



Center for High-Efficiency Electrical Technologies for Aircraft: Phase I Final Report

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Prepared under Cooperative Agreement 80NSSC19M0125

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Acknowledgments

This work was supported by NASA as part of the Center for High-Efficiency Electrical Technologies for Aircraft (CHEETA) under Cooperative Agreement 80NSSC19M0125. The authors would like to thank NASA for their support of this work, and in particular we would like to gratefully recognize the staff and administrators of the University Leadership Initiative. A significant number of individuals contributed to the work covered by this report, spanning an array of institutions and technical disciplines. While the authors of this report represent the senior investigators from each respective institution, a significant proportion of intellectual contributions were drawn broadly from work of students, postdoctoral researchers, and other research staff, and we would also like to recognize these individuals. From the University of Illinois Urbana-Champaign: Shivang Agarwal, Thanatheepan Balachandran, Arijit Banerjee, Ghanendra Das, Joshua Feldman, Himavath Jois, Matthew Lauer, Karthik Mahesh, Prateek Ranjan, John Reband, Noah Salk, Elias Waddington, and Jianqiao Xiao. From Boeing Research & Technology: Chellappa Balan, James Falcone, Shengyi Liu, and Tina Stoia. From Chicago State University: Valerie Goss and Archie Peters. From General Electric: Constantinos Minas. From Massachusetts Institute of Technology: Durgesh Chandel and Andrew White. From The Ohio State University: Christopher Kovacs. From The Pennsylvania State University: Dalton Decerio and Matthew Ibasitas. From Rensselaer Polytechnic Institute: Abhijit Khare and Meagan Podlaski. From Stony Brook University: Samuel Defaz and Mustafeez-ul-Hassan. From University of Arkansas: Md Maksudul Hossain, Asif Imran Emon, Rosten Sweeting, Arman Ur Rashid, Yuqi Wei, and Zhao Yuan. From University of Dayton Research Institute: Bang-Hung Tsao.

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Summary

Under the University Leadership Initiative (ULI), the Center for High-Efficiency Electrical Technologies for Aircraft (CHEETA) was established to develop and mature early-stage technologies pertaining to H₂-electric power and energy systems for aircraft. In particular, integration of these technologies on an aircraft system is envisioned to leverage the high specific energy content of liquid hydrogen (LH₂) with fuel cell energy conversion and an electrically driven ducted fan system to provide an ultra-efficient propulsion drivetrain. For this concept, the LH₂ system is not just used as an energy storage mechanism, but also as a cryogen to enable highly efficient superconducting (SC) electric systems. The end result of this concept is an integrated aircraft system with a quiet, efficient propulsion architecture that produces zero CO₂, NO_x, SO_x, and particulate matter emissions at the vehicle level.

Research outcomes of this program have led to the development of numerous leaps in H₂-electric aircraft technology, including

- H₂ energy systems
 - Next-generation composite-reinforced tanks
 - Multitank LH₂ management

*At Georgia Institute of Technology, Atlanta, Georgia, since 2022.

- LH₂ flow systems and engineering controls
- Advanced thermal management strategies for H₂-electric power systems
- Aerodynamics and propulsion technology
 - Optimization tools for the coupled design of aero-propulsive systems
 - Powered lift concepts for low-speed operations
 - Boundary layer integration systems to reduce power requirements for propulsion
 - Swirl and distortion-tolerant fan designs
- Cryogenic electrical power systems
 - Ultra-efficient, flight-weight SC electric machine technology
 - Lossless and low-voltage SC power transmission system
 - Cryogenic power components and integrated electronics

The integration of these technologies into a future vision vehicle aircraft are anticipated to meet the performance (payload, range) of current transport-class aircraft while consuming less energy and producing zero direct emissions at the aircraft level. The SC electrical power system uniquely enables the capabilities of this platform, as the use of cryogenic power systems results in electrical drivetrain losses below 2%, as well as component rated and specific power capabilities in excess of 2.5 MW and 22 kW/kg, respectively. Further maturation and demonstration of these technologies will provide an enabling capability to future zero-emissions flight vehicles and concepts.

1.0 Introduction and Program Goals

The aeronautics industry has been challenged on many fronts to increase efficiency, reduce emissions, and decrease dependency on petroleum-based fuels. In 2018, for the first time, CO₂ emissions directly attributable to the global aviation industry exceeded a production rate of 1 billion metric tons per year (Ref. 1). The increased concentration of CO₂ (Ref. 2) across the atmosphere has demonstrated unquestionable correlations with global surface temperature anomalies, resulting in logical conclusions that link anthropogenic activities to changes observed in the global climate (Ref. 3). While aviation only constitutes 2% to 3% of total annual global CO₂ emissions, the net environmental effect of hydrocarbon use in aviation is further complicated by the short-term and long-term effects of NO_x and contrail- or aviation-induced cirrus cloud formations. Together, these emissions contribute to an effective radiative forcing impact predicted to be greater than or comparable to that of CO₂, but to a degree that is difficult to precisely quantify (Ref. 1). While the overall efficiency and carbon intensity of aircraft have demonstrated dramatic improvements across previous decades (Ref. 4), growth of the air transportation sector has generally outpaced technology developments, leading to a steady rise in greenhouse gas emissions attributable to aviation that is likely to accelerate. With a mid-term growth of air traffic at an annual rate of 3.6% to 3.9% projected by both Boeing (Ref. 5) and Airbus (Ref. 6), the goal of a zero-emissions aviation future by 2050 (Ref. 7) necessitates a fundamental transition to new approaches to power and energy for aviation.

For these reasons, the current University Leadership Initiative (ULI) research team has been tasked with the development of novel technologies for electrified aircraft propulsion systems. A graphical overview of the various technical research aspects of the ULI study is shown in Fig. 1. These technologies are motivated by potential use on a fully electric commercial transport-class aircraft that utilizes a H₂ energy storage system and hybrid fuel cell/battery power system. While the high specific energy of H₂ makes it compelling for energy storage in aviation, the low energy density produces challenging volumetric requirements. For this reason, cryogenic LH₂ is envisioned to reduce energy storage volume.

However, since the energy storage medium is already cryogenic, it can also be used as an enabling medium to promote a superconducting (SC) electrical power system. Use of SC technologies permits large increases in the rated power and power density of electrical components, making them highly desirable for large-scale aircraft propulsion applications. The interconnectivity of the component-level research is also exemplified through the power and propulsion system schematic overview shown in Fig. 2 with a rendering of the aircraft concept.

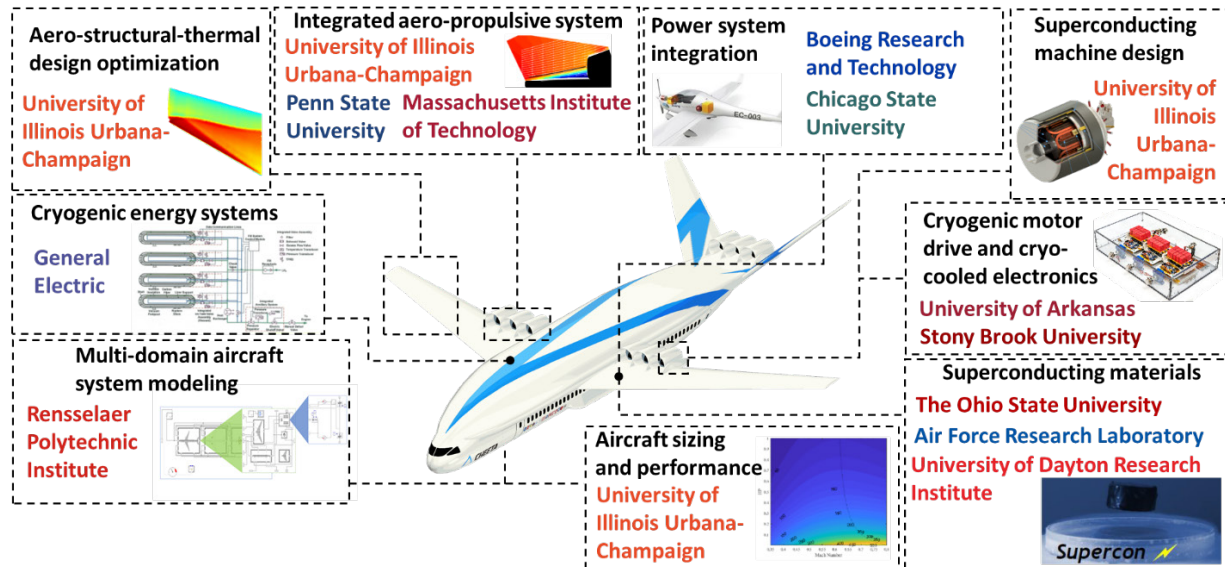


Fig. 1.—Summary of technical research contributions of CHEETA ULI team members.

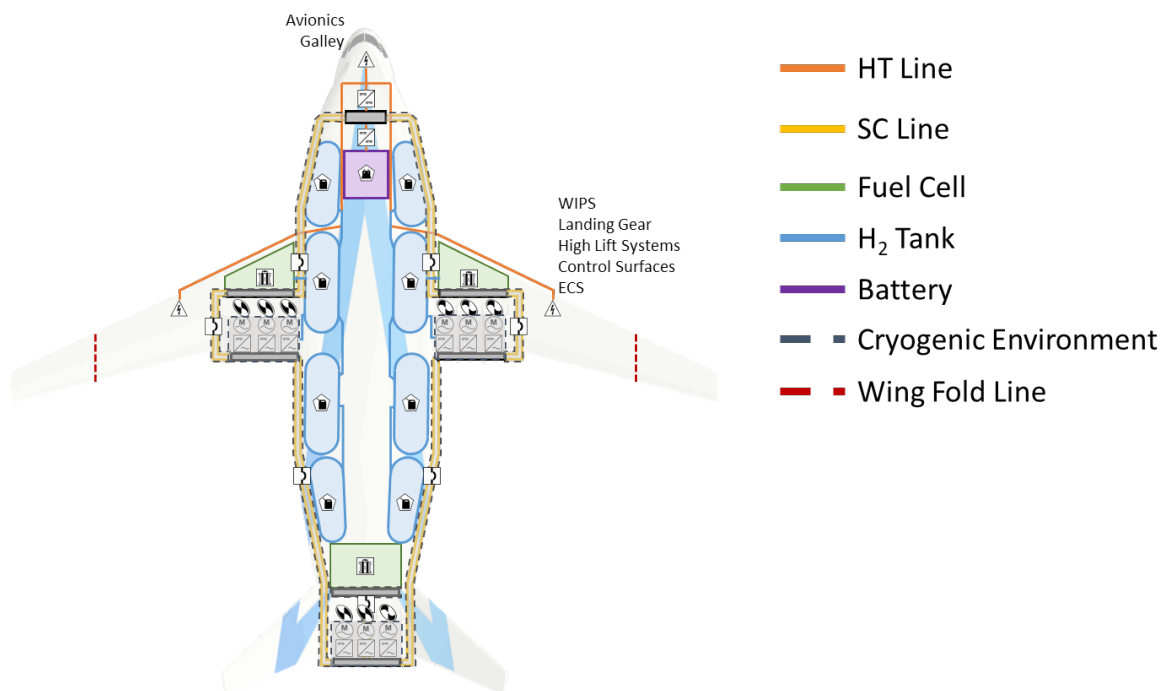


Fig. 2.—Interconnectivity of components across electric aircraft power and propulsion system. Wing ice protection system, WIPS; environmental control system, ECS.

2.0 Aero-Propulsive System Technology

Throughout the CHEETA research program, several key technologies were studied, matured, and integrated into the future flight aircraft concept. In particular, advancements were made in the areas of distributed electric propulsion (DEP) system design, fan designs for electric machine drive systems and boundary-layer ingestion (BLI), and aeroelastic multidisciplinary design optimization (MDO) using high-fidelity computational methods.

2.1 Distributed Electric Propulsion

As described by Kim et al. (Ref. 8), DEP represents a strategy for propulsion integration wherein a multitude of electrically driven propulsors are integrated across an aircraft concept for a purposeful benefit to the system-level performance. Multiple benefits of DEP approaches are discussed within this section, though the contributions of this work have revealed new strategies for advanced design of these highly airframe-integrated propulsion systems.

2.1.1 Aero-Propulsive Design Strategies

The design of aero-propulsive systems is complicated by the two-way coupling of external aerodynamic flows (i.e., those across airfoil or wing surfaces) and internal propulsive flows (i.e., those within a propulsive streamtube). Historical aerodynamic analysis strategies have lacked a practical way of codesigning highly integrated aero-propulsive systems, which is further addressed through contributions of the current study.

