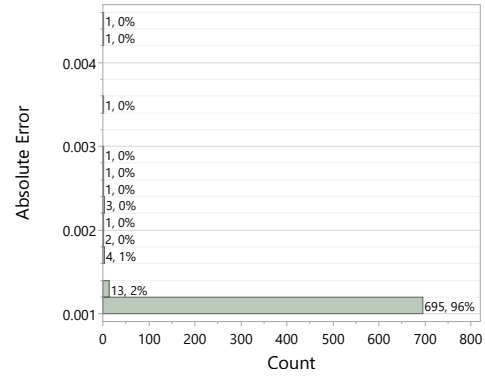
4.2.1. Generalized and FAA Climb Performance Requirement Comparison

After constructing the generalized performance requirements, a test was performed to assess the difference between them and those prescribed by the FAA for each aircraft considered in the operational analysis from Section 3. The test was performed by computing the absolute difference between each FAA and generalized climb performance requirement, as described in Eq. 14. Ideally, the required climb gradients minimally deviate from each other, maintaining consistency for previously certified aircraft. Figure 9 plots histograms of the absolute error for all 2-engine aircraft examined in the operational analysis. Additional histograms for 3- and 4-engine aircraft are provided in Figures E.14 and E.15 within Appendix E.

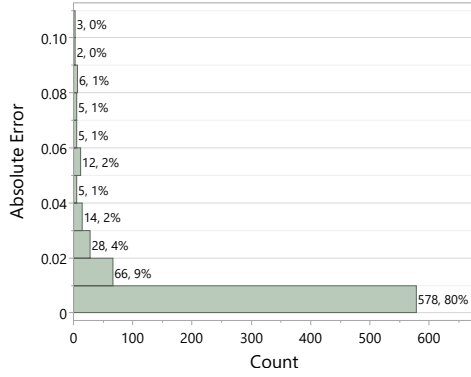
$$\text{Error} = |\hat{\gamma} - \gamma_{\text{FAA}}| \quad (14)$$



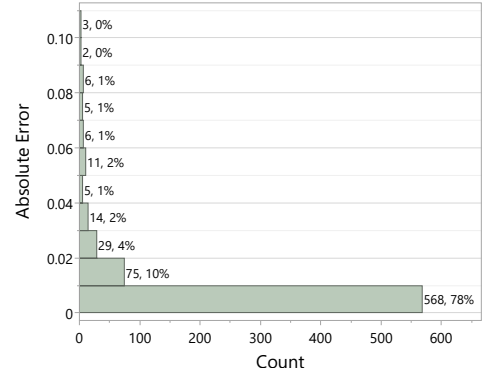
(a) 25.111 and 25.121c.



(b) 25.121a.



(c) 25.121b.



(d) 25.121d.

Figure 9: Absolute error between FAA and proposed flight performance requirements for 2-engine aircraft.

The histograms in Fig. 9 indicate that at least 74% of the climb gradients are modeled within an absolute error of 0.01 for 2-engine aircraft, indicating reasonable consistency between the generalized and existing FAA requirements. Such consistency is maintained because the required climb gradients slightly increase for SEPLFs smaller than the median one and marginally decrease for SEPLFs larger than the median one. Similar behavior is observed in Fig. E.15 for 4-engine aircraft (see Appendix E); the largest absolute error is no larger than 0.0125, approximately 8- to 10-times smaller than that for 2-engine aircraft. The median SEPLF for 4-engine aircraft lies on the upper asymptote of each Sigmoid Curve and is farther away from the

regions of significant curvature change than that for 2-engine aircraft, thus explaining the reduction in absolute error.

However, for 3-engine aircraft, the absolute error is large as 0.3 (see Fig. E.14 in Appendix E), indicating possible inconsistencies between the generalized and FAA climb performance requirements. The discrepancy is a result of the median SEPLF for 3-engine aircraft residing in the regions of the Sigmoid Curve with the steepest slopes, resulting in significant climb performance requirement differences for small SEPLF perturbations. As a result, existing 3-engine aircraft may be over- or under-designed for the generalized climb performance requirements. Despite the error observed, the results are less concerning because all 3-engine examined in this study entered into service between 1963 and 1975, and are no longer frequently manufactured or operated.

In summary, the results of the test performed suggests that the proposed performance requirements are nominally equivalent to those already defined in FAA Part 25 regulations for 2- and 4-engine aircraft, and can be useful for designing future aircraft with both conventional and advanced propulsion systems.

4.3. Defining Generalized Engine Inoperative Correction Factors

Next, an expression for the “engine inoperative correction factor”, $\hat{k}$, was constructed as a function of the SEPLF. Rather than mapping the median SEPLFs to the FAA climb performance requirements as done in Section 4.2, they were mapped to traditional $n/(n - 1)$ engine inoperative correction factors previously listed in Tab. 2. Then, a curve fit to the data is performed, again minimizing the RMSE with respect to the previously described mapping.

Additionally, the engine inoperative correction factor (also referred to as the “correction factor”) must satisfy three requirements:

1. Return a value of 1 for a SEPLF of 0, suggesting that the propulsion system need not be enlarged if no specific excess power is lost.
2. Be monotonically increasing, implying that the propulsion system must become larger as more specific excess power is lost.
3. Correction factors should not be excessively large for configurations with SEPLFs that approach 1, preventing existing twin-engine aircraft from being over-sized relative to their current design points.

Due to the plethora of parametric curves available for fitting the data, an optimization problem was formulated and tested on multiple candidates. The optimization problem, listed in Eq. 15, minimizes the curve fit's RMSE subject to three constraints, each of which satisfy one of the three requirements listed previously. The first two constraints are enforced mathematically by requiring $\hat{\kappa}(0) \geq 1$ and $d\hat{\kappa}/dP_{s_{\text{loss}}} \geq 0$ for all SEPLFs. The third constraint was satisfied by limiting the maximum allowable engine inoperative correction factor to be 2.25, up to 12.5% larger than the traditional $n/(n-1)$ correction factors.

$$\begin{aligned}
&\text{Minimize: } \text{RMSE} = \sqrt{\frac{\sum_{i=1}^n [(\kappa_i - \hat{\kappa}_i)^2]}{n}} \\
&\text{With Respect to: } \text{Curve Parameters } \mathbf{c} \\
&\text{Subject to: } \hat{\kappa}(0) = 1 \\
&\quad d\hat{\kappa}/dP_{s_{\text{loss}}} \geq 0 \\
&\quad \hat{\kappa}(1) \leq 2.25
\end{aligned} \tag{15}$$

The curves tested, their general equations, and their corresponding RMSE errors are listed in Tab. 10. All candidate curves return an RMSE on the order of 10^{-1} or 10^{-2} , suggesting that there are many suitable candidates. Further qualitative inspection of each curve fit is required to find the optimal one. To this end, the six candidate curves are plotted in Fig. 10.

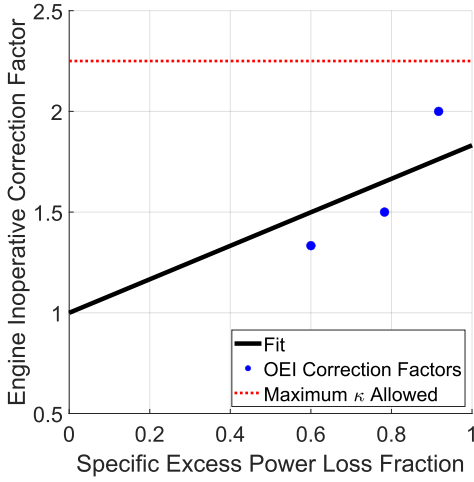
Table 10: Curve fits tested for the OEI correction factor, maximum correction factor allowed is 2.25.

Curve	General Equation	RMSE
Linear	$c_1x + c_2$	1.885×10^{-1}
Quadratic	$c_1x^2 + c_2x + c_3$	1.098×10^{-1}
Cubic	$c_1x^3 + c_2x^2 + c_3x + c_4$	5.735×10^{-2}
Exponential	$c_1 \exp[c_2(x - c_3)] + c_4$	6.357×10^{-2}
Tangent	$c_1 \tan[c_2(x - c_3)] + c_4$	7.045×10^{-2}
Hyperbolic Sine	$c_1 \sinh[c_2(x - c_3)] + c_4$	6.257×10^{-2}

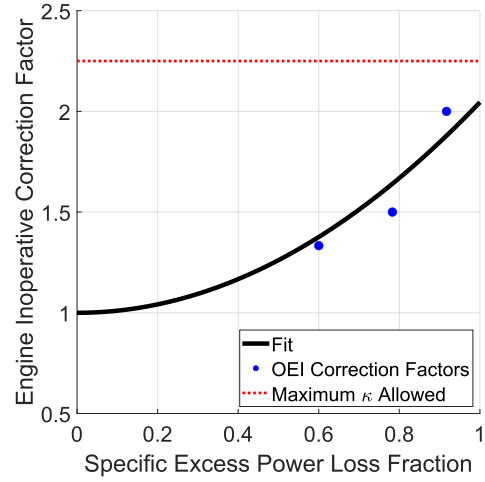
First, the cubic, exponential, tangent, and hyperbolic sine curves are considered. For SEPLFs greater than 0.7, each curve's slope is significantly larger than those of the linear and quadratic ones, suggesting that the propulsion system size increases rapidly for more severe propulsion system failures. Given that tube-and-wing aircraft designs with turbofan engines are

technologically mature, the propulsion system size should not significantly grow for modest increases in the propulsion system failure severity, rendering these four curves less desirable.

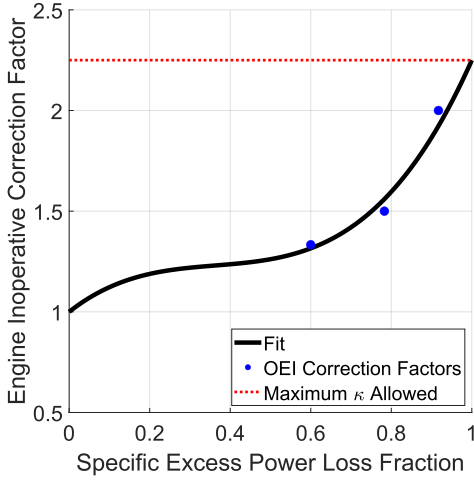
There are multiple distinguishing features between the linear and quadratic curves. The linear function has a shallower slope than the quadratic curve at larger SEPLFs, but fails to reach a correction factor of 2, resulting in aircraft being under-sized relative to those that are presently



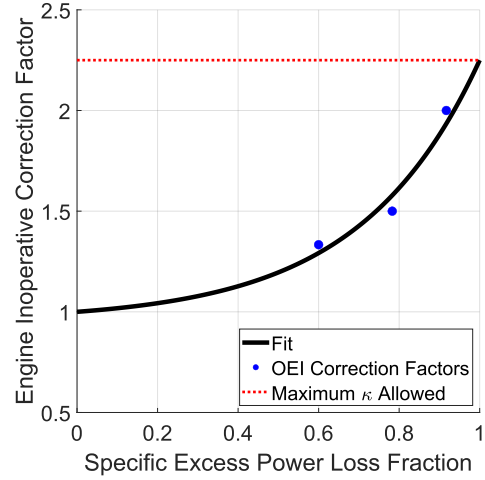
(a) Linear function.