BLI and distributed propulsion (DP) are currently being considered as potential aircraft technologies of the future. BLI allows for higher propulsive efficiencies by strategically placing propulsors in the flow path of boundary layers, typically by embedding the propulsors within, or in close proximity to, aerodynamic surfaces. The propulsors ingest the low-kinetic-energy flow in the boundary layer, along with the high-kinetic-energy inviscid flow, which lowers the average kinetic energy entering the propulsion system. This approach thus decreases the net work required to achieve a desired thrust and increases propulsive efficiency. However, there are potential challenges to the implementation of BLI systems on aircraft, namely the flow-distortion effects on the fan operation and complications in the governing propulsor flow physics, both of which have been subject to prior study but remain open areas of investigation (Refs. 9 to 16).

DP offers improved safety through redundancy, permits propulsion-controlled aircraft maneuvering, and increases efficiency through direct filling of the aircraft wake (Ref. 17). Several recent studies have increased understanding of the complex aerodynamic effects and the net aero-propulsive system efficiency gains associated with DP (Refs. 10, 17 to 19). Many aircraft concepts have been studied that take advantage of BLI, DP, or both. These include the N3-X (Ref. 20), the NASA CHEETA vehicle (Refs. 14 and 21) (see Fig. 3), the Single-Aisle Turboelectric Aircraft with Aft Boundary-Layer Propulsion (STARC-ABL) (Refs. 15 and 22), the D8 “Double Bubble” (Refs. 11 and 13), the ECO-150 (Ref. 23), the NASA Subsonic Single Aft eEngine (SUSAN) Electrofan (Refs. 24 to 26), and several others (Ref. 27).

One challenging question to answer in the realm of BLI and DP aircraft design is how the aerodynamic surfaces and distributed propulsors should be codesigned in to maximize the benefits of integration of these subsystems. The coupling of aerodynamic configurations with propulsors fundamentally connects the boundary conditions of both systems, producing a highly complex design space. Furthermore, propulsors could theoretically be placed almost anywhere on the aircraft, but some locations are particularly advantageous. For example, embedding propulsors into the trailing edge of the fuselage could take advantage of the larger boundary layer there compared to the rest of the aircraft. The STARC-ABL is one

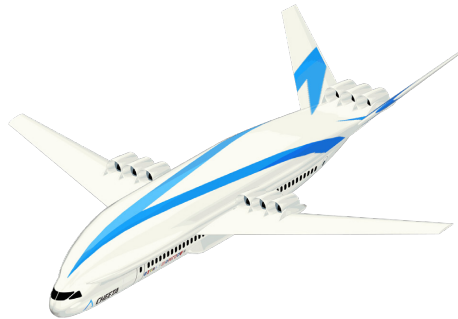


Fig. 3.—CHEETA aircraft design.

aircraft concept that leverages the high kinetic energy defect at the trailing edge of the fuselage relative to other aerodynamic surfaces. Recent research has focused on aero-propulsive shape optimization of the STARC-ABL to maximize the BLI benefit (Refs. 28 and 29). The results clearly indicate that the aerodynamics and propulsion systems must be optimized together to achieve the best results.

Several aero-propulsive shape optimization studies have kept simple either part or all of the geometric parametrization of the propulsion system. A primary reason for this simplicity is that gradient-based optimization methods are often used. These gradient-based methods can quickly fail if the optimization process selects an irregular propulsor geometry—a common occurrence when working with a parametrization with a high degree of control over the detailed geometry. To overcome this limitation, a gradient-free optimization method is used in the present study. The genetic algorithm (GA) can easily deal with ill-posed geometries by simply discarding them through the computational version of natural selection. The stochastic nature of the GA lends itself toward a higher probability of locating global optima, especially for highly complex, nonlinear systems.

One drawback of the use of a GA is the high computational cost relative to gradient-based optimization methods. For this reason, a two-dimensional (2D) aero-propulsive system is considered rather than a three-dimensional (3D) system, and a medium-fidelity computational fluid dynamics (CFD) tool is chosen over a high-fidelity CFD tool. The application of these tools to an aero-propulsive optimization problem, in conjunction with an in-house parametrization method that gives a high degree of control over the geometry while conforming to a set of logical constraints, is discussed here.

2.1.1.1 Analysis and Optimization Methods

2.1.1.1.1 Geometric Parameterization

The parametrization framework in the software package pymead (Python Multi-Element Airfoil Design) (Ref. 30), developed during the CHEETA program, was used as the geometric parametrization method for design and perturbation of the propulsor-airfoil systems. This method allows for direct, intuitive control over the airfoil shape while automatically satisfying point, slope, and curvature (G_0 , G_1 , and G_2) continuity across all surfaces. The software contains built-in methods for implementing complex geometric constraints, such as intra- and inter-airfoil distance and symmetry constraints, which are satisfied before and after parameter set updates during optimization.

The parametrization method builds the airfoils as a series of Bézier or B-spline curves joined together. Point (G_0) continuity is achieved by enforcing a coincident constraint on the first and last control points at the joint. Slope (G_1) continuity is achieved by enforcing a collinearity constraint across the end control point's adjacent control points. Curvature (G_2) continuity is achieved by selecting a value for radius of curvature at the curve joint and calculating the required distance between the primarily adjacent

and secondarily adjacent control points using Eq. (1). A visual reference for the geometric variables in Eq. (1) can be found in Fig. 4.

$$L_c = \frac{L_t^2}{R \left(1 - \frac{1}{n}\right) \sin \psi} \quad (1)$$

For the present study, an underwing airfoil system configuration was studied. The integrated aero-propulsive configuration was broken up into three airfoils: the main airfoil, the nacelle airfoil, and the hub airfoil. A sample underwing propulsor-airfoil system is shown in Fig. 5. The design shown in Fig. 5 exemplifies a 2D representation of a possible cross-section of the wing in Fig. 3 at the inboard-most propulsor.

Fig. 5 also shows the geometric constraints enforced throughout the optimization. The most unique constraint is the symmetry constraint, where part of the main element lower surface and hub element upper surface are mirrored about the hub chord line to define part of the nacelle lower surface and the hub lower surface, respectively. In addition to this symmetry constraint, several distance constraints were imposed. The first distance constraint was enforced at the joint between the second and third hub curves and the joint between the fourth and fifth hub curves, which defined a fixed hub diameter at the fan plane (see the pink dotted line). The second distance constraint was enforced at the joint between the second and third hub curves and the third and fourth nacelle curves, representing a fixed fan blade size on the upper half of the fan. A similar constraint was imposed at the joint between the fourth and fifth hub curves and the first and second main element curves, representing a fixed fan blade size on the lower half of the fan.

The 3D representation of this 2D airfoil system can be obtained by revolving the nacelle and hub airfoils about the chord line of the hub, as shown in Fig. 6, after adjusting the hub radius to match the desired internal flow area. A large fan hub-to-tip ratio of 0.38 was used here based on outcomes of related propulsor-machine co-optimization studies (Ref. 31).

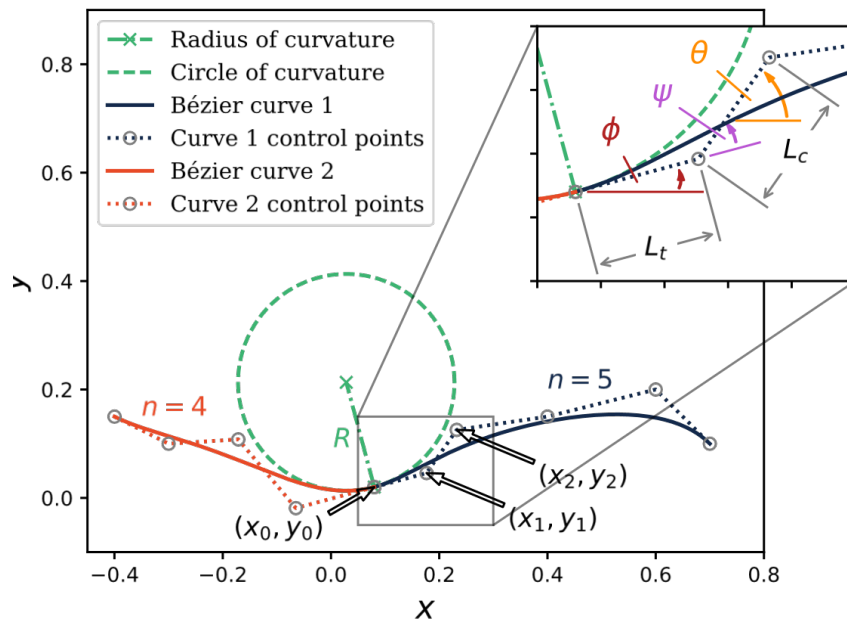


Fig. 4.—Bézier curve joint with point, slope, and curvature continuity preserved. Here $\phi = 15^\circ$, $\theta = 55^\circ$, $\psi = 40^\circ$, $L_t = 0.1$, $R = 0.2$, $L_c = 0.0972$, and $n = 5$. Both Bézier curve 1 and Bézier curve 2 have radius of curvature 0.2 at (x_0, y_0) .

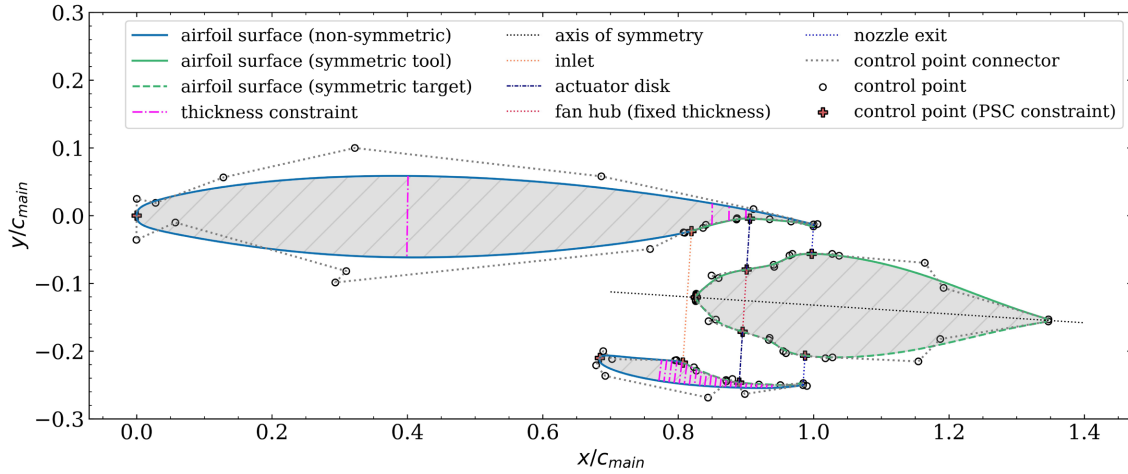


Fig. 5.—Example airfoil system configuration analyzed in present study.

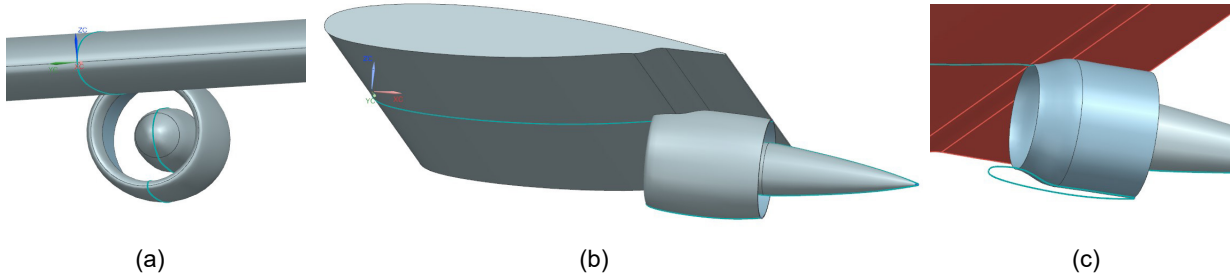


Fig. 6.—Conceptual 3D model of propulsor-airfoil system after main airfoil extrusion and nacelle and hub revolutions about hub chordline. Model generated from baseline airfoil system (shown in teal). Auxiliary views: (a) Front. (b) Side. (c) Internal (nacelle external surface hidden). Internal view shows that symmetric constraints on airfoil system as shown in Fig. 5 correspond to axisymmetric set of surfaces from just upstream of fan to nozzle exit.

2.1.1.1.2 Aerodynamic Simulation Framework

The aerodynamic simulation framework used for the present study was the MSES suite, which solves the Euler equations coupled with a boundary-layer method for viscous cases (Refs. 32 and 33). For each case, the solver options are configured to conserve streamline entropy (Euler equations) everywhere except at the leading edge of each airfoil element, where isentropic flow is assumed and a Full-Potential solver is used. The far-field boundary condition used was “vortex+source+doublet” (Ref. 33). For the viscous cases, a Reynolds number of $Re = 28.1 \times 10^6$ was used based on a chord length of 5.569 m and freestream flow with $M_\infty = 0.698$ of the CHEETA aircraft. The angle of attack was set by MSES to reach a target lift coefficient of $C_l = 0.6055$ for the design condition. The flow was artificially tripped from laminar to turbulent at 1% chord length for each airfoil, which was observed to better achieve convergence for each viscous case across the broad range of parameter perturbations considered in this study. A dissipation constant of $C_\mu = 1.11$ was used for flow regions with $M > M_{crit} = 0.95$ to improve shock stability. A representative mesh (generated by MSET, the computational-grid-generation tool in the MSES suite) is shown in Fig. 7.

The default convergence criteria were used: the solution was considered converged when all tracked residuals fell below 1×10^{-5} . The residuals considered included the root mean square and maximum density, grid node displacement, angle of attack, and circulation, along with maximum Reynolds shear stress coefficient, transition amplification factor, momentum thickness, displacement thickness, and shape factor for viscous cases. All cases presented here were fully converged.

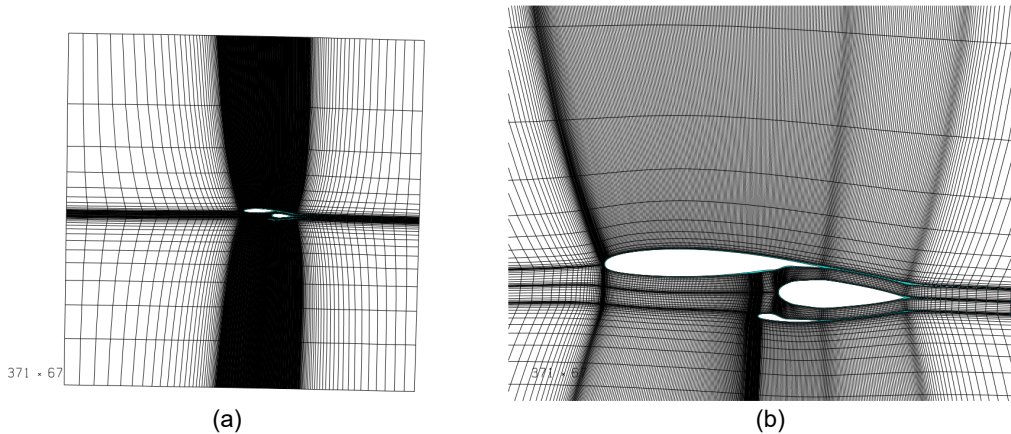


Fig. 7.—MSET mesh for baseline configuration. (a) Full view. (b) Detail view.

The MSES actuator disk (AD) model assumes a uniform jump in stagnation enthalpy and entropy at a specified streamwise station matching a specified total pressure ratio and fan efficiency. For the present study, the two ADs were specified to emanate from the hub Bézier curve interfaces denoted by the red “plus” symbols just downstream of the hub leading edge, as shown in Fig. 5. The fan total pressure ratio used for the on-design case was 1.28, and the fan efficiency used was 0.95. The AD model, while sufficient to show the salient physical trends of the propulsion system influence on the airfoil, does not capture the effects of the AD on the boundary layer with a high degree of accuracy. The AD influences the inviscid flow region upstream of the fan, but it only influences the viscous region upstream by capturing the effects imposed by the inviscid streamtube on the viscous streamtube. The AD does not necessarily influence the boundary layer velocity profile in the same way it would if the AD momentum and energy jump extended all the way to the surface rather than only to the boundary layer edge.

The main benefit to using a lower-fidelity CFD tool such as MSES as a starting point for a genetic-algorithm optimization study is the fast convergence time. Practically, many of the good propulsion-airframe-integrated airfoil system designs produced converged flow fields within 15 to 25 iterations, which took on the order of 10 to 20 s in serial on a high-powered computer with Intel® i9-10940X central processing units (CPUs) at 3.30 GHz. This fast CFD flow-field convergence is useful for applying a genetic algorithm to a complex problem, such as that considered in the current study, where thousands to tens of thousands of function calls are required.

2.1.1.1.3 Performance Characterization

In aerodynamic shape optimization, common performance metrics include the airfoil drag coefficient, C_d , maximum lift-to-drag ratio, $(L/D)_{\max}$, and range efficiency as assessed by Mach number times lift-to-drag ratio, ML/D . The best choice of performance metric for an optimization problem depends strongly on the application. For example, C_d may be used in a single airfoil optimization for a cruise configuration, whereas $(L/D)_{\max}$ might be used for optimization of a high-lift, multi-element airfoil configuration. Since the goal of the current study is to optimize for cruise, C_d was investigated first as a performance metric. Significant reductions in C_d were found to be achievable; however, the geometry local to the internal flow region varied dramatically between similar test cases, particularly at the hub nose and around the AD. In some optimized cases, sudden changes in curvature upstream of the AD location were present. This high-curvature-rate geometry does not necessarily affect the C_d to a high degree because it is in a favorable pressure gradient region, but it could pose problems in terms of flow distortion and fan efficiency due to ill-conditioned flow. For these reasons, a different performance metric was sought for the objective function.

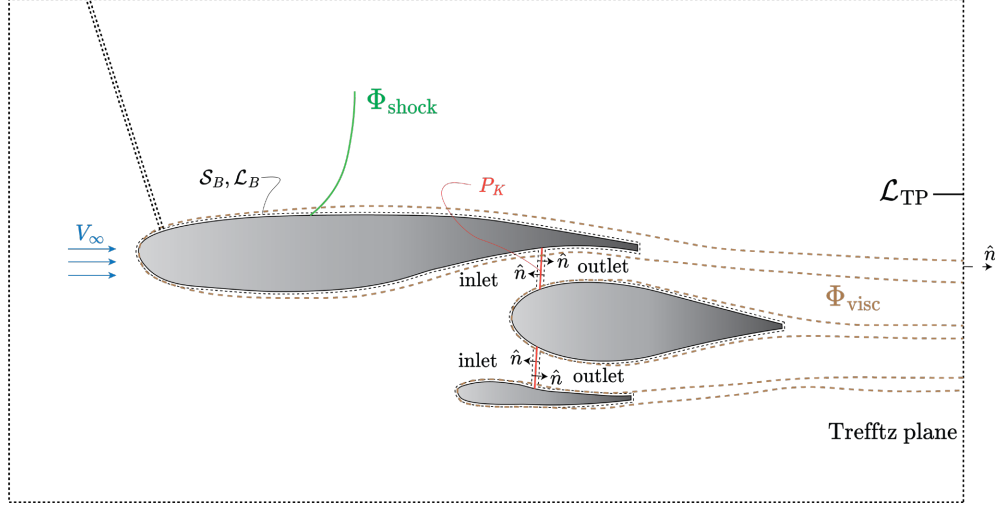


Fig. 8.—Control volume for mechanical flow power calculation of 2D underwing-propulsor geometry. Boundary layer size exaggerated for clarity; outer boundary size shown small relative to airfoil system for adequate visual resolution.

One performance metric that has been used in the past for highly coupled aero-propulsive systems is the mechanical flow power, P_K (Refs. 13 and 34). This metric captures the influence of BLI and the upstream pressure field on fan power. Combined with an upper limit on the streamwise force, it was found that the mechanical flow power yielded airfoil systems suitable for both external and internal flows. Therefore, all results presented in this work take a nondimensional form of mechanical flow power as the primary performance metric.

Two methods of measuring the mechanical flow power are possible. In the “direct” method, mechanical flow power is measured by directly measuring the pressure energy and kinetic energy flow rates imparted by the AD. In the “indirect” method, power consumption is calculated by all sources inside the control volume (see Fig. 8), including dissipation and mechanical energy flow rate out of the control volume outer boundaries. The direct method of mechanical flow power calculation is shown in Eq. (2) (Ref. 35). A control volume for the present application, drawn similarly to the control volume drawn in Figure 1 of Reference 35, is shown in Fig. 8. Note that the line integral evaluates to zero everywhere except at the propulsor inlet and outlet planes due to the no-slip condition imposed at the body surfaces.

$$P_K = \oint \left[(p - p_\infty) + \frac{1}{2} \rho (V^2 - V_\infty^2) \right] \mathbf{v} \cdot \hat{n} dS_B \quad (2)$$

In the indirect method for calculation of C_{PK} , the power consumption is calculated rather than the power input. For the present study, this indirect method was used. A power balance for transonic flow yields a similar result to that of Reference 34, with the addition of a shock dissipation term to account for the dissipation due to local shock waves:

$$P_K = \Phi_{visc} + \Phi_{shock} + \dot{\epsilon} - F_{||V_\infty} \cdot V_\infty \quad (3)$$

To nondimensionalize this equation, a procedure similar to the nondimensionalization procedure for the direct C_{PK} calculation method is followed. First, the length terms are nondimensionalized by casting Eq. (3) in terms of main-chord-relative, spanwise values:

$$\widetilde{P}_K = \frac{\Phi'_{visc}}{c_{main}} + \frac{\Phi'_{shock}}{c_{main}} + \frac{\dot{\epsilon}'}{c_{main}} - \frac{F_{||V_\infty}}{c_{main}} \cdot V_\infty \quad (4)$$

The nondimensionalization is complete after dividing through by the product of freestream dynamic pressure and freestream velocity magnitude:

$$C_{PK} = \frac{\widetilde{P_K}}{\frac{1}{2}\rho_\infty V_\infty^2} = C_{\Phi_{visc}} + C_{\Phi_{shock}} + C_{\dot{\varepsilon}} - C_{F||V_\infty} \quad (5)$$

A more detailed description of each of the terms on the right-hand side in Eq. (5) can be found in Reference 35. The viscous dissipation coefficient is calculated using the kinetic energy defect:

$$C_{\Phi_{visc}} = \frac{2}{\rho_\infty V_\infty^3} \frac{\Phi'_{visc}}{c_{main}} = \sum_{i=1}^{n_{as}} \left[\left(\frac{\rho_e}{\rho_\infty} \right)_i \left(\frac{V_e}{V_\infty} \right)_i^3 \theta_i^* \right]_{x_{max}} = 2 \sum_{i=1}^{n_{as}} K_i|_{x_{max}} \quad (6)$$

For the boundary layer, the calculation of C_{PK} is performed using the power balance principles described in Section 4.4 of Reference 34. In the present application, the nondimensional form of kinetic energy defect, K , is used rather than the dimensional form, κ , described in Reference 34. The nondimensional form is given in Eq. (7) and is a direct output of the MSES boundary layer data.

$$K = \frac{1}{2} \frac{\rho_e}{\rho_\infty} \left(\frac{V_e}{V_\infty} \right)^3 \theta^* \quad (7)$$

The kinetic energy thickness θ^* is defined in Eq. (8).

$$\theta^* = \int_0^\delta \left(\frac{\rho u}{\rho_e u_e} - \frac{\rho u^3}{\rho_e u_e^3} \right) dy \quad (8)$$

In other words, the viscous dissipation coefficient is only a function of the nondimensional kinetic energy defect of each boundary layer evaluated at the Trefftz plane. The shock dissipation coefficient is considerably more difficult to calculate accurately. An approximation adapted from Reference 35 is given in Eq. (9):

$$\begin{aligned} C_{\Phi_{shock}} &= \frac{2}{\rho_\infty V_\infty^3} \frac{\Phi'_{shock}}{c_{main}} \\ &\approx \frac{2}{\rho_\infty V_\infty^3} \int \Delta p_t \mathbf{V} \cdot \hat{n}_{shock} d \left(\frac{\mathcal{L}_{shock}}{c_{main}} \right) = \int \Delta C_{p_t} \left(\frac{u}{V_\infty} \hat{i} + \frac{v}{V_\infty} \hat{j} \right) \cdot \hat{n} d \left(\frac{\mathcal{L}_{shock}}{c_{main}} \right) \end{aligned} \quad (9)$$

To calculate $C_{\Phi_{shock}}$ for all shock waves inside the control volume, a robust shock identification routine is required. An algorithm was developed that identifies shock fronts based on a total pressure coefficient drop threshold. The first positive change in static pressure coefficient identified in the groupings of cells located by this threshold is used as the shock front delimiter to mitigate error introduced by artificial dissipation. The actual total pressure coefficient change used in Eq. (9) was calculated using the Mach number normal to the shock in the normal shock relation as described by Eq. (10):

$$\begin{aligned} \Delta C_{p_t} &= C_{p_{t,y}} - C_{p_{t,x}} = C_{p_{t,x}} \left(\frac{C_{p_{t,y}}}{C_{p_{t,x}}} - 1 \right) \\ &= C_{p_{t,x}} \left[\left(\frac{\frac{\gamma+1}{2} M_{n,x}^2}{1 + \frac{\gamma-1}{2} M_{n,x}^2} \right)^{\frac{\gamma}{\gamma-1}} \left(\frac{2\gamma M_{n,x}^2}{\gamma+1} - \frac{\gamma-1}{\gamma+1} \right)^{\frac{1}{1-\gamma}} - 1 \right] \end{aligned} \quad (10)$$