(b) Quadratic function.



(c) Cubic function.



(d) Exponential function.

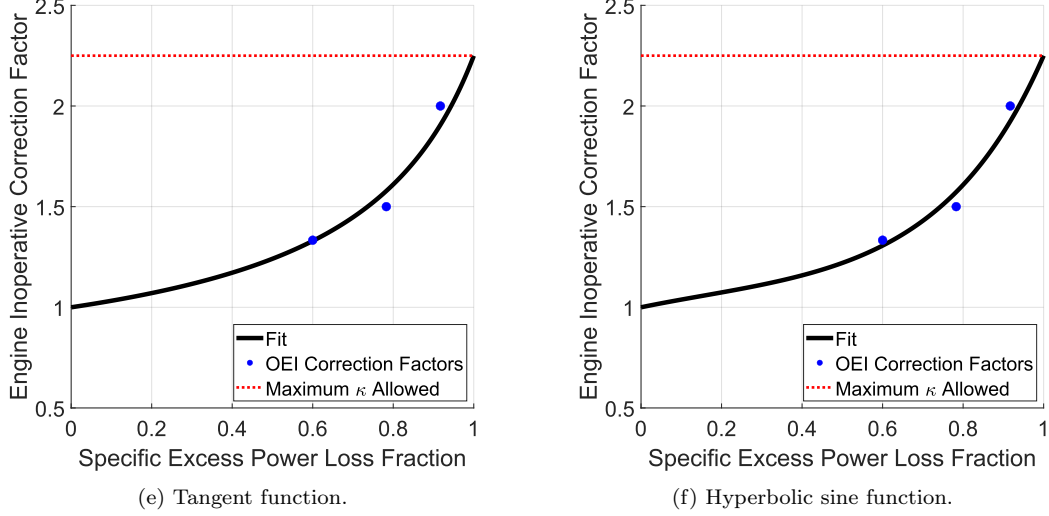


Figure 10: Candidate engine inoperative correction factor curve fits.

sized with $n/(n - 1)$ correction factors. The quadratic function, however, marginally exceeds a correction factor of 2 for SEPLFs greater than 0.95, rendering the constraint on $\hat{\kappa}(1)$ to be inactive. As a result, the propulsion system size would not significantly increase relative to existing ones sized with $n/(n - 1)$ correction factors.

While both curves over-predict the correction factors for 3- and 4-engine aircraft and under-predict that for 2-engine aircraft, the quadratic curve returns a lower RMSE than the linear curve. Thus, the quadratic curve was selected as the preferred one to be utilized for defining an engine inoperative correction factor. Equation 16 defines the engine inoperative correction factor with its optimized parameters:

$$\hat{\kappa}(P_{s_{\text{loss}}}) = 1.045 (P_{s_{\text{loss}}})^2 + 1.000 \quad (16)$$

4.3.1. Alternative Maximum Correction Factor Values

Previously, the maximum correction factor value was heuristically selected to be 2.25. As a result, the cubic, exponential, tangent, and hyperbolic sine curves exhibited lower RMSE values than the quadratic function selected, but the required propulsion system size increased rapidly for more severe propulsion system failures, making them less desirable choices

for representing the correction factor. The maximum correction factor output by the quadratic curve was 2.05 (see Eq. 16), resulting in the constraint on the maximum correction factor to be inactive, as opposed to being active for the other four curves. Therefore, an experiment is necessary to investigate whether or not the quadratic curve fit remains the preferred option if the maximum correction factor were to change.

The same curve fit outlined in Eq. 15 is performed, but with a maximum correction factor of 2.05, matching the maximum correction factor output by the quadratic function. Table 11 lists the updated RMSE from each curve fit, with plots of each curve fit illustrated in Fig. F.16 (see Appendix F).

Table 11: Curve fits tested for the OEI correction factor, maximum correction factor allowed is 2.05.

Curve	RMSE
Linear	1.885×10^{-1}
Quadratic	1.098×10^{-1}
Cubic	9.875×10^{-2}
Exponential	1.027×10^{-1}
Tangent	1.090×10^{-1}
Hyperbolic Sine	1.027×10^{-1}

As the maximum allowable correction factor decreases, the RMSE of the cubic, exponential, tangent, and hyperbolic sine curves all increase and begin to approach that of the quadratic function. As observed in Fig. 10, the four aforementioned curves all over-predicted the correction factor for 3-engine aircraft but under-predicted that for 2- and 4-engine aircraft. By imposing a smaller maximum engine inoperative correction factor, the exponential, tangent, and hyperbolic sine functions all over-predict the engine inoperative correction factor for 3- and 4-engine aircraft but under-predict that for 2-engine aircraft, exhibiting a very similar behavior to the quadratic function.

The convergent behavior towards that of the quadratic curve suggests that it remains as the preferred candidate to represent the engine inoperative correction factor, particularly if the maximum engine inoperative correction factor is to decrease further and be constrained to a value of 2, matching the largest $n/(n - 1)$ correction factor traditionally used. Further studies would be necessary to identify the preferred curve if the maximum allowable correction factor were to increase beyond 2.25. However, this seems unlikely

because the size of existing propulsion systems for tube-and-wing aircraft may further increase, making them larger than necessary to meet the FAA climb performance requirements.

5. Case Study: Notional Elysian E9X Sizing

The generalized climb requirements and engine inoperative correction factor established in Section 4 are now applied to size a notional Elysian E9X (hereinafter referred to as the “E9X”), a battery-electric aircraft that carries 90 passengers and utilizes a distributed propulsion system with eight propulsors [26, 44]. First, the notional E9X is modeled in FAST to identify the SEPLFs associated with multiple novel engine inoperative scenarios. Next, separate constraint diagrams for each engine inoperative scenario are generated using FAA Part 25 regulations and the generalized climb performance requirements from Section 4, enabling a comparison between the selected sizing points. Lastly, the selected sizing points are utilized to resize the notional E9X. The system-level impacts observed highlight the importance of generalizing the performance requirements for future aircraft concepts with advanced propulsion systems.

In this study, the traditional “one engine inoperative” condition utilized on 2- to 4-engine aircraft is extrapolated to an aircraft with a distributed propulsion system. Rather than simulating traditional “one engine inoperative” conditions, use cases of 1-6 propulsor failures are considered. The constraint diagrams generated and design points selected for each failure scenario are derived from the generalized climb performance requirement and engine inoperative correction factor constructions presented in Section 4. Such constructions are conceptual design heuristics grounded in the fleet behavior of previously certified aircraft, and are only intended to provide a first-order sizing estimate rather than a means to certify a future aircraft concept.

Rather than prescribing compliance, the framework developed in this paper is intended to help the airframe manufacturer explore the design space and predict the sensitivity of the aircraft’s system-level performance with respect to multiple candidate primary propulsion system failure modes. Given that the presently published FAA Part 25 requirements were developed only for conventional aircraft powered by 2, 3, or 4 engines, the authors envision that certification authorities would consider multi-propulsor failures as a “special condition,” pertaining to only the aircraft seeking certification;

the multi-propulsor failures considered under the special conditions would be driven by the airframe manufacturer’s system safety assessments.

5.1. Aircraft Modeling

A notional E9X model was developed in FAST [34], serving as a baseline for identifying the SEPLFs under multiple novel engine inoperative scenarios. All parameters used to model the notional E9X are included in Tab. G.25 in Appendix G, along with all modeling assumptions and approximations made. The boundary conditions for an assumed design mission profile are defined in Tab. G.26 in Appendix G, as no detailed one was defined in the literature. To limit the battery size, the E9X flies its reserve mission utilizing a “Reserve Energy System” comprised of a fuel-powered turbogenerator, producing electrical power for the distributed propulsion system. However, the weight breakdowns in Refs. [26, 44] fail to provide exact weight contributions of such systems, obfuscating their overall contribution to the maximum takeoff weight (MTOW) and system-level performance. As a result, the analysis in this work was simplified by excluding the reserve mission from the flight simulation and assuming that the Reserve Energy System weight is accounted for in the aircraft’s operational empty weight.

The notional E9X was calibrated by defining an “airframe calibration factor” (included in Tab. G.25 in Appendix G) to tune the model until its MTOW and battery weight closely matched those published in the literature. Although a battery weight was not explicitly reported, Fig. 3 in Ref. [26] estimates that the battery weight is 46% of MTOW, enabling model validation. As shown in Tab. 12, the minimal error in MTOW and less than 3% error on the battery mass is a reasonable level of error for a conceptual design study, indicating that model calibration was successful.

Table 12: Model calibration for the notional E9X.

Mass	Literature (kg)	FAST Model (kg)	Percent Error (%)
MTOW	76,000	76,047	+0.06
Battery	34,960	35,985	+2.93

After calibrating the notional E9X, novel engine inoperative scenarios were defined for one to six propulsor failures, assuming that they all simultaneously failed immediately after takeoff. Although multiple propulsor failures are extreme and highly unlikely, they account for operational

anomalies, such as bird strikes, as well as flight control system failures, requiring multiple propulsors to be shutdown to maintain lateral and directional stability. These failures were simulated on the generic diversion mission profile previously outlined in Tab. A.17 as part of Appendix A. For this case study, two changes were made to the mission profile:

1. The climb speed was set equal to the takeoff speed ($\Delta V = 0$ m/s).
2. The diversion cruise speed was reduced to Mach 0.40, as defined in the literature [26].

The SEPLFs were computed for each engine inoperative scenario, as outlined in Tab. 13. Due to the lack of publicly available geometric data available for the notional E9X, a detailed aerodynamic model could not be used, meaning that windmilling and trim drag were not accounted for. As a result, the SEPLFs are underestimated relative to their actual value if engine inoperative conditions were to be simulated in a physical environment.