The mechanical energy outflow through the control volume is assumed to only consist of mechanical energy flowing through the Trefftz plane. Mechanical energy flow rate through the inlet is zero because $p = p_\infty$ and $V = V_\infty$ everywhere on the inlet plane. As noted in Reference 35, the outflow through the side cylinder is only important in supersonic flows. This outflow was measured for a variety of transonic aero-propulsive cases, and it was found that this total outflow was several orders of magnitude smaller than the outflow through the Trefftz plane; therefore, it was neglected. The inviscid mechanical energy flow rate through the control volume is then evaluated by integrating a nondimensional form of Eq. (2) with the integrand negated along the entire Trefftz plane depicted in Fig. 8:

$$(C_{\dot{E}})_{inviscid} = \oint \left\{ \left[\frac{\rho}{\rho_\infty} \left(\left[\frac{V}{V_\infty} \right]^2 - 1 \right) + C_p \right] \left[\left(\frac{u}{V_\infty} \hat{i} + \frac{v}{V_\infty} \hat{j} \right) \cdot \hat{n} \right] \right\} d \left(\frac{\mathcal{L}_{TP}}{c_{main}} \right) \quad (11)$$

The viscous mechanical energy flow rate out of Trefftz plane is calculated for this 2D case based on Eqs. (70) and (75) in Reference 35:

$$(C_{\dot{E}})_{visc} = \sum_{i=1}^{n_{as}} \left[\left(\frac{\rho}{\rho_\infty} \right)_i \left(\frac{V_e}{u_\infty} \right)_i^3 (2\theta_i - \theta_i^*) \right]_{x_{max}} \quad (12)$$

The total mechanical energy flow rate out of the control volume is then given by

$$C_{\dot{E}} = (C_{\dot{E}})_{inviscid} + (C_{\dot{E}})_{visc} \quad (13)$$

2.1.1.1.4 Optimization

For the current study, a solution to the following multipoint aero-propulsive optimization problem is approximated for a three-airfoil system composed of a main element, a hub element, and a nacelle element:

$$\begin{aligned} & \text{minimize } J_F = \sum_{i=1}^{n_{mp}} \mathcal{W}_i (C_{F||V_\infty,i} - C_{d_{h,i}}), J_P = \sum_{i=1}^{n_{mp}} \mathcal{W}_i (C_{P_K,i}) \\ & \text{subject to } C_{F||V_\infty} \leq C_{F||V_\infty,req} \\ & \left(\frac{t}{c} \right)_{i,nac} \Big|_{(x/c)_{i,nac}} \geq \left(\frac{t}{c} \right)_{req,i,nac}, \quad i = 1, \dots, n_{tcn} \\ & \left(\frac{t}{c} \right)_{i,main} \Big|_{(x/c)_{i,main}} \geq \left(\frac{t}{c} \right)_{req,i,main}, \quad i = 1, \dots, n_{tcm} \\ & \max \left[\left(\frac{t}{c} \right)_{1,main}, \dots, \left(\frac{t}{c} \right)_{n_{tep},main} \right] \geq \left(\frac{t}{c} \right)_{req,main} \\ & \min \left[\left(\frac{|R|}{c_{main}} \right)_{1,main}, \dots, \left(\frac{|R|}{c_{main}} \right)_{n_{a_{main},main}} \right] \geq \left(\frac{R}{c_{main}} \right)_{req,main} \\ & \min \left[\left(\frac{|R|}{c_{main}} \right)_{1,hub}, \dots, \left(\frac{|R|}{c_{main}} \right)_{n_{a_{hub},hub}} \right] \geq \left(\frac{R}{c_{main}} \right)_{req,hub} \\ & \min \left[\left(\frac{|R|}{c_{main}} \right)_{1,nac}, \dots, \left(\frac{|R|}{c_{main}} \right)_{n_{a_{nac},nac}} \right] \geq \left(\frac{R}{c_{main}} \right)_{req,nac} \\ & lb_i \leq x_i \leq ub_i, i = 1, \dots, n_x \end{aligned} \quad (14)$$

The objective functions, J_F and J_P , are a weighted sum of the streamwise force coefficient less the thrust contribution and mechanical flow power coefficient, respectively. Although minimizing for only the mechanical flow power coefficient also tends to minimize the streamwise force coefficient automatically, it was found in practice that explicitly including the streamwise force coefficient as an objective led to faster convergence of C_{PK} . Requirements at the aircraft system level directed the development of the first constraint, which effectively places a lower bound on the amount of force in the direction opposite drag. As will be seen in the optimization results, this constraint amounts to an equality constraint in the power-optimized result because more positive values of the net streamwise force coefficient yields smaller values of required mechanical flow power. It should be noted that in MSES, the “drag” due to the AD and any wave drag inside the capture streamtube are inherently coupled, so wave drag inside the capture streamtube is included in the $C_{d,h}$ term and therefore excluded from the J_F .

It was found in some cases that the optimizer tended to make the main and nacelle trailing edge regions unmanufacturably thin, so the second and third constraints were added to alleviate this issue. In this constraint, the thickness-to-chord ratio at a set of specified x/c -locations on the nacelle must exceed a set minimum value at each of those locations. The fourth constraint places a minimum value on the maximum thickness-to-chord ratio of the main airfoil element to ensure that the airfoil section can accommodate normal wing root bending moments. The fifth, sixth, and seventh constraints enforce a minimum value of radius of curvature everywhere on each airfoil, which was helpful in reducing areas of extreme curvature near the AD in some optimization cases. The eighth and final constraint simply states that each design parameter is bounded on both ends. Values for the multipoint stencil and the constraints used in the present optimization studies are shown in Table 1 and Table 2, respectively.

TABLE 1.—MULTIPOINT STENCIL USED FOR THE WEIGHTED-AVERAGE
STREAMWISE FORCE CALCULATION, WITH STENCIL
POINTS LISTED IN ORDER OF CFD EVALUATION

Point (i)	$\mathcal{W}_i$	$C_{L,i}$	$P_{r,i}$	Description
1	0.15	0.5555	1.28	Low lift coefficient
2	0.40	0.6055	1.28	Design condition
3	0.15	0.6555	1.28	High lift coefficient
4	0.15	0.6055	1.26	Low fan pressure ratio
5	0.15	0.6055	1.30	High fan pressure ratio

TABLE 2.—SCALARS USED IN THE OPTIMIZATION
PROBLEM CONSTRAINT DEFINITIONS

Parameter	Value	Description
$C_F V_{\infty,req}$	−0.065	Maximum allowed net streamwise force coefficient
$\left(\frac{t}{c}\right)_{req,main}$	0.12	Minimum main airfoil element thickness-to-chord ratio
$\left(\frac{R}{c_{main}}\right)_{req,main}$	0.004	Minimum required main airfoil radius of curvature
$\left(\frac{R}{c_{main}}\right)_{req,nac}$	0.004	Minimum required hub airfoil radius of curvature
$\left(\frac{R}{c_{main}}\right)_{req,hub}$	0.004	Minimum required nacelle airfoil radius of curvature

The nacelle and main airfoil thickness constraints were chosen by trial and error. The value of -0.065 for the required $C_{F||V_\infty}$ was chosen based on a nominal cruise thrust of 11,300 lbf distributed across nine propulsors and an estimated airfoil system cruise drag coefficient of 100 counts. While the 2D to 3D comparison is not exact, it provides a reasonable starting point for the airfoil system optimization. To solve the optimization problem, a unified version of the third iteration of the Non-dominated Sorting Genetic Algorithm (U-NSGA-III) (Ref. 36) was used. The Multi-Objective Optimization for Python (pymoo) package, which contains a variety of optimization algorithms for single-, multi-, and many-objective optimization (Ref. 37), was used to implement U-NSGA-III in the current study.

In many optimization problems, a conventional sampling method such as Latin hypercube sampling is chosen. It was found in practice for the present application that with the wide parameter bounds assigned, Latin hypercube sampling (where each parameter can vary from the minimum to the maximum) produced an insufficient ratio of converged offspring to total offspring, leading to excessive computational time required in the sampling procedure. Therefore, a custom sampling method named “constrained random sampling” was used to generate the initial population. The probability distribution used was effectively a truncated uniform distribution, with equal probability of each new parameter value being greater than or less than the baseline parameter value. This distribution is described in Fig. 9, with spaghetti plots of the underwing propulsion-airframe integration system using 50 constrained random samples shown in Fig. 10. Note that the areas of the rectangles (the probability) on either side of the dotted line are equal in both Fig. 9(a) and Fig. 9(b). For the current study, a sampling width of $w_s = 0.08$ (as shown in Fig. 10(b)) was chosen to balance the introduction of diversity into the population with computational efficiency of the sampling step.

Simulated binary crossover (Ref. 37) and polynomial mutation (Ref. 37), as described in Reference 38, were chosen as the crossover and mutation operators for the genetic algorithm. These operators both use probability distribution functions bounded by the lower and upper bounds of each design variable and peak at the value for each design variable corresponding to the parent(s) for crossover or mutation. In each generation, a specified number of offspring, N_{off} , were generated; these offspring were evaluated using the aerodynamic simulation framework described in Section 2.1.1.1.2 until at least N_{pop} individuals were converged. To account for a relatively high percentage of unconverged individuals, especially in the sampling stage, N_{off} was chosen by trial and error, usually with $N_{off} \geq 10N_{pop}$. Two optimization cases were run, one with $N_{pop} = 50$ and another with $N_{pop} = 150$. The parents for each population were chosen based on distance to the nearest reference line and objective function value according to Reference 36.

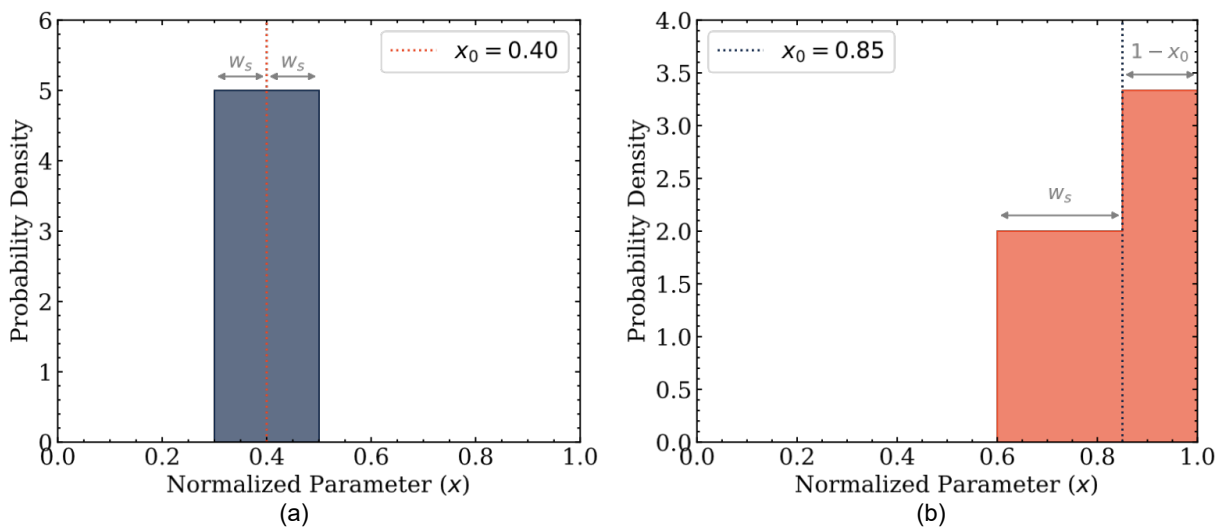


Fig. 9.—Probability density function for (a) baseline parameter value of $x_0 = 0.4$ and sampling width of $w_s = 0.10$ and (b) baseline parameter value of $x_0 = 0.85$ and sampling width of $w_s = 0.25$.

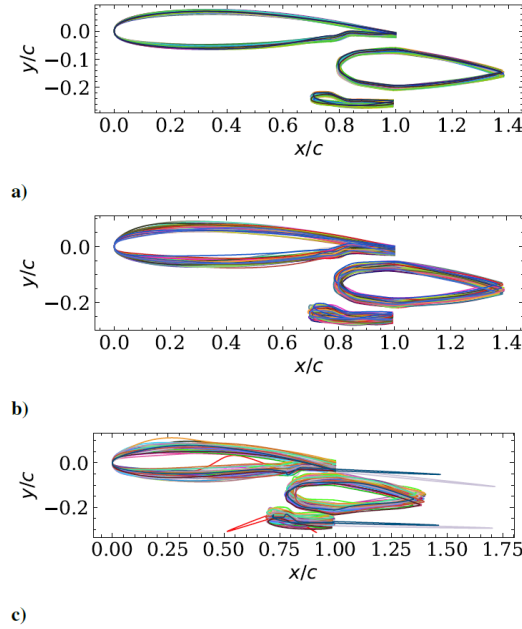


Fig. 10.—Spaghetti plots for underwing propulsion airframe integration system using (a) $w_s = 0.04$, (b) $w_s = 0.08$, and (c) $w_s = 0.12$.