Table 13: SEPLFs for each novel engine inoperative scenario.

Propulsors Failed	Specific Excess Power (m/s)	Specific Excess Power Loss Fraction
0	11.6531	—
1	9.7607	0.1624
2	7.8698	0.3247
3	5.9820	0.4867
4	4.0972	0.6484
5	2.2156	0.8099
6	0.3370	0.9711

Relative to AEO conditions, the SEPLF increments by approximately 0.16 for each propulsor failure. Compared to an engine inoperative scenario on an average 4-engine aircraft, the SEPLF for one engine inoperative on the notional E9X is approximately 3.75-times smaller, highlighting that distributed propulsion systems facilitate a more graceful flight performance degradation during such failures. The SEPLFs are then used to construct constraint diagrams under both the FAA and generalized climb performance requirements.

5.2. Constraint Diagram Generation and Sizing Point Comparison

First, a notional E9X constraint diagram is generated and validated against one published in the literature. Private communications with the authors of Ref. [26] confirmed that the propulsion system size was corrected for engine inoperative conditions (using a traditional $n/(n - 1)$ correction factor for an 8-engine aircraft) and MTOW during the sizing process. Second, the constraint diagram is regenerated using the proposed performance requirements and corresponding correction factors. For each of the constraint diagrams generated, the selected design point is utilized to resize the E9X in FAST, enabling system-level performance and electric motor sizes to be compared.

5.2.1. Notional E9X Constraint Diagram and Validation

A constraint diagram for the notional E9X was generated using FAA Part 25 climb performance requirements for a 4-engine aircraft and accounted for the aforementioned correction factors. In addition to the climb constraints, the authors of Ref. [26] also imposed an AEO climb gradient to ensure that the E9X had enough power to climb after takeoff. All parameters to generate the constraint diagram are listed in Tab. G.27 within Appendix G, including the aerodynamic performance modeling methodology and assumptions. The resulting constraint diagram is plotted in Fig. 11, with an optimum sizing point corresponding to a wing loading of 5 kN/m² and power loading of 0.067 N/W, matching the values obtained in prior literature [26].

The constraint diagram was validated by utilizing an online digitizer³ to capture the power loading values corresponding to each constraint at wing loadings of 4 and 6 kN/m². Table 14 lists the digitized values and the absolute/relative error for a wing loading of 4 kN/m². Table G.28 in Appendix G reports the absolute/relative error for a wing loading of 6 kN/m². The approach and landing field length constraints are not included in the tables because their values only depend on the wing loading, and were matched exactly.

The error reported during validation is attributed to two key modeling differences. First, the authors in Ref. [26] model the takeoff field length using a semi-empirical method from Torenbeek [45], dividing the takeoff field length into rolling and airborne distances, and requires knowledge of the

³<https://apps.automeris.io/wpd4/>

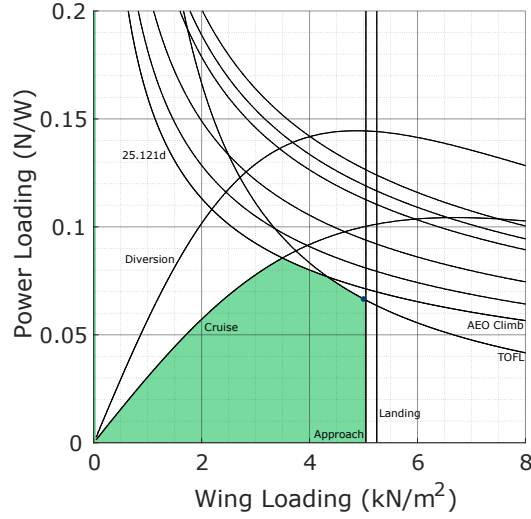


Figure 11: Baseline constraint diagram to match literature. The feasible region is shaded in green.

Table 14: Constraint diagram comparison at a 4 kN/m^2 wing loading.

Constraint	Model	Literature	Absolute Error	Relative Error
FAA 25.121a	0.1438	0.1672	-0.0234	-13.995%
FAA 25.121b	0.1278	0.1302	-0.0024	-1.843%
FAA 25.121c	0.1353	0.1472	-0.0119	-8.084%
FAA 25.121d	0.0918	0.0979	-0.0061	-6.231%
FAA 25.119	0.1065	0.1050	+0.015	-1.429%
AEO Climb	0.0808	0.0811	-0.0003	-0.370%
Cruise	0.0930	0.0958	-0.0028	-2.923%
Diversion	0.1438	0.1354	+0.0084	+6.204%
Takeoff Field Length	0.0843	0.0906	-0.0063	-6.954%

second-segment climb speed (V_2 , previously shown in Fig. 1). In this work, the takeoff field length is simply approximated using the takeoff parameter provided by Roskam [30]. Second, due to the lack of publicly available aerodynamic data, the cruise and diversion constraints were scaled by a factor of $1/\sigma^a$, where σ is the density ratio at the respective altitude and a is a

user-selected parameter to tune the constraint; this factor typically accounts for gas turbine engines lapsing. Although electric motors do not lapse at altitude, tuning the constraints in such a manner enables the modeling approach to be seamlessly adapted for concepts utilizing gas turbine engines or turbogenerators in the future, particularly if the E9X's Reserve Energy System is modeled in greater detail. Given that the aerodynamic parameters were estimated using first-order approximations and that little geometric information is provided about the E9X, the level of modeling error present is deemed acceptable.

5.2.2. Notional E9X Sizing with the Generalized Requirements

To fully assess the system-level impacts of sizing an aircraft for different propulsion system failure severities, constraint diagrams are now regenerated utilizing the generalized climb performance requirements and engine inoperative correction factor. The resultant constraint diagrams are plotted in Fig. 12 for 1- and 6-engine inoperative conditions; additional constraint diagrams for the other engine inoperative conditions are in Fig. G.17 within Appendix G.4. Again, given that the windmilling and trim drag were not computed due to the lack of geometric data publicly available, the SEPLFs are underestimated, resulting in an optimistically sized aircraft. If the windmilling and trim drag were modeled, the selected power loadings from the constraint diagram would be smaller.

As the propulsion system failure severity increases, the required power loading monotonically decreases, resulting in tighter constraints that shift towards the bottom-left corner of the constraint diagram. At first, the observed behavior may seem to contradict that observed in Fig. 8; as the SEPLF (propulsion system failure severity) increases, the climb performance requirements are loosened, requiring a shallower climb gradient and enabling the propulsion system to be downsized. However, for increasing SEPLFs, the generalized engine inoperative correction factor increases, requiring the propulsion system size to grow. Thus, there is an interaction between the generalized requirements and engine inoperative correction factor constructed – increasing the propulsion system failure severity decreases the required climb gradients but increases the engine inoperative correction factor. As observed in Fig. 12, the constraints are tightened for an increasing propulsion system failure severity, shifting the contours towards the bottom-left of the constraint diagram and revealing that the engine inoperative correction factor's behavior dominates the interaction.

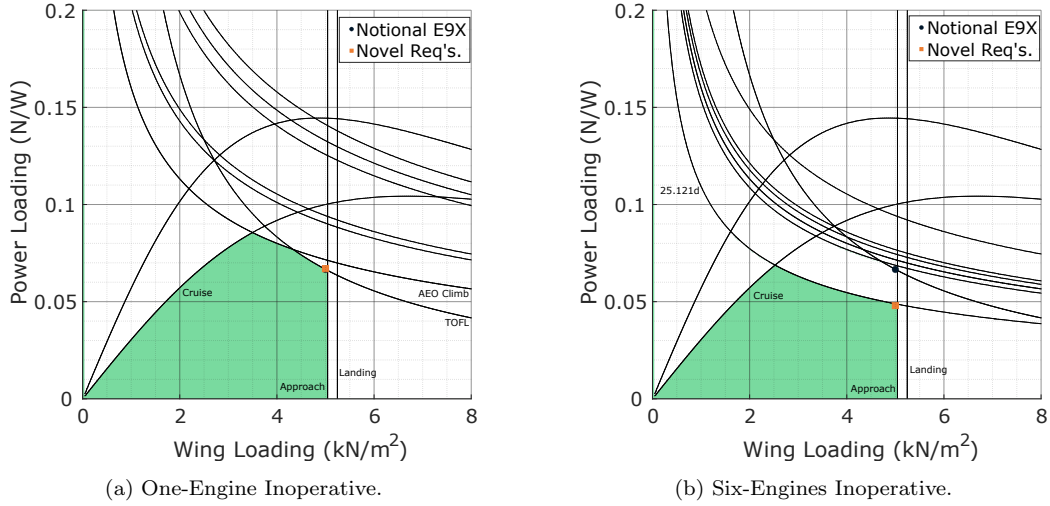


Figure 12: Constraint diagrams using the proposed climb requirements and engine inoperative correction factor. The feasible region is shaded in green.

For the 1-, 2-, and 3-engine inoperative scenarios, the generalized climb performance requirements are loosened relative to those constraining the notional E9X size, shifting them towards the upper-right corner of the constraint diagram. However, the takeoff field length (TOFL) and AEO climb constraints remain active, preventing a design point with a larger power loading from being selected. If these constraints were to be neglected, the power loading could be increased, possibly resulting in improved system-level performance, such as reduced MTOW or battery energy stored on the aircraft. Further investigation would be necessary to confirm that such system-level performance improvements are observed.

For the 4-, 5-, and 6-engine inoperative scenarios, the generalized climb performance requirements are tightened relative to those constraining the notional E9X size, shifting them towards the lower-left corner of the constraint diagram. As a result, the generalized Part 25.121d constraint becomes active while the TOFL and AEO climb constraints become inactive, increasing the required power loading and propulsion system size.

Using the newly identified design points for all novel engine inoperative scenarios, the notional E9X was resized in FAST. Table 15 reports the system-level performance parameters and electric motor size relative to the notional E9X. Since the windmilling and trim drag was not modeled, the

system-level benefits reported are optimistic, and would be worse if such aerodynamic considerations were accounted for.

Table 15: Performance parameters and weights for the redesigned E9Xs. Percentages listed are the relative difference with respect to the notional E9X.

Configuration	Power Loading (N/W)	MTOW (kg)	Battery Weight (kg)	Battery Energy (MWh)	Motor Weight (kg)	Motor Power (MW)	Time to Climb (min)
Notional E9X	0.067	76,047	35,985	12.95	322	1.61	19.47
1-Engine Inop.	0.067 (+0.00%)	76,047 (+0.00%)	35,985 (+0.00%)	12.95 (+0.00%)	322 (+0.00%)	1.61 (+0.00%)	19.47 (+0.00%)
2-Engines Inop.	0.067 (+0.00%)	76,047 (+0.00%)	35,985 (+0.00%)	12.95 (+0.00%)	322 (+0.00%)	1.61 (+0.00%)	19.47 (+0.00%)
3-Engines Inop.	0.067 (+0.00%)	76,047 (+0.00%)	35,985 (+0.00%)	12.95 (+0.00%)	322 (+0.00%)	1.61 (+0.00%)	19.47 (+0.00%)
4-Engines Inop.	0.064 (-3.90%)	77,139 (+1.44%)	36,596 (+1.70%)	13.17 (+1.66%)	340 (+5.65%)	1.70 (+5.65%)	18.16 (-6.71%)
5-Engines Inop.	0.056 (-15.92%)	83,171 (+9.37%)	39,897 (+10.87%)	14.36 (+10.85%)	419 (+30.20%)	2.09 (+29.89%)	14.65 (-24.74%)
6-Engines Inop.	0.048 (-27.93%)	88,777 (+16.74%)	42,726 (+18.73%)	15.38 (+18.72%)	521 (+61.89%)	2.61 (+62.20%)	12.83 (-34.09%)

Since no new sizing point was identified for the E9X resized under 1-, 2-, or 3-engine inoperative scenarios, no system-level benefits are attained. However, for the E9X resized under 4-, 5-, and 6-engine inoperative scenarios, system-level performance worsens as a result of the smaller power loadings selected from the constraint diagram. The optimal power loading identified from the constraint diagrams was between 0.048 and 0.064 N/W, resulting in a 4 to 28% decrease relative to the notional E9X, resulting in the electric motors' size increasing by 5 to 60%. The power loading decrease is observed because the engine inoperative correction factor is larger than that of the notional E9X, reflecting the higher propulsion system failure severity relative to legacy one-engine inoperative failures on conventional aircraft. As a result of the increased propulsion system size, the aircraft's MTOW increases, requiring more battery energy to be stored onboard the aircraft. The only system-level performance benefit from increasing the propulsion system size is that it takes less time to climb to cruise altitude during AEO operations.

More generally, if the selected propulsion system failure mode results in a SEPLF corresponding to an engine inoperative correction factor greater than the traditional $n/(n - 1)$ correction factor, then the aircraft's propulsion system size increases and system-level benefits will not be observed. Conversely, system-level benefits may be observed if the selected

propulsion system failure mode results in a SEPLF corresponding to an engine inoperative correction factor less than the traditional $n/(n - 1)$ correction factor. Thus, the failure mode chosen by the engineer significantly impacts the resultant aircraft size. Based on this observation, the failure mode may become a critical design variable during aircraft conceptual design, particularly for concepts leveraging advanced and/or distributed propulsion systems.

5.2.3. *Aside: Neglecting TOFL and AEO Climb Constraints*

The results presented in Fig. 12 and Tab. 15 revealed that system-level benefits were not observed if the notional E9X was resized for 1-, 2-, or 3-engine inoperative conditions because the TOFL and AEO Climb constraints remained active. For experimentation purposes only, the constraint diagrams for 1-, 2-, and 3-engine inoperative conditions were re-considered by neglecting the TOFL and AEO Climb constraints. If this were to occur, the generalized Part 25.121d constraint becomes active. As a result of neglecting these two constraints, the feasible region increases and larger power loadings are admitted, enabling the propulsion system size to decrease. The larger power loadings make it possible to realize system-level benefits such as an MTOW reduction or decreased battery energy stored onboard.

Table 16 lists the system-level performance parameters and electric motor size for the resized notional E9X under 1-, 2-, and 3-engine inoperative conditions while neglecting the TOFL and AEO Climb constraints. The corresponding power loadings selected for 1-, 2-, and 3-engine inoperative scenarios are 9 to 35% larger than that of the notional E9X, enabling the propulsion system to be downsized. Despite downsizing the propulsion system, system-level benefits are only observed in the configuration re-sized for 3-engine inoperative conditions, resulting in a 1.70% MTOW reduction and a 1.62% battery energy savings. System-level benefits are not realized for the configurations sized under 1- and 2-engine inoperative conditions because the time to climb to cruise altitude becomes prohibitively large, requiring additional battery energy to climb for an extended period of time.

Although primarily illustrative due to the neglected TOFL and AEO Climb constraints, the propulsion system failure mode still impacts the overall system-level performance. If the SEPLF causes the propulsion system size to be significantly downsized, then climb performance during nominal (AEO) operations is degraded, resulting in system-level performance

Table 16: Performance parameters and weights for the redesigned E9Xs. Percentages listed are the relative difference with respect to the notional E9X.

Configuration	Power Loading (N/W)	MTOW (kg)	Battery Weight (kg)	Battery Energy (MWh)	Motor Weight (kg)	Motor Power (MW)	Time to Climb (min)
Notional E9X	0.067	76,047	35,985	12.95	322	1.61	19.47
1-Engine Inop.	0.090 (+35.14%)	83,293 (+9.53%)	41,234 (+14.59%)	14.84 (+14.55%)	261 (-18.90%)	1.30 (-19.21%)	37.95 (+94.96%)
2-Engines Inop.	0.083 (+24.62%)	76,673 (+0.82%)	36,929 (+2.62%)	13.29 (+2.59%)	260 (-19.21%)	1.30 (-19.21%)	30.59 (+57.15%)
3-Engines Inop.	0.073 (+9.61%)	74,756 (-1.70%)	35,388 (-1.66%)	12.74 (-1.62%)	289 (-10.20%)	1.44 (-10.51%)	23.12 (+18.77%)

penalties. Such an observation further emphasizes the significance of the failure mode chosen by the engineer and its impact on the overall aircraft size and system-level performance during conceptual design.

6. Conclusions

Future aircraft concepts leveraging distributed propulsion are key enablers for more efficient flight, but face major certification gaps prior to entering service. One identified gap is the poor applicability of OEI performance requirements, which have only been defined in FAA Part 25 regulations for conventional aircraft utilizing two, three, or four gas turbine engines. Further challenges emerge for advanced aircraft concepts utilizing heterogeneously-sized propulsors, as the present regulations implicitly assume that all propulsors are homogeneously sized.

To overcome the poor applicability of presently published OEI performance requirements, generalized climb performance requirements were constructed and parameterized by the SEPLF, overcoming limitations on existing FAA Part 25 regulations parameterized by the number of engines installed on the airframe. The SEPLF accounts for the reduced thrust power and increased drag power from the propulsion system failure within a scalar quantity, and can be easily estimated for any aircraft concept and propulsion system. Additionally, the Sigmoid Curves' asymptotic behavior bounds the generalized flight performance requirements within the presently published FAA regulations, guaranteeing that future aircraft concepts are designed with flight performance requirements commensurate to those of conventional aircraft previously certified. The mapping constructed in this work is extensible, allowing others to utilize a different set of climb

performance requirements than the FAA-published ones used in this work. Overall, the generalized performance requirements constructed enhance the state-of-the-art by reducing the need for ad-hoc OEI requirement adaptations while ensuring consistent sizing of conventional and electrified transport aircraft with disparate propulsion system architectures.

The generalized climb performance requirements were also pivotal for validating prior assumptions made in the literature while sizing advanced aircraft concepts. In the absence of generalized climb performance requirements, it is reasonable to size an aircraft with a distributed propulsion system using the FAA’s 4-engine climb performance requirements, as its propulsion system failure severity is likely less than that of a conventional aircraft.

To support conceptual aircraft design studies, an engine inoperative correction factor was also constructed as a function of the SEPLF. The engine inoperative correction factor accommodates advanced propulsion systems with heterogeneously-sized propulsors, such as NASA’s SUSAN concept, and enhances the state-of-the-art relative to traditional $n/(n - 1)$ scale factors that assume all propulsors are homogeneously sized.

The generalized climb performance requirements and engine inoperative correction factor were then utilized to size a notional Elysian E9X under multiple novel propulsion system failure modes. If the primary propulsion system failure mode produces a smaller SEPLF than a conventional aircraft under a legacy one-engine inoperative failure mode, then system-level performance may improve if the climb performance constraints are active. Additional trade studies may be necessary to identify a propulsion system size that improves system-level performance while modestly penalizing climb performance during nominal (all engine operative) flight operations. The case study revealed that the primary failure mode chosen during advanced aircraft design significantly impacts system-level performance, particularly for aircraft leveraging advanced/distributed propulsion systems.

All results generated in this paper are reproducible using the publicly available data, code, and input files in Ref. [36]. Replicating these analyses requires specific versions of the FAST to be downloaded, and are available in the prior reference listed and on GitHub⁴. The earliest version of MATLAB

⁴<https://github.com/ideas-um/FAST>, Commit CE9DBE1, located on the Aerodynamic-FLOPSbased branch.

tested with this code was Version R2019b.

Four steps were identified to extend or enhance this research. First, the data collected to define the SEPLFs only consisted of turbofan-powered aircraft certified under FAA Part 25 regulations. Including data from turboprop-powered aircraft carrying 50-80 passengers, such as the ATR 42/72 families, would augment the existing dataset and refine the SEPLFs utilized to construct the generalized climb performance requirements and engine inoperative correction factor.

Second, the methodology developed in this work could be extended to normal category aircraft certified under FAA Part 23 requirements. Such requirements are currently prescribed by categorizing aircraft based on the number of engines installed, number of passengers carried, and their cruise speed (faster or slower than Mach 0.6). Applying the methodology described in this work could unify the existing flight performance requirements under a similar parameterization.

Third, the Sigmoid Curve constructions revealed that future climb performance requirements could be governed by an upper/lower limit depending on the propulsion system failure severity. Future work is necessary to identify the corresponding propulsion system severity threshold that would cause a switch between the upper/lower limits, better contextualizing the FAA-prescribed climb gradients for advanced aircraft and propulsion system design. Such a requirement formulation would directly impact 3-engine aircraft and their propulsion systems, as their current climb performance requirements are in-between the upper/lower limits published by the FAA. As a result, further studies should explore how the size of 3-engine aircraft would change if only the upper/lower climb performance requirements existed. Despite this area of future work, the Sigmoid Curve parameterization still remains as the currently preferred model because it is a value- and slope-continuous function and can be easily integrated into a multidisciplinary design optimization environment, particularly if derivatives are computed analytically.