To reduce required computational time, airfoil systems violating any of the geometric constraints in Eq. (14) or containing self-intersections were effectively discarded by skipping their objective function evaluations and assigning to them $J_F = J_P = 1000.0$, a large value that guaranteed U-NSGA-III would not allow the airfoil systems to be included as members of the population after survival.

Because of the randomness fundamental to the crossover and mutation operators, even the valid airfoil geometries were not guaranteed to yield well-behaved flow fields or even converged CFD simulations. To address this problem, a timeout of 35.0 s per stencil point (as determined by trial and error) was used for the subprocesses that deployed MSES to prevent hanging CFD processes. Additionally, individual airfoils that did not produce a converged solution after 100 iterations were considered infeasible and were assigned $J_F = J_P = 1000.0$, as in the case of airfoil systems with invalid geometry according to the constraints. Populations were evaluated in parallel using 20 of the 28 total available logical processors on 14 Intel® i9-10940X CPUs at 3.30 GHz. A flow chart describing the parallel implementation of U-NSGA-III specific to the current study is shown in Fig. 11.

2.1.1.2 Aero-Propulsive Optimization Results

Two aero-propulsive optimization studies were performed: one with a population size of 50, and another with a population size of 150. Because the optimization problem was formulated with two objectives, a Pareto front of solutions is generated in each case. Several individual solutions from the studies are analyzed in this section, and Table 3 describes shorthand tags used for each of these solutions. The abbreviations L, M, and R are used to mean left, middle, and right, where left is the solution with the smallest value of J_P , right is the solution with the largest value of J_P , and middle is an intermediate solution chosen based off a 50-50 weighting using an Achievement Scalarizing Function (Ref. 39). This means that in some generations, especially those toward the start of the optimization, the middle solution can coincide with either the left or right solution. In the rare, degenerate case where only one nondominated solution is found in a given generation, the left, middle, and right solutions are the same.

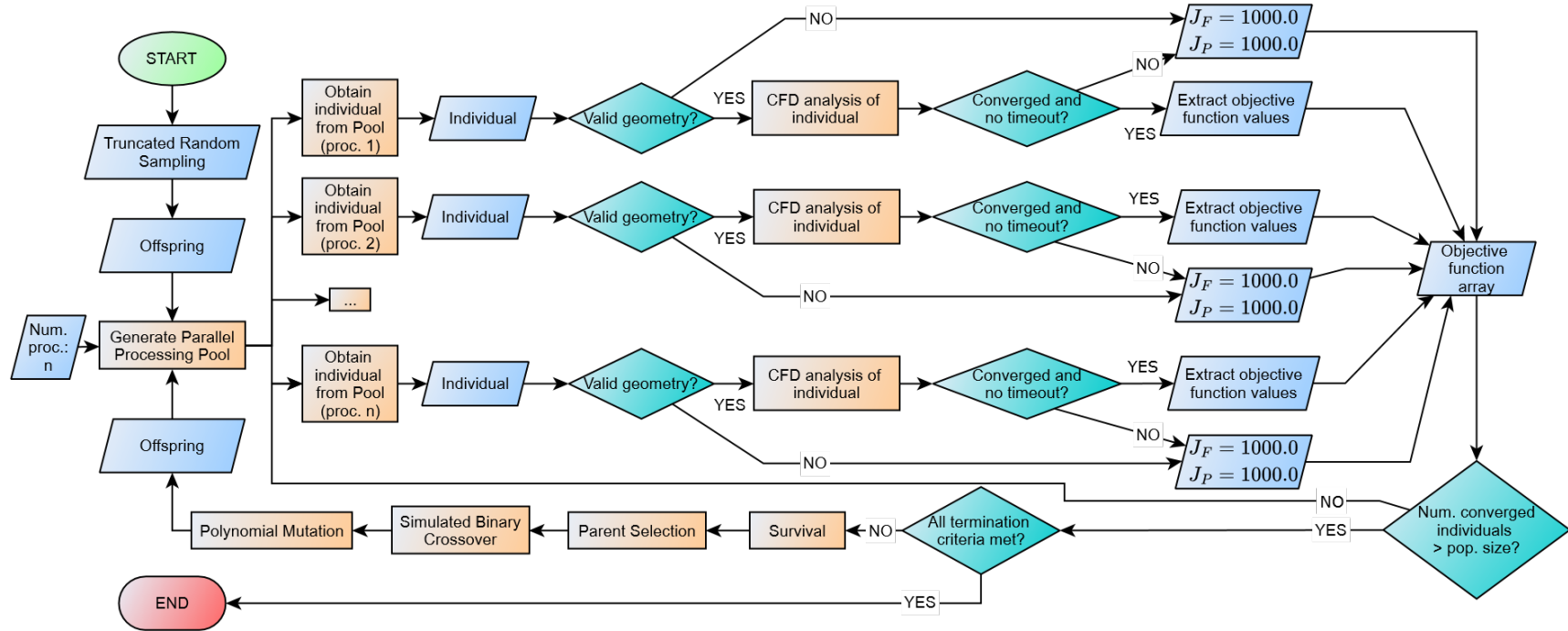


Fig. 11.—Flow chart describing U-NSGA-III genetic algorithm with detail on custom fitness evaluator.

The Pareto fronts for both optimization studies are shown in Fig. 12. Both Pareto fronts are nearly completely linear by the final generation, which indicates that no change in J_P yields an appreciable change in J_F . In fact, each Pareto-optimal solution corresponds to an airfoil system where the $C_{F\parallel V_\infty} - C_{d,h}$ term is minimized, but the streamwise force coefficient itself is varied using the AD thrust to produce an array of J_P values. The point corresponding to the leftmost edge of the final Pareto front (designated as POP50L for $N_{pop} = 50$ and POP150L for $N_{pop} = 150$) is the point limited by the net streamwise force coefficient constraint; therefore, these two solutions represent the intended final design. Any point further to the left (a lower value of J_P) would violate this constraint. All points on the Pareto front to the right of this point are effectively optimized airfoil systems for varying thrust requirements. Of course, the particular airfoil system was designed with a single system-level thrust requirement, which is practically implemented using the first constraint in Eq. (14). This constraint would ideally be an equality constraint; however, equality constraints are difficult to achieve in evolutionary optimization. For the J_P -optimal solution, though, this constraint is effectively an equality constraint. In other words, the input mechanical flow power is minimized when the required net streamwise force coefficient is the lowest allowed value. As will be shown in the results later in this section, the exact value enforced using the inequality constraint in Eq. (14) and Table 2 is achieved by both POP50L and POP150L.

It is evident from Fig. 12(a) that the $N_{pop} = 50$ Pareto fronts collapse to a single line by around generation 220, and from Fig. 12(b) that the $N_{pop} = 150$ Pareto fronts collapse to a single line by around generation 100. This also means that convergence is roughly achieved by these generation counts. Because the case in Fig. 12(b) has triple the population size of the case in Fig. 12(a), it is logical that the case in Fig. 12(b) converges in significantly fewer generations.

TABLE 3.—CASE STUDY SOLUTION TAGS

Tag	Description
POP50L	$N_{pop} = 50$, far left edge of the Pareto front
POP50M	$N_{pop} = 50$, solution from the middle of the Pareto front
POP50R	$N_{pop} = 50$, far right edge of the Pareto front
POP150L	$N_{pop} = 150$, far left edge of the Pareto front
POP150M	$N_{pop} = 150$, solution from the middle of the Pareto front
POP150R	$N_{pop} = 150$, far right edge of the Pareto front

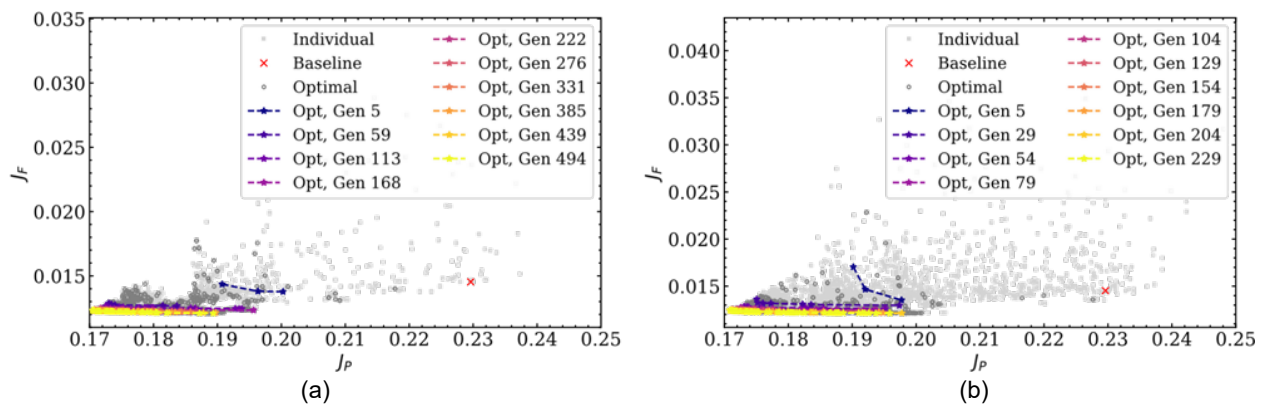


Fig. 12.—Pareto front evolution for optimization studies with (a) $N_{pop} = 50$ and (b) $N_{pop} = 150$.

Fig. 12(b) also shows a set of Pareto fronts that is more filled out from left to right. The U-NSGA-III algorithm being forced to keep this wide range of diversity in the population is one of the probable reasons that a bi-objective optimization problem formulation was found to perform better than a single-objective problem formulation for this application. Further tuning of the crossover and mutation parameters, sampling parameters, and design variable bounds could potentially improve the performance of both formulations.

The evolution of a select set of aerodynamic and propulsion characteristics is displayed for both optimization studies in Fig. 13 and Fig. 14. As the Pareto front follows a similar path in both optimization cases, it is natural that there should be many similarities between the two figures. At a glance, one of the significant differences is that the terms directly affected by increased propulsor thrust, namely $C_{d,h}$ (Fig. 13(f) and Fig. 14(f)), C_{PK} (Fig. 13(i) and Fig. 14(i)), and $C_{\dot{e}}$ (Fig. 13(l) and Fig. 14(l)), take a wider range of values between the L and the R solutions by the final generation. This maps to the increased length of the final Pareto front in Fig. 12(b) compared to Fig. 12(a). Therefore, the POP150R result has a higher value of required mechanical input power and a higher thrust value compared to POP50R.

Another interesting note is that the system angle of attack shifts significantly in the negative direction relative to the baseline in both cases. From Fig. 13(a), POP50L and POP50R converge to approximately -2.21° and -2.14° , respectively. In Fig. 14(a), POP150L and POP150R converge to approximately -2.81° and -2.42° , respectively. The design lift coefficient of 0.6055 is possible even at these negative angles of attack because of the higher pressure on the pressure surface entering the propulsor. This higher pressure is a natural consequence of optimizing for mechanical flow power, as will be seen later in this section.

Large negative values of C_m are also produced by the optimization study. From Fig. 13(c) and Fig. 14(c), POP50L, POP50R, POP150L, POP150R take values of -0.088 , -0.105 , -0.090 , and -0.105 , respectively. These large negative values of C_m may lead to an unacceptable level of trim drag for the aircraft, so it may be desirable to add a pitching moment constraint to the optimization problem formulation. This constraint was left out of the formulation to analyze the trend in the unconstrained pitching moment throughout the optimization studies.

In both optimization studies, the design value of C_{PK} takes on nearly identical values to J_P , and $C_{F||V_\infty} - C_{d,h}$ takes on nearly identical values to J_F . This indicates that the symmetric multipoint stencil chosen (see Table 1) yields a highly linear trend in terms of both drag and power. If a larger stencil were chosen, this trend might become less linear. This larger stencil would likely lead to larger values of L_t and L_c at the leading edge of the main airfoil element. It would also lead to better overall performance for the entire flight envelope. However, increasing the size of the stencil also comes at a computational cost because more MSES iterations would be required between each stencil point since the flow-field solutions would be further apart.