Lastly, an additional study could be performed on an aircraft with heterogeneously-sized propulsors to explore how its size is impacted by varying the propulsion system failure mode. For example, on NASA's SUSAN aircraft, either the aft turbofan engine or part of its distributed electric propulsion system could be defined as the primary failure mode, resulting in different failure severities. A trade study could explore the sensitivity of SUSAN's system-level performance with respect to the flight performance

requirements and primary propulsion system failure mode.

Together, the generalized climb performance requirements and engine inoperative correction factor pave a pathway for sizing future aircraft concepts under conditions similar to those defined in FAA Part 25 regulations while providing engineers the flexibility to reap system-level benefits from advanced propulsion technologies.

Appendix A. Diversion Mission Profile Boundary Conditions

In Section 3.1, a diversion mission was flown to estimate the specific excess power loss after an engine failure. Table A.17 lists the boundary conditions for each segment flown. The engine was assumed to fail immediately after takeoff as the aircraft begins the first climb segment. Both the takeoff and climb speeds, V_{Tko} and V_{Clb} , respectively, were varied during the operational analysis.

Table A.17: Generic diversion mission profile. All airspeeds are reported as true airspeeds.

Segment	Initial Altitude (m)	Final Altitude (m)	Initial Airspeed (m/s)	Final Airspeed (m/s)
Takeoff	0	0	0	V_{Tko}
Climb	0	914	V_{Tko}	V_{Clb}
Climb	914	3,048	V_{Clb}	128.6
Cruise	3,048	3,048	128.6	128.6
Descent	3,048	0	128.6	$1.1V_{\text{Tko}}$
Landing	0	0	$1.1V_{\text{Tko}}$	0

Appendix B. Lift-to-Drag Correction Factors

In Section 3.1, a correction factor was imposed on the cruise and climb/descent lift-to-drag ratios to account for windmilling and trim drag caused by an engine failure. Tables B.18 and B.19 list the correction factors for 2- and 4-engine aircraft, respectively, under the defined operational strategies. The correction factors for 3-engine aircraft were computed as the mean of each corresponding correction factor for 2- and 4-engine aircraft.

Table B.18: Lift-to-drag ratio correction factors for 2-engine aircraft under one engine inoperative conditions. ΔV represents the difference between the takeoff and climb speeds, previously outlined in Tab. 4.

V_{Tko} (m/s)	ΔV (m/s)					
	0.00	5.14	10.29	15.43	20.58	25.72
66.88	0.8091	0.8084	0.8082	0.8039	0.8043	0.8021
72.02	0.8082	0.8082	0.8047	0.8027	0.8004	0.8030
77.17	0.8054	0.8036	0.8016	0.7993	0.8050	0.8000
82.31	0.8029	0.8009	0.7986	0.7954	0.7999	0.7979
87.46	0.8118	0.8045	0.8011	0.7993	0.7978	0.7962
92.60	0.7988	0.8012	0.7994	0.7980	0.7964	0.7873
97.74	0.8016	0.7998	0.7984	0.7967	0.7876	0.7791
102.89	0.8005	0.7990	0.7973	0.7880	0.7802	0.7731

Table B.19: Lift-to-drag ratio correction factors for 4-engine aircraft under one engine inoperative conditions. ΔV represents the difference between the takeoff and climb speeds, previously outlined in Tab. 4.

V_{Tko} (m/s)	ΔV (m/s)					
	0.00	5.14	10.29	15.43	20.58	25.72
66.88	0.8244	0.8386	0.8607	0.7676	0.7628	0.8554
72.02	0.8571	0.7590	0.7678	0.7629	0.7558	0.8846
77.17	0.8733	0.8455	0.8349	0.8310	0.8284	0.8263
82.31	0.8498	0.8329	0.8295	0.8271	0.8251	0.8233
87.46	0.8324	0.8290	0.8268	0.8248	0.8231	0.8214
92.60	0.8290	0.8267	0.8249	0.8231	0.8214	0.8110
97.74	0.8268	0.8249	0.8232	0.8215	0.8110	0.7954
102.89	0.8253	0.8236	0.8219	0.8113	0.7958	0.7826

Appendix C. Trim Drag Modeling

The trim drag model is a first-order, physics-based model to predict the parasite drag coefficient increment corresponding to the necessary rudder input to maintain lateral stability during one engine inoperative conditions.

First, the yaw moment required from the vertical stabilizer is computed in Eq. C.1, considering both the thrust from the remaining operative engine, T , and the windmilling drag from the inoperative engine, D_{Windmill} , multiplied by the moment arm between the engines and the fuselage centerline, r .

$$N_v = (T + D_{\text{Windmill}}) r \quad (\text{C.1})$$

Next, Eq. C.2 computes the yaw moment coefficient, and is a function of the wing area and span.

$$C_n = N_v / (q \times S_w \times b_w) \quad (\text{C.2})$$

Then, the lift curve slope of a two-dimensional airfoil, $C_{l_{\alpha v}}$, on the vertical tail was estimated as 90% of that of a thin airfoil ($2\pi / \text{rad}$). Using this information, the lift curve slope for the vertical tail (converted to units of $/\text{deg}$) was predicted in Eq. C.3.

$$C_{L_{\alpha v}} = \frac{C_{l_{\alpha v}}}{1 + \frac{C_{l_{\alpha v}}}{\pi A R_{vev}}} \times \frac{\pi}{180} \quad (\text{C.3})$$

After that, the flapped area fraction was computed as the ratio of the rudder area with respect to the vertical tail area, S_r/S_v , and input into a digitized version of Figure 2.21 in Nelson’s *Flight Stability and Automatic Control* [46], returning the “flap effectiveness parameter”, τ .

With this information, the yaw moment coefficient for the rudder deflection was computed in Eq. C.4, assuming that the dynamic pressure on the vertical tail is equal to that on the wing ($\eta = 1$). Note that l_v is the moment arm between the wing and tail mean aerodynamic chords.

$$C_{n_{\delta r}} = \eta \frac{l_v S_v}{S_w b_w} C_{L_{\alpha v}} \tau \quad (\text{C.4})$$

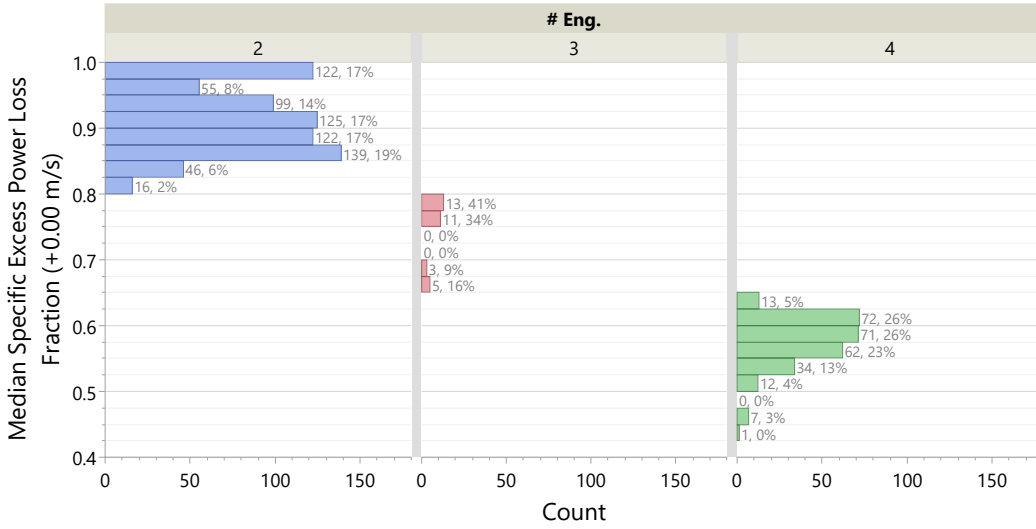
Lastly, the rudder deflection was computed using Eq. C.5. The result was input into Eq. 12.37 from Raymer’s *Aircraft Design: A Conceptual Approach* [47] to predict the parasite drag increment from deflecting the rudder (Eq. C.6).

$$\delta r = C_n / C_{n_{\delta r}} \quad (\text{C.5})$$

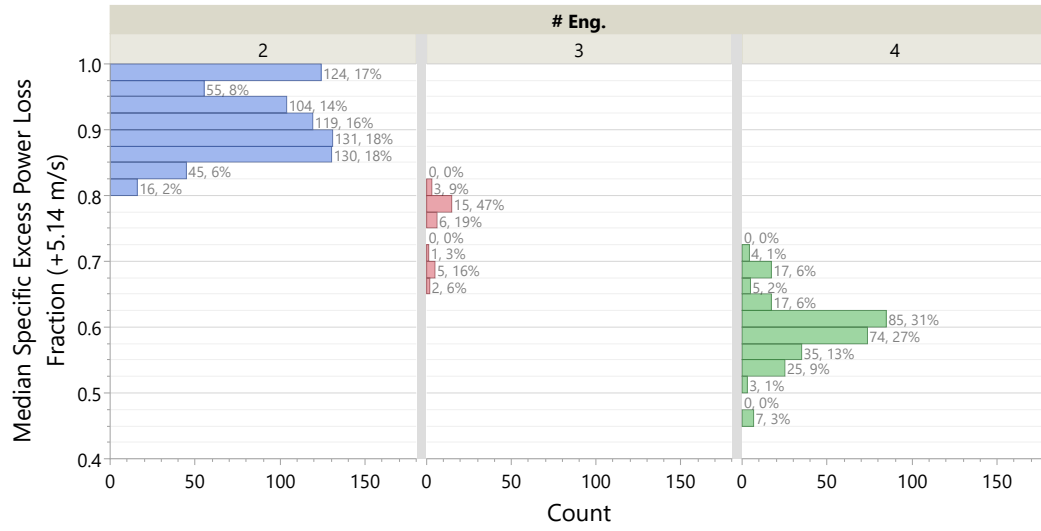
$$\Delta C_{D_0} = 0.0023 \delta r (b_r / b_w) \quad (\text{C.6})$$