Ultimately, the POP50L and POP150L cases yield similar increases in optimized airfoil system performance relative to the baseline. The baseline case has $C_{PK} = 0.230$ and $C_{F||V_\infty} = 0.0145$ at the design point. POP50L achieves $C_{PK} = 0.170$ at the design point after 494 generations, which is a 26.1% reduction relative to the baseline. The value of $C_{F||V_\infty}$ reduces by approximately 22 counts to 0.0123, which is a 15.4% reduction relative to the baseline. POP150L achieves $C_{PK} = 0.170$ at the design point after 229 generations, which is a 25.9% reduction relative to the baseline. The value of $C_{F||V_\infty}$ reduces by approximately 21 counts to 0.0125, which is a 14.2% reduction relative to the baseline. It should be noted that the aerodynamic drag reduction does not include the reduction/elimination of wave drag inside the capture streamtube, since this value is inherently coupled to $C_{d,h}$.

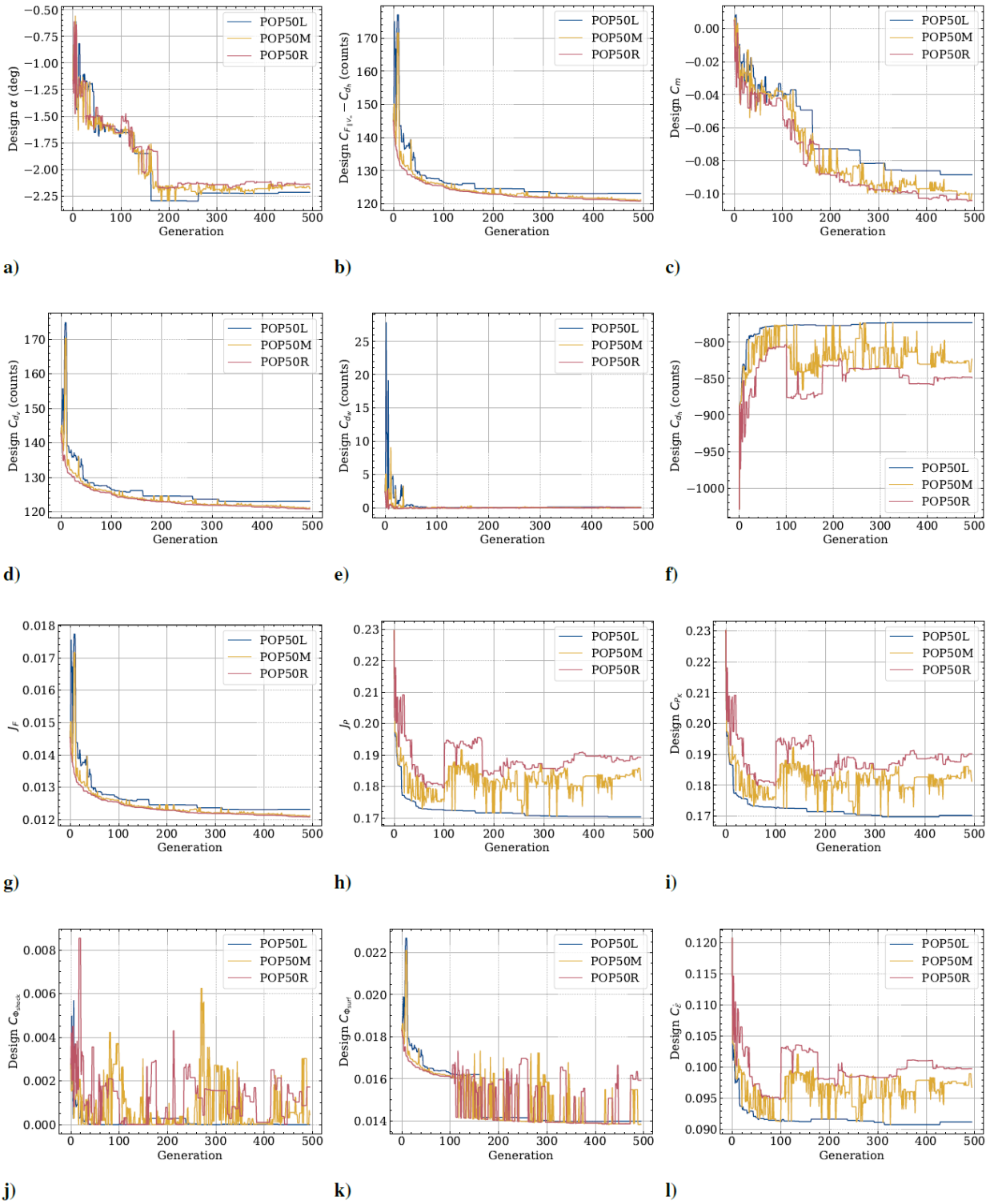


Fig. 13.—Evolution of aerodynamic and propulsion characteristics at design condition throughout optimization with $N_{pop} = 50$. (a) Angle of attack. (b) Streamwise force coefficient less AD drag coefficient. (c) Pitching moment coefficient. (d) Viscous drag coefficient. (e) Wave drag coefficient. (f) Actuator disk drag coefficient. (g) Streamwise force objective function. (h) Mechanical flow power objective function. (i) Mechanical flow power coefficient. (j) Shock dissipation coefficient. (k) Surface dissipation coefficient. (l) Mechanical outflow energy rate coefficient.

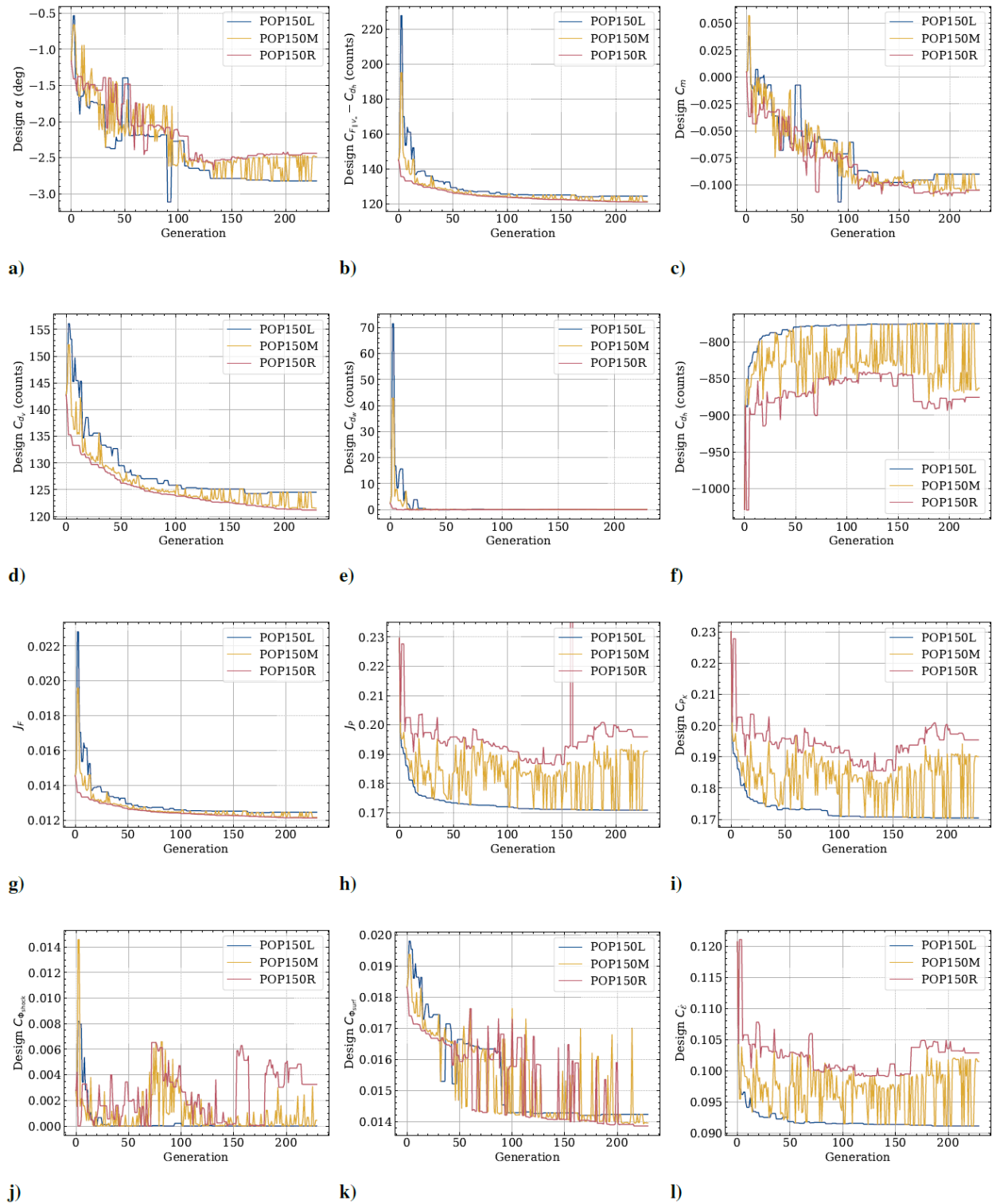


Fig. 14.—Evolution of aerodynamic and propulsion characteristics at design condition throughout optimization with $N_{pop} = 150$. (a) Angle of attack. (b) Streamwise force coefficient less AD drag coefficient. (c) Pitching moment coefficient. (d) Viscous drag coefficient. (e) Wave drag coefficient. (f) Actuator disk drag coefficient. (g) Streamwise force objective function. (h) Mechanical flow power objective function. (i) Mechanical flow power coefficient. (j) Shock dissipation coefficient. (k) Surface dissipation coefficient. (l) Mechanical outflow energy rate coefficient.

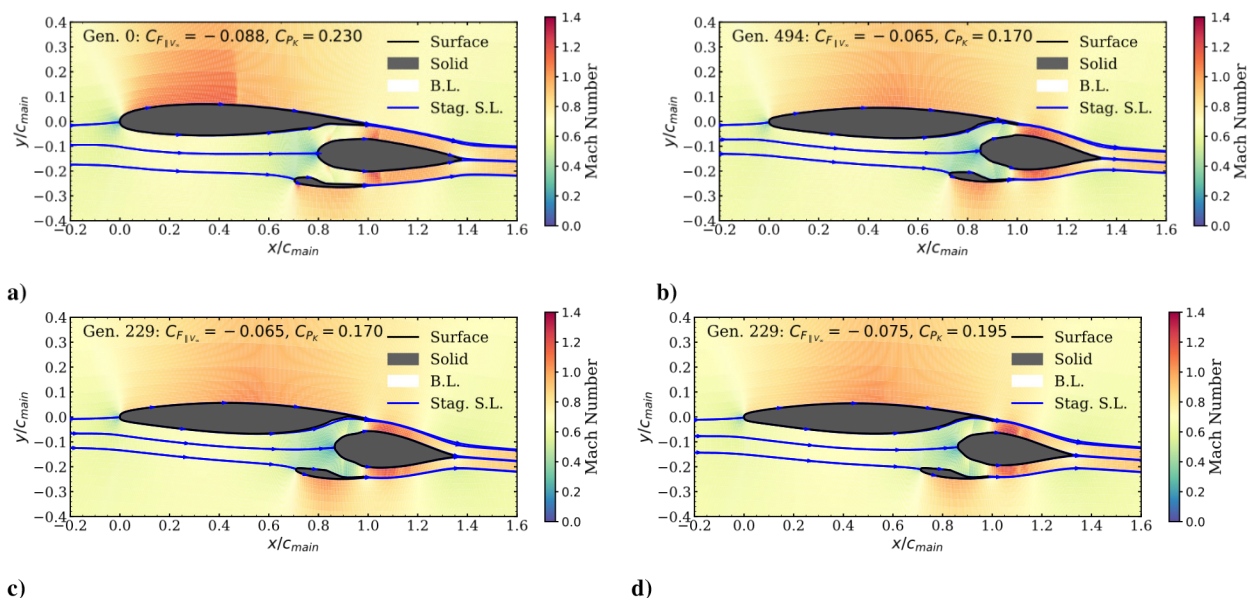


Fig. 15.—Contours of Mach number for (a) baseline airfoil system, (b) POP50L case, (c) POP150L case, and (d) POP150R case.

Mach number color maps for the baseline, POP50L, POP150L, and POP150R resultant airfoil systems are shown in Fig. 15. The baseline airfoil system in Fig. 15(a) has several undesirable flow features: there is a relatively strong shock wave on the main airfoil element upper surface, a relatively weak shock wave on the nacelle lower surface, shock waves on both hub surfaces, and nearly supersonic flow entering the AD near the hub surface. In each of the other airfoil systems shown, most or all of these undesirable flow features are either mitigated or eliminated completely.

In POP50L and POP150L (Fig. 15(b) and Fig. 15(c)), all of the aforementioned shock waves are eliminated. This corresponds to the blue curve in the shock dissipation coefficient graph in Fig. 13(j) and Fig. 14(j) converging to a zero value. The POP150R solution in Fig. 15(d) shows a small amount of shock dissipation in the hub pressure recovery region, which is likely a result of the increased AD mass flow due to the increased nozzle exit area and shortened hub recovery region. The increase in AD mass flow rate is shown by the increased capture streamtube area (height between the main airfoil and nacelle airfoil stagnation streamlines shown in blue). However, this solution does not represent the design intent and is merely shown to illustrate the opposite end (far right edge) of the Pareto front in Fig. 12(b).