Appendix D. Specific Excess Power Loss Fractions

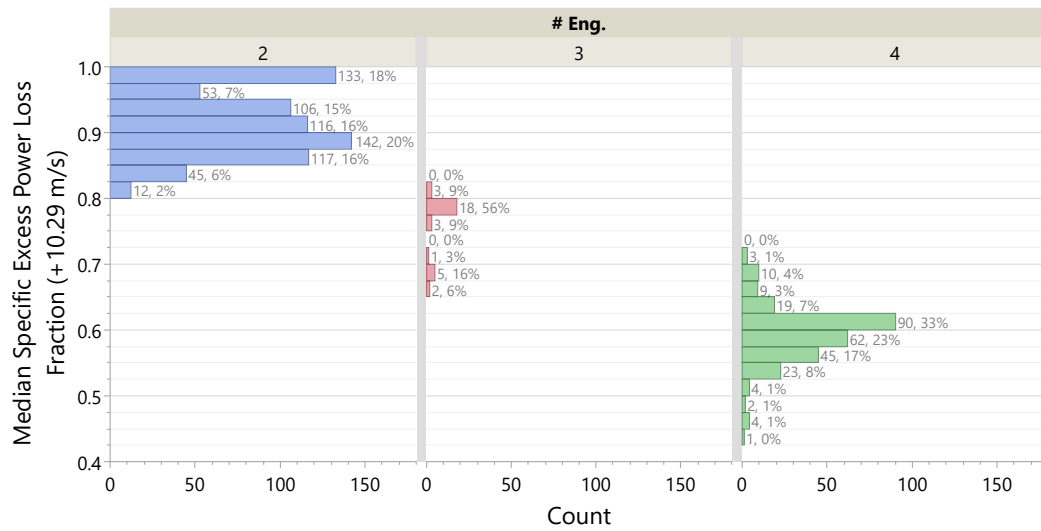
In Section 3.2, a histogram of the median SEPLF was plotted for a ΔV of 25.72 m/s. Additional histograms for ΔV between 0.00 and 20.58 m/s are included in Fig. D.13.



(a) $\Delta V = 0.00$ m/s.



(b) $\Delta V = 5.14$ m/s.



(c) $\Delta V = 10.29$ m/s.

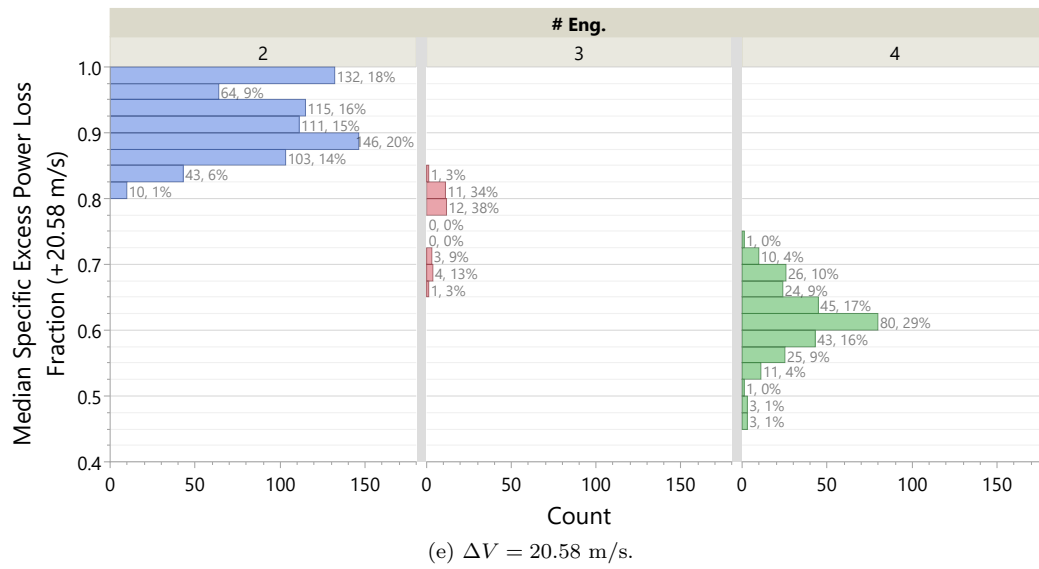
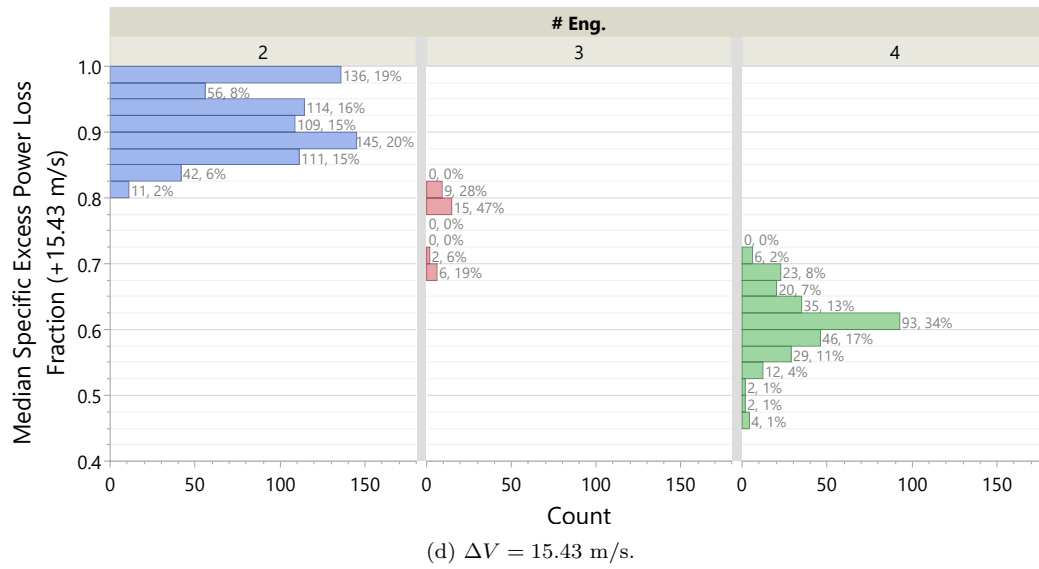


Figure D.13: SEPLF distributions.

As done in Tab. 7 within Section 3.2, the distributional anchors for the remaining ΔV are included in Tabs. D.20 through D.24.

Table D.20: Selected quantities across the SEPLF distributions for $\Delta V = 0.00$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8532	0.6683	0.5339
25	0.8720	0.6984	0.5567
50	0.9099	0.7692	0.5812
75	0.9489	0.7842	0.6034
90	0.9816	0.7859	0.6165

Table D.21: Selected quantities across the SEPLF distributions for $\Delta V = 5.14$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8544	0.6754	0.5435
25	0.8737	0.7222	0.5735
50	0.9113	0.7796	0.5976
75	0.9494	0.7861	0.6132
90	0.9822	0.8091	0.6392

Table D.22: Selected quantities across the SEPLF distributions for $\Delta V = 10.29$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8556	0.6780	0.5448
25	0.8753	0.7170	0.5653
50	0.9128	0.7827	0.5989
75	0.9512	0.7876	0.6155
90	0.9824	0.8072	0.6410

Table D.23: Selected quantities across the SEPLF distributions for $\Delta V = 15.43$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8585	0.6806	0.5558
25	0.8769	0.7252	0.5855
50	0.9144	0.7864	0.6099
75	0.9536	0.8010	0.6337
90	0.9826	0.8175	0.6771

Table D.24: Selected quantities across the SEPLF distributions for $\Delta V = 20.58$ m/s.

Percentile	Engines Installed		
	2	3	4
10	0.8571	0.6816	0.5601
25	0.8766	0.7281	0.5924
50	0.9141	0.7812	0.6146
75	0.9557	0.8149	0.6442
90	0.9829	0.8222	0.6847

Appendix E. Climb Gradient Comparisons

In Section 4.2.1, a test was performed to compare the required climb gradients output by the generalized and FAA Part 25 flight performance requirements. Figure 9 in Section 4.2.1 reported histograms for 2-engine aircraft only. Additional histograms for 3- and 4-engine aircraft are illustrated in Figs. E.14 and E.15, respectively.

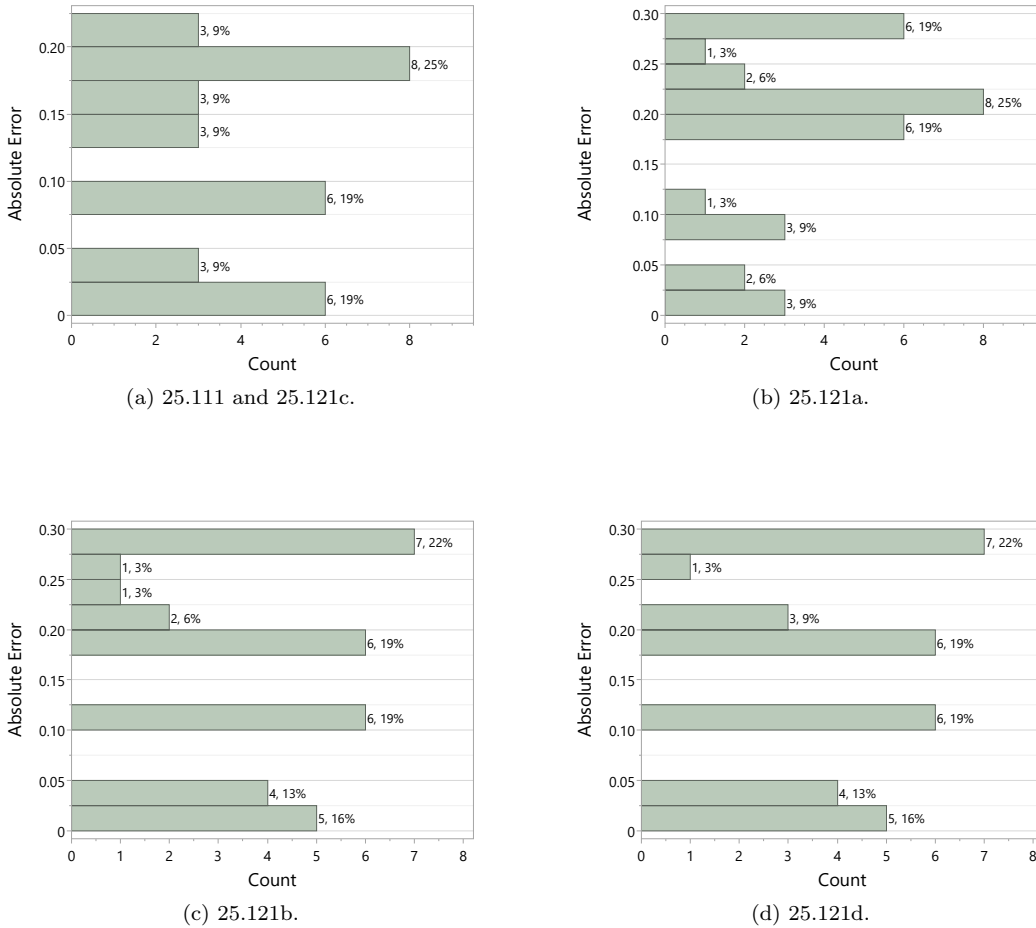
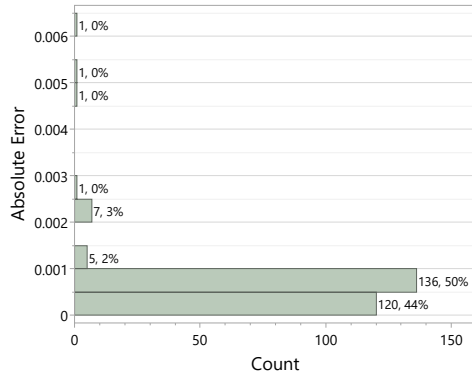
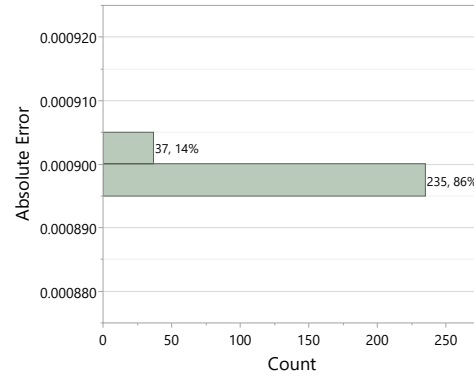


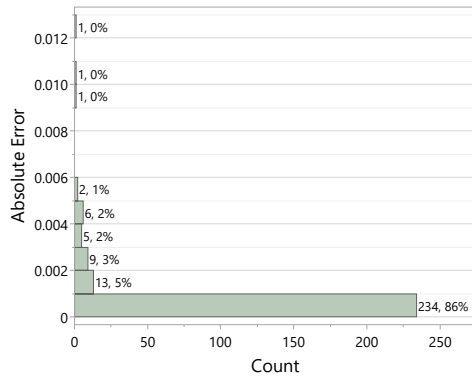
Figure E.14: Absolute error between FAA and proposed flight performance requirements for 3-engine aircraft.



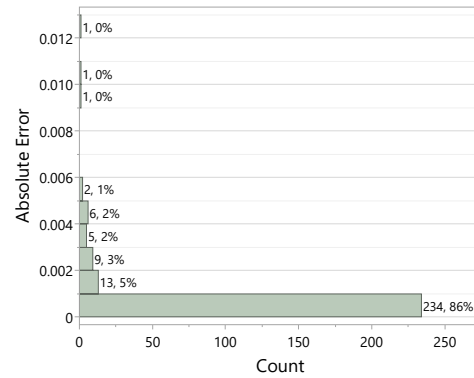
(a) 25.111 and 25.121c.



(b) 25.121a.



(c) 25.121b.

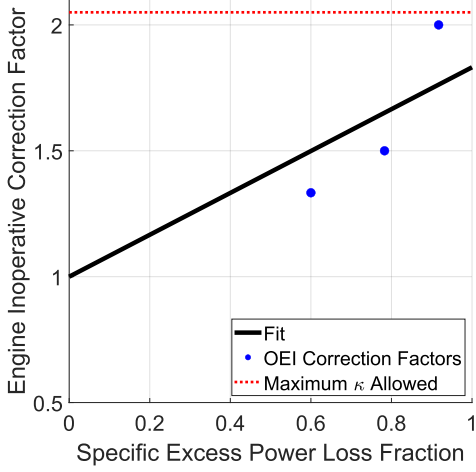


(d) 25.121d.

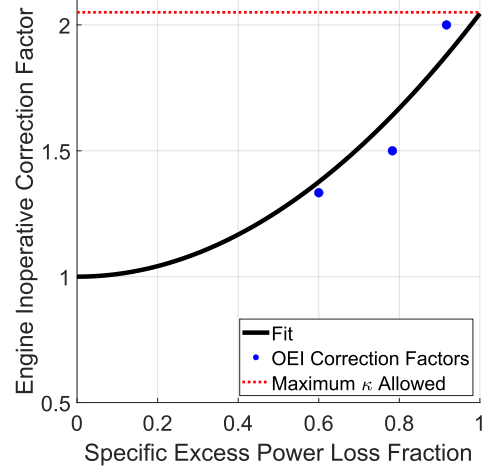
Figure E.15: Absolute error between FAA and proposed flight performance requirements for 4-engine aircraft.

Appendix F. Alternative Engine Inoperative Correction Factor Function Constructions

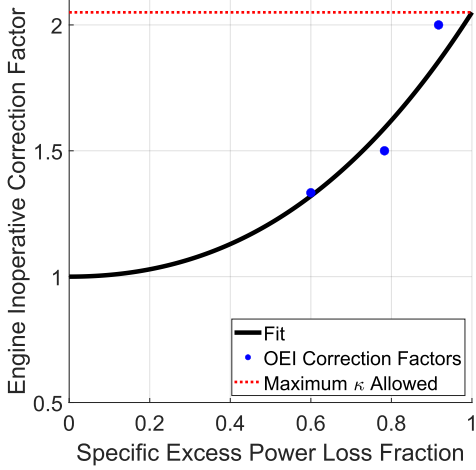
In Section 4.3.1, curve fits for defining the engine inoperative correction factor were generated by replacing the inequality constraint of $\hat{\kappa}(1) \leq 2.25$ with $\hat{\kappa}(1) \leq 2.05$. The resulting curve fits are illustrated in Fig. F.16.



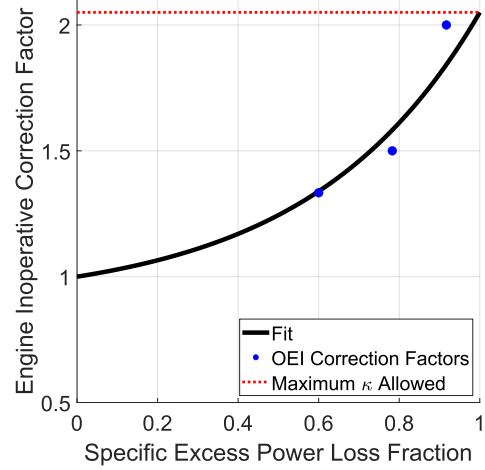
(a) Linear function.



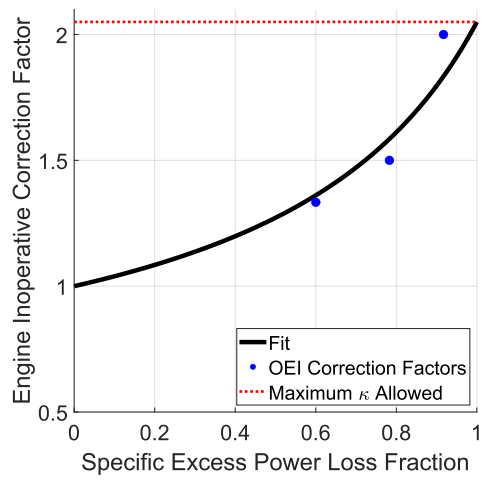
(b) Quadratic function.



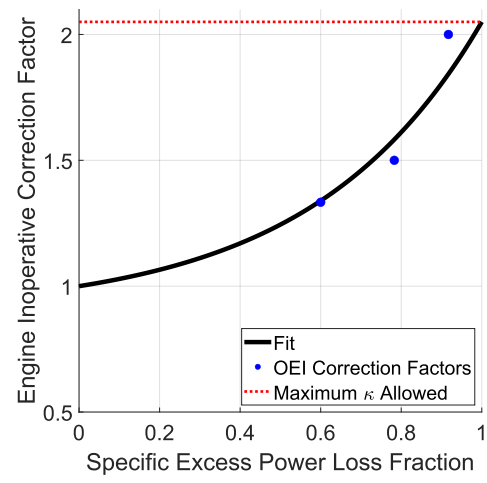
(c) Cubic function.



(d) Exponential function.



(e) Tangent function.



(f) Hyperbolic sine function.

Figure F.16: Candidate engine inoperative correction factor curve fits.

Appendix G. Notional Elysian E9X Modeling

A notional Elysian E9X (hereinafter referred to as the “E9X”) was modeled in Section 5 to demonstrate how an aircraft is sized using the proposed performance requirements and engine inoperative correction factor. This appendix lists the modeling parameters and assumptions made during the various modeling stages.

Appendix G.1. Sizing in FAST

First, Tab. G.25 lists the parameters utilized to size the notional E9X in FAST [34]. In addition to the modeling information provided in Section 5, three assumptions were made:

1. The takeoff speed was assumed to be 10% faster than the stall speed.
2. The maximum rate of climb was set as a heuristic to ensure passenger comfort.
3. The climb/descent lift-to-drag ratio was approximated as 80% of the cruise lift-to-drag ratio.

The notional E9X was sized using the design mission profile specified by the boundary conditions in Tab. G.26.

Table G.25: Notional E9X sizing parameters. Values with asterisks were either assumed or modeled using information outside of Ref. [26].

Parameter	Value	Units
Takeoff Altitude*	0	m
Cruise Altitude	7,600	m
Takeoff Speed*	62.857	m/s
Approach Speed	74.594	m/s
Cruise Speed	0.600	Mach
Design Range	1,000	km
Maximum Rate of Climb*	11.43	m/s
Maximum Takeoff Weight	76,000	kg
Airframe Weight Calibration Factor	0.829	—
Cruise Lift-to-Drag Ratio	23.00	—
Climb/Descent Lift-to-Drag Ratio*	18.40	—
Wing Loading	509.68	kg/m ²
Power Loading	0.0666	N/W
Propeller Efficiency	87	%
Electric Motor Efficiency	95	%
Battery Efficiency	95	%
Battery Gravimetric Specific Energy	360	Wh/kg
Electric Motor Power-to-Weight Ratio	5	kW/kg

Table G.26: Assumed notional E9X design mission profile. Airspeeds listed in m/s are denoted as true or equivalent airspeeds.