The other key takeaway from the Mach color maps is that the fan face Mach number is much lower in the optimized cases than in the baseline airfoil system. While the fan face Mach number could be implemented as a constraint, this is generally unnecessary with the current optimization problem formulation because optimization of the mechanical flow power naturally leads to additional diffusion of the flow into the AD. Pre-fan diffusion increases the propulsive efficiency because an equivalent ΔV (or momentum increase) across the AD requires a smaller increase in kinetic energy. Therefore, the optimizer tends to increase the static pressure upstream of the AD face to minimize the required mechanical flow power to achieve a given net streamwise force. A lower pressure upstream of the AD is indeed achieved in all the optimized airfoil systems. In each of the cases in Fig. 15(b) to (d), the fan face Mach number falls approximately in the range from 0.4 and 0.6, much lower than the values seen in the baseline configuration. This lower Mach number corresponds directly to a higher static pressure.

Surface pressure coefficient distributions for the baseline and optimized airfoil systems are compared in Fig. 16. It is clear that the optimizer has significantly increased the static pressure ahead of the AD for

all the solutions shown. The location of the AD in these images can be inferred from the sudden increase in pressure coefficient (sharp vertical drop on the graph). The pressure coefficient immediately ahead of this drop in the graph is as much as 0.6 higher than the corresponding C_p in the baseline airfoil system. The difference is less in Fig. 16(c) because this case has a higher AD mass flow rate, and thus the increased induced velocity into the AD prevents the pressure from reaching as high a value at that location.

The surface pressure coefficient distributions provide further evidence of the shock removal. On the main airfoil element in each optimized case, the static pressure rise associated with the baseline airfoil shock at around $x/c_{\text{main}} = 0.45$ is eliminated by the optimizer. The leading-edge shock on the nacelle is also removed. The shape of the hub airfoil C_p distribution at the recovery region is changed from a sharp increase in pressure and a flat line following to gradual, roughly linear change in C_p . This change helps reduce or remove the shock in all three of these optimized cases.

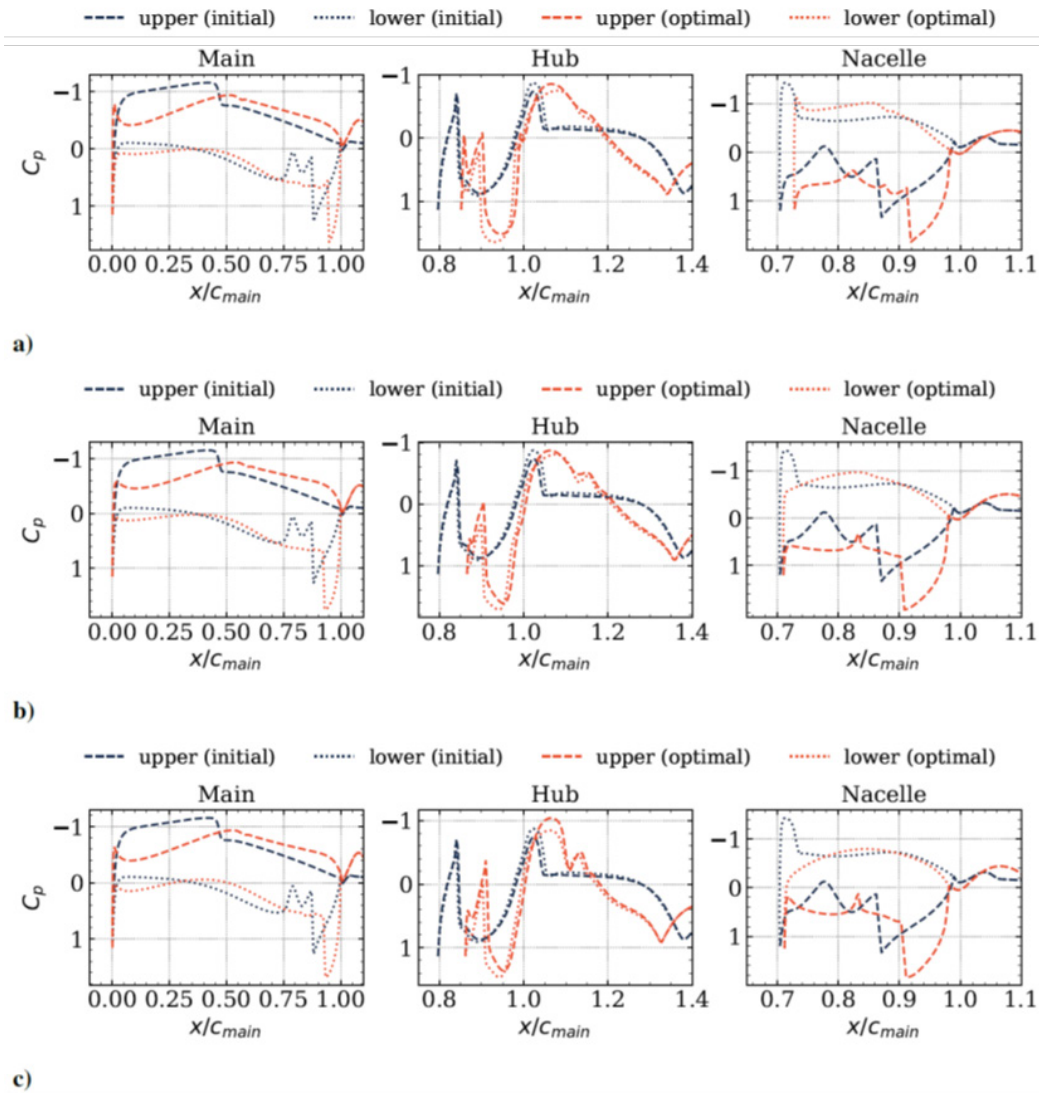


Fig. 16.—Comparison of baseline surface pressure coefficient distribution to surface pressure coefficient distributions for (a) POP50L case, (b) POP150L case, and (c) POP150R case.

The increase in minimum C_p over the main airfoil element upper surface in each of the optimized cases corresponds with a shifting in the loading to the aft portion of the main airfoil element, which partially explains the decrease in pitching moment coefficient seen in Fig. 13(c) and Fig. 14(c). It is likely that an active pitching moment constraint would keep more of this loading toward the fore portion of the main airfoil. However, doing so would also make the main airfoil element subject to normal shock wave formation at higher C_l or α relative to the design condition in the multipoint stencil.

The airfoil system geometry for the baseline and optimized configurations are shown in Fig. 17. One of the more noticeable changes across the optimized configurations relative to the baseline is that the main element's upper surface is displaced downward between about 0% and 50% chord, and the main element's lower surface is displaced downward as well between about 50% and 100% chord. This directly corresponds with the shift in loading toward the aft portion of the airfoil as mentioned in the previous discussions.

Another major change in the geometry is a shortening of the hub and an aft movement of the AD location. The hub is the shortest in Fig. 17(e), which is likely due to the fact that the nozzle exit area is larger and the hub thinner at this point. A thinner hub means that the pressure can recover an equivalent amount over a smaller chordwise distance. The aft movement of the AD location is likely because this change allows the flow to diffuse even further prior to entering the AD. The radius of curvature at the hub leading edge has also been decreased, likely to achieve the same effect.

The nacelle airfoil element has also been shortened slightly and flattened across the first half of the chordwise distance relative to the baseline airfoil system. The nacelle leading edge position and shape has also changed to be well-tailored to the capture stagnation streamline. In other words, the camber line of the airfoil and the nacelle stagnation streamline are at least approximately continuous in the optimized airfoil system configurations.

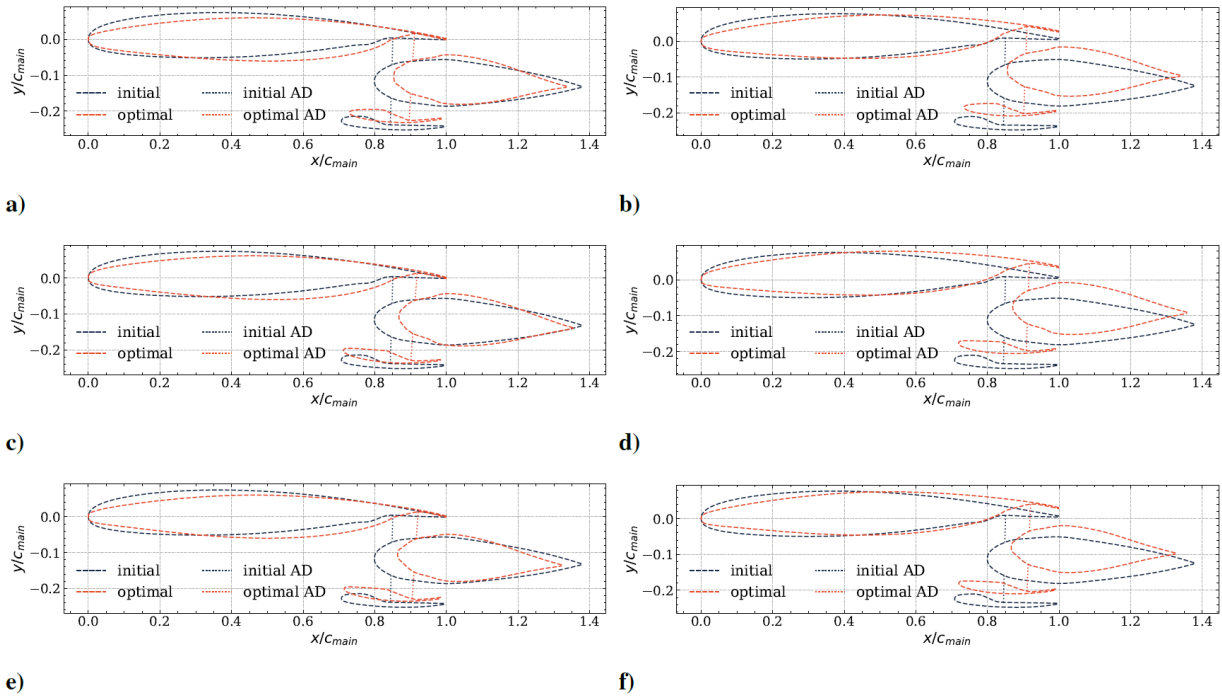


Fig. 17.—Airfoil system geometry comparison. Here, the three rows represent the three cases studied (row 1: POP50L, row 2: POP150L, row 3: POP150R), and the two columns represent two different rotations of the airfoil systems. In (a), (c), and (e), the airfoil systems are rotated about the origin such that the main airfoil is at $\alpha = 0^\circ$. In (b), (d), and (f), the airfoil systems are rotated to the angle they would be mounted on an aircraft.

2.1.1.3 Aero-Propulsive Optimization: Conclusions

The solution to a bi-objective, multipoint aero-propulsive shape optimization problem was solved for an underwing propulsor airfoil system configuration using the U-NSGA-III algorithm, the MSES viscous-inviscid-coupled simulation suite, and a custom geometry parametrization method using two population sizes. This study demonstrated that complex geometric systems with strict constraints can be optimized using aero-propulsive characterization and evolutionary optimization.

To fully realize the impact of optimization of an aero-propulsive system, a 3D shape optimization would be required. A 3D optimization would also eliminate the limiting current approximation of flow into a propulsor as 2D flow into an AD. In addition, a higher-fidelity propulsor model, such as a body-force model, could be used to increase the fidelity of the simulation and increase confidence in the optimal solution. However, such optimization studies are currently high in terms of computational cost, so they were out of scope for the present study.

Future research in the field of 2D aero-propulsive shape optimization could involve a variety of additional focuses. Several features could be added to the optimization problem formulation to make the resultant airfoil system more manufacturable and technically feasible. A fan-face Mach number constraint could be imposed, or perhaps included as an objective function to minimize so that it does not throttle the optimization. A pitching moment constraint could be implemented to prevent excessive trim drag on the system level. In addition, tighter thickness constraints on the main and nacelle trailing edge surfaces could be implemented depending on the design specification. Finally, a radial distortion threshold could be placed on the AD to balance BLI benefit with fatigue placed on a rotating fan.

To improve the convergence of the genetic algorithm for this aero-propulsive shape optimization problem, tuning or even optimization of some of the hyperparameters could be performed. These hyperparameters include the sampling width, the crossover probability, the mutation probability, and the maximum crossover and mutation step widths. Further analysis could also investigate the sensitivities at the optimum point to verify the optimality of the final solutions.