Segment	Initial Altitude [m]	Final Altitude [m]	Initial Airspeed [m/s or Mach]	Final Airspeed [m/s or Mach]
Takeoff	0	0	0.00 (TAS)	62.86 (TAS)
Climb	0	914	62.86 (TAS)	102.89 (EAS)
Climb	914	7,600	102.89 (EAS)	102.89 (EAS)
Acceleration	7,600	7,600	102.89 (EAS)	M0.60
Cruise	7,600	7,600	M0.60	M0.60
Deceleration	7,600	7,600	M0.60	102.89 (EAS)
Descent	7,600	914	102.89 (EAS)	102.89 (EAS)
Descent	914	0	102.89 (EAS)	74.59 (TAS)
Landing	0	0	74.59 (TAS)	0.00 (TAS)

Appendix G.2. Constraint Diagram Modeling

Constraint diagrams were generated to assess the impacts of the proposed performance requirements and engine inoperative correction factor. The parameters used to model the E9X are included in Tab. G.27. With exception to the constraint modeling differences discussed in Section 5, all other constraints were modeled using the procedure provided in Ref. [26].

Table G.27: Notional E9X constraint diagram modeling parameters. Values with asterisks were either assumed or modeled using information outside of Ref. [26].

Parameter	Value	Units
Takeoff Altitude*	0	m
Cruise Altitude	7,600	m
Service Ceiling	8,000	m
Stall Speed	57.143	m/s
Takeoff Speed*	62.857	m/s
Approach Speed	74.594	m/s
Cruise Speed	0.600	Mach
Diversion Speed	0.400	Mach
Takeoff Field Length	2,000	m
Landing Field Length	2,000	m
Maximum Takeoff Weight	76,000	kg
Landing Weight Fraction	1.00	—
Aspect Ratio	12.00	—
Takeoff Lift Coefficient	2.50	—
Cruise Lift Coefficient*	1.00	—
Landing Lift Coefficient	3.10	—
Takeoff Parasite Drag Coefficient*	0.0618	—
Cruise Parasite Drag Coefficient*	0.0168	—
Landing Parasite Drag Coefficient*	0.1068	—
Takeoff Oswald Efficiency*	0.7468	—
Cruise Oswald Efficiency*	0.7861	—
Landing Oswald Efficiency*	0.7075	—
Cruise Power Calibration Factor	0.10	—
Diversion Power Calibration Factor	0.28	—

Given the limited public information available about the E9X, multiple

values were either modeled using first-order estimations or assumed to be a fixed value.

Cruise Lift Coefficient Estimation. The cruise lift coefficient was varied until the constraint contour for FAA Part 25.121c reasonably matched that of the constraint diagram in the literature [26], and was estimated to be 1.0. As a result of the limited geometric information known about the aircraft, the cruise lift coefficient is likely overestimated to ensure that the contours on the constraint diagram reasonably matched those in literature.

Parasite Drag Coefficient Estimation. The parasite drag coefficients were estimated using Raymer’s Equivalent Skin Friction Method for transport aircraft [47]. This approach assumes that the wetted area is a function of the maximum takeoff weight, $S_{\text{wet}} = 10^c (\text{MTOW})^d$, where $c = 0.0199$, and $d = 0.7531$. The cruise parasite drag coefficient is then estimated as $C_{D_0} = C_f S_{\text{wet}} / S_{\text{ref}}$, with $C_f = 0.003$. The takeoff and landing parasite drag coefficients were estimated by adding 0.05 and 0.10, respectively, to the cruise parasite drag coefficient to account for flap deflections and landing gear deployment.

Oswald Efficiency Factor Estimation. To estimate the Oswald efficiency factor, the authors of Ref. [26] utilize Obert’s *Aerodynamic Design of Transport Aircraft* (Fig. 40.36) [48]. In this figure, there are three equations relating the aspect ratio to the Oswald efficiency factor. Since the E9X is a transformative flight concept, the most aggressive (drag-minimizing) relation was selected: $e = (1.05 + 0.0070\pi\mathcal{AR})^{-1}$. Since the E9X’s fuselage is shorter relative to its wingspan, the authors of Ref. [26] add 0.025 to the Oswald efficiency factor after using the previously selected equation. The climb and landing Oswald efficiency factors were obtained by multiplying the cruise Oswald efficiency factor by 95% and 90%, respectively, to account for flap deflections and landing gear deployment.

Cruise and Diversion Constraints. As discussed in Section 5, the cruise and diversion constraints were scaled by a factor of $1/\sigma^a$, where σ is the density ratio at the respective altitude, and a is a user-selected parameter to tune the constraint. For the cruise and diversion constraints, a was tuned to be 0.1 and 0.2, respectively. These values are reasonably close to 0, highlighting the minimal tuning required for these constraints, especially since the E9X is powered by electric motors that do not lapse.

Such scale factors, however, are typically used to account for gas turbine engine lapsing. Although the E9X is a battery-electric aircraft, utilizing the tuning parameter enables the constraints to be seamlessly adapted to propulsion systems utilizing gas turbine engines, especially if the E9X's Reserve Energy System is modeled in future work.

Appendix G.3. Constraint Diagram Validation

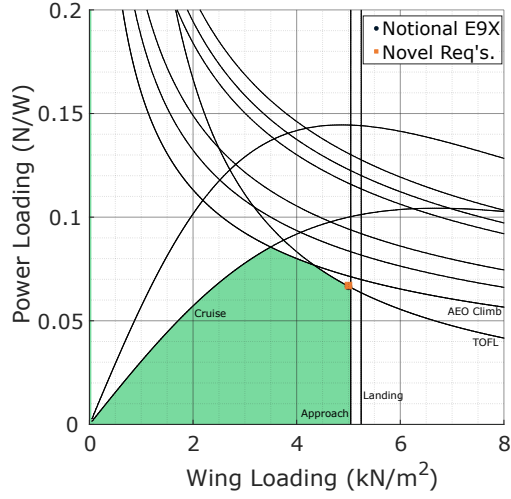
In Section 5, the results from Tab. 14 validated the constraint diagram at a wing loading of 4 kN/m². Table G.28 validates the constraint diagram at a wing loading of 6 kN/m².

Table G.28: Constraint diagram comparison at a 6 kN/m² wing loading.

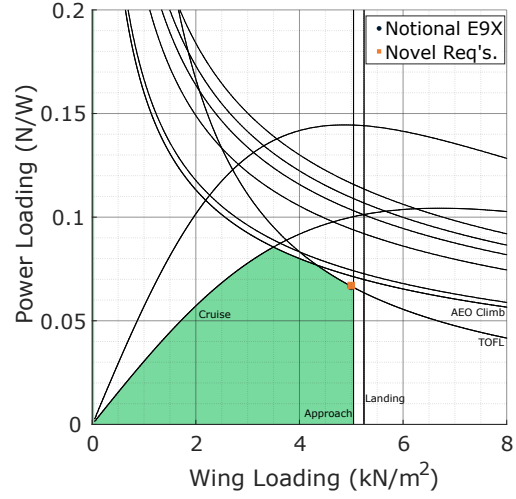
Constraint	Model	Literature	Absolute Error	Relative Error
FAA 25.121a	0.1175	0.1363	-0.0188	-13.793%
FAA 25.121b	0.1045	0.1063	-0.0018	-1.693%
FAA 25.121c	0.1105	0.1201	-0.0096	-7.993%
FAA 25.121d	0.0750	0.0797	-0.0047	-5.897%
FAA 25.119	0.0870	0.0856	+0.0014	+1.636%
AEO Climb	0.0660	0.0662	-0.0002	-0.302%
Cruise	0.1050	0.1040	+0.0010	+0.961%
Diversion	0.1433	0.1445	-0.0012	-0.830%
Takeoff Field Length	0.0560	0.0505	+0.0055	+10.891%

Appendix G.4. Constraint Diagrams under Engine Inoperative Conditions

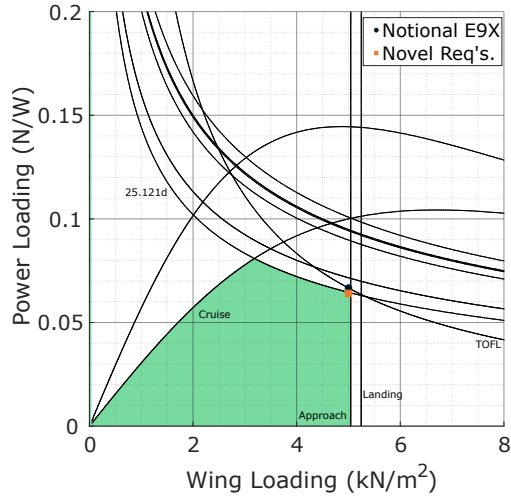
In Section 5, Fig. 12 illustrated the constraint diagrams for 1- and 6-engine inoperative conditions. Figure G.17 illustrates the constraint diagrams for 2-, 3-, 4-, and 5-engine inoperative conditions.



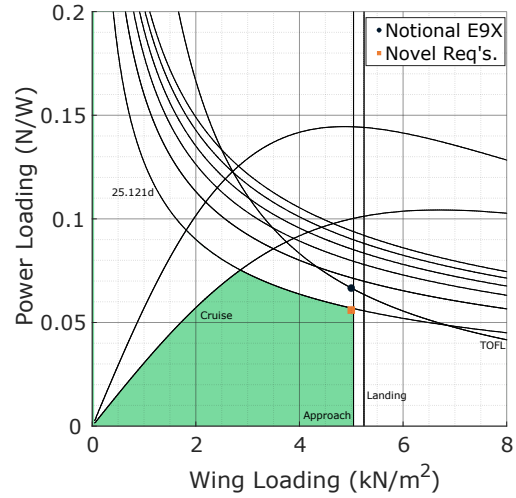
(a) Two-Engines Inoperative.



(b) Three-Engines Inoperative.



(c) Four-Engines Inoperative.



(d) Five-Engines Inoperative.

Figure G.17: Constraint diagrams using the proposed climb requirements and engine inoperative correction factor. The feasible region is shaded in green.

Funding Sources

This work was supported by the NASA Aeronautics Research Mission Directorate and Electrified Powertrain Flight Demonstration project, “Development of a Parametrically Driven Electrified Aircraft Design and Optimization Tool”, Glenn Engineering and Research Support Contract (GEARS) Contract 80GRC020D0003.

Declaration of Interest Statement

There are no additional relationships, patents, or activities to disclose.

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