2.1.2 Powered High-Lift Systems

The integration of DEP systems with wings, or other lifting surfaces, represents a tremendous opportunity for enhancing low-speed, high-lift performance of future aircraft. The high-momentum propulsive streamtube produced by propulsive devices can be utilized, alongside trailing-edge flaps or other high-lift devices, to enhance circulatory and noncirculatory lift generating mechanisms.

Recent interest in powered lift has increased as innovative concepts for short/vertical takeoff and landing (STOL/VTOL) aircraft have been introduced. These concepts are often enabled by the use of DEP systems—hybrid-electric or fully electric propulsion systems in which multiple electric fans are spread spanwise across a fixed wing. Industry interest in DEP is well founded (Ref. 8). For the CHEETA ULI program, the H₂ fuel-cell power system has enabled the use of DEP on inboard wing sections, thus warranting further study into how DEP can be integrated into the airframe. DEP concepts leverage the advantages of electric propulsors: they can be designed with a much smaller form factor than conventional gas turbines, and their energy storage and power transmission systems can be centralized away from the actual propulsors themselves (Ref. 8). Whereas the work of Section 2.1.1 describes the high-speed aero-propulsive integration, this portion will describe the integration of DEP for a high-lift system.

When configured to augment the high-lift capabilities of an aircraft, DEP systems belong to a larger class of powered-lift systems. There are many potential benefits of powered lift augmentation (Ref. 40) that are shared by DEP concepts. These benefits have also been demonstrated with aircraft prototypes such as the Boeing YC-14. May and Bean demonstrated experimentally that the sectional lift coefficient

for this aircraft along its powered span cross section reached as high as $C_l = 8$ in ideal cases (Ref. 41). Blown lift systems create additional lift in two primary ways: (1) the deflection of a jet wake streamtube normal to the streamwise direction, or thrust vectoring; and (2) the compounding of wing leading-edge circulation induced by circulation manifest in the shear layer between the jet wake and freestream flow, or supercirculation. Thrust vectoring has been utilized in many production aircraft, primarily in STOL/VTOL applications (Ref. 42). The supercirculation effect has been modeled previously in 2D vortex panel codes, which show the potential for the use of DEP in a high-lift system on a conventional fixed-wing aircraft (Refs. 43 and 44).

This portion of the report is intended to develop an analytical scheme capable of modeling the performance of DP systems using vortex panel methods. These methods cannot provide the same level of flow feature detail as Navier-Stokes CFD solvers but do enable rapid exploration of the airframe design space. The scheme has also been validated through a 2D wind tunnel experiment. Beyond validation, the experiment will also provide insight into the key performance parameters for a DEP system, as well as addressing the challenges of integration for high lift regimes of flight. The following sections are adapted from the present authors' recent published work on this topic (Refs. 44 and 45).

2.1.2.1 Computational Scheme

2.1.2.1.1 Potential Flow Framework

The simulation effort closely follows the methodology presented by Shollenberger (Ref. 43) and is described extensively by Jois and Ansell (Ref. 44). The method is extended to consider arbitrary 2D lifting geometries through the use of a discrete, linearly varying vortex panel method as given by Katz and Plotkin (Ref. 46). This same method is used for the inviscid model component of the popular XFOIL analysis framework (Ref. 47). Unlike Shollenberger's previous method, the present method models the thickness of lifting surfaces, which means that the pressure coefficient distribution around each surface can be resolved. This capability is important for the detailed design of airfoils to be used in coupled aero-propulsive applications.

The analysis presented does not directly include the influence of a propulsion system, such as through the implementation of an AD model. Rather, the change in total pressure that occurs from the presence of a propulsive actuator is simulated by binding circulation to two wake boundary elements that emanate from the trailing edge of any two specific surface elements. Therefore, the exact location of a propulsive actuator is never specified, just that such an actuator exists somewhere between two specific surface elements and that the actuator provides some prescribed value of total pressure change across it. This prescribed value manifests in the strength of circulation bound to the trailing wake boundaries. A limitation of this technique is that it cannot predict detailed pressure distributions or streamlines inside a ducted fan nacelle, for example. However, this approach does adequately resolve the induction influences of the jet wake on all lifting surfaces.

As input, the numerical algorithm takes a set of surface element geometries and calculates the surface pressure distributions, section lift coefficient, streamline patterns, and the boundaries of a powered wake emanating from the trailing edge of any two surface elements. Unlike the method presented by Spence (Ref. 48), this method can resolve both the upper and lower boundaries of a powered jet wake. The solution depends on a prescribed value for the change in total pressure across the powered wake. The circulation strengths and positions of the wake boundaries are variable according to the circulation distribution bound to the lifting surfaces. Therefore, the algorithm iteratively calculates the circulation distribution on lifting surfaces, as well as the circulation and spatial position of the wake boundaries to meet the boundary conditions. Each iteration of the algorithm utilizes a vortex panel method to solve for the circulation distribution around each lifting surface and wake.

2.1.2.1.2 Assumptions

The assumptions follow directly from Reference 43. A 2D, perfect, inviscid, and incompressible flow domain is considered; therefore, the effects of jet flow entrainment across the wake boundary and boundary layers are ignored. The jet is also assumed to be perfectly expanded, such that the static pressure of the high-energy flow exhausted from the propulsor outlet is equal to that immediately outside of the jet wake boundary. While any number of 2D lifting elements (airfoils) can be included, the scheme presently considers that the flow between only two of these elements is powered.

2.1.2.1.3 Boundary Conditions

As in Reference 43, the analysis considers two potential flow regions: an outer region of the flow around the airfoil elements, and an inner powered region within the propulsion exit streamtube, each governed by a separate velocity potential function. The inner flow region is bounded by both an upper and lower wake element that emanates from the trailing edge of two surface elements. According to the assumptions presented, we can apply Bernoulli's equation at the jet wake boundary for both regions, given a known difference in total pressure across the streamtube boundaries. Since the jet is assumed to be perfectly expanded, the static pressure at both locations must be the same. Therefore, only the jet velocity and total pressure jump are different between these two formulations.

The use of a potential flow approach implies that there is a discontinuity in velocity across the boundaries of the wake or inner regions, which in turn implies that these boundaries can be replaced with vortex sheets with some distributed strength. Algebraic manipulation and nondimensionalization gives an expression for the distribution of vorticity strength as a function of total pressure jump between the inner and outer flow regions. The nondimensionalized pressure jump is presented as the thrust coefficient, or C_T , throughout this report. Fig. 18 is a schematic of the lifting surface and wake boundary discretization.

2.1.2.1.4 Algorithm

Implementation of the numerical algorithm follows similarly to that presented in Reference 43 with some modifications. Namely, this implementation allows the analysis of any number of surface elements, all of arbitrary geometry. The algorithm requires the following inputs to proceed:

- A set of discrete coordinates describing the geometry of each surface element (clockwise from trailing edge to trailing edge)
- The angle of attack α for the system of surface elements
- The desired value of C_T
- The far-field distance where the wake boundaries are resolved
- The number of panels to be used to discretize the wake boundaries

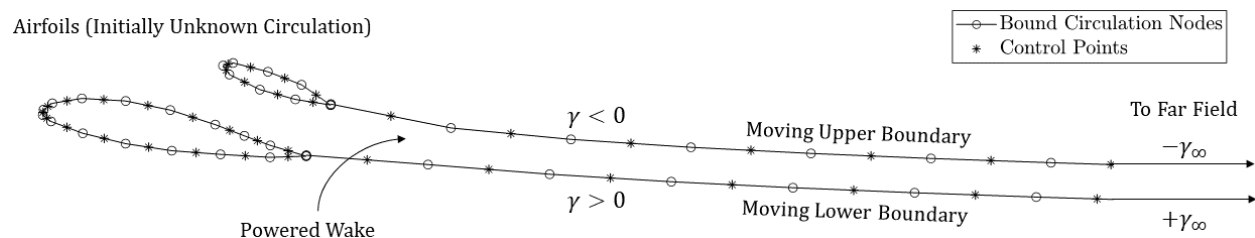


Fig. 18.—Representation of vortex panel algorithm for nominal aero-propulsive configuration. Each boundary is moved iteratively to enforce flow tangency across all wake panels.

A detailed description of the vortex panel method implementation is presented in (Ref. 44). First, the algorithm implements the linearly-varying vortex panel method described in Reference 46 to solve for the circulation distribution around the solid lifting surfaces. Each surface is discretized into coordinate endpoints, each with assigned circulation strength that is initially unknown. The boundary condition of flow tangency is enforced at a control point at the center of each panel. This condition is satisfied by setting the sum of all freestream and induced velocity contributions normal to a panel equal to zero. Similarly, the wake boundaries are discretized. However, since the jet wake boundaries are not solid surfaces, the flow tangency condition is produced by moving the wake boundaries themselves in an iterative procedure.

The velocity induced at any given spatial point, including the control point on each panel, is a sum of the velocities by the freestream velocity, the vortex distribution on the lifting surfaces, near-field wake boundaries, and semi-infinite vorticity in the far-field wake boundaries. The induced velocities at each panel control point are calculated using the Biot-Savart law. Since this formulation includes provisions for an unlimited number of lifting surfaces, the Kutta condition is enforced independently on each surface by setting the sum of the circulation at the trailing edge of the surface to zero. Fig. 19 shows a schematic of a vortex panel and velocities it induces.

Once the circulation distributions around the surface elements are calculated, the induced velocities on the wake boundaries must be solved, again using the Biot-Savart law. With new induced velocities, the flow tangency condition is no longer met at the wake boundaries. Therefore, each boundary is moved spatially to re-enforce the flow tangency condition. The algorithm iterates until the wake boundaries no longer move and the circulation distribution change on all discretized elements is within a specified residual. Typical configurations take less than 10 iterations to converge to a residual of 10^{-4} . Once the iteration process was complete, streamlines, velocity quiver plots, and surface pressure coefficient distributions were produced. The 2D sectional lift coefficient of the system was also calculated. Initial validation work for this simulation is presented in Reference 44, along with a case study of an airfoil system modified for use on the CHEETA program (Ref. 49). The rest of this section details the adaptation of this airfoil system for a 2D wind tunnel experiment, intended as additional validation for the computational scheme.

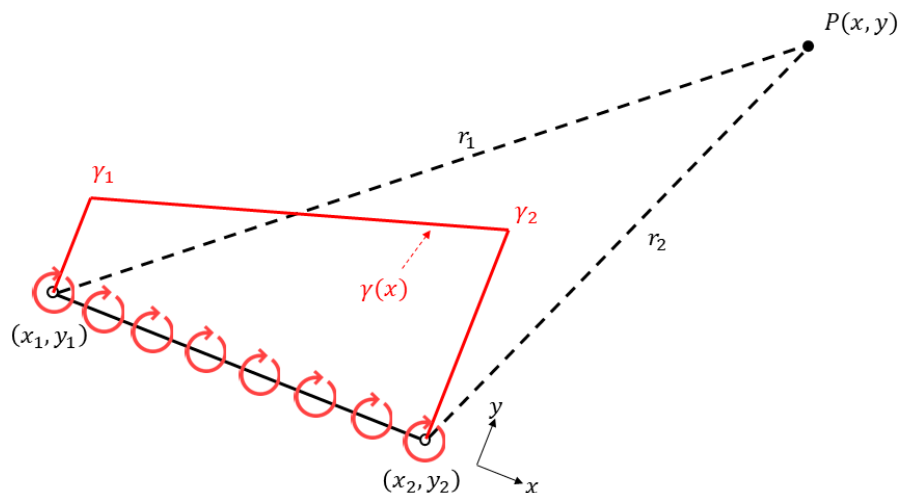


Fig. 19.—Geometric visualization of panel with bound circulation linearly distributed on surface, inducing velocity on point P .

2.1.2.2 Experimental Validation

2.1.2.2.1 Aero-Propulsive Configuration

The aero-propulsive configuration presented in this study is adapted from that designed for the CHEETA program (Ref. 49). In the cited work, the authors leverage a Bézier curve parameterization scheme to optimize an aero-propulsive geometry for a high-altitude cruising condition (Ref. 30). The discussed 2D configuration includes a main airfoil element producing most of the lift, a nacelle airfoil element that creates a ducted internal flow, and a fairing around the electric motor assembly at the heart of the propulsion system. This configuration is used as the reference upon which a high-lift system is integrated.

However, adapting such a configuration for the high-lift regime requires modifications to the leading edge of the main airfoil element and aft portion of the propulsion system. As discussed by Jois and Ansell, potential flow simulations of the unmodified geometries show large leading-edge suction peaks that likely cannot be sustained by the main element upper-surface boundary layer (Ref. 44). Using a Krueger flap was shown to alleviate leading-edge suction values for high angles of attack in the cited work's case study. The Krueger flap designed for this series of experiments combines the lower surface of the main airfoil element up to 10% chord, a circular leading edge, and G^2 continuous closeout section. The flap is placed at $x/c = 0.07$ and at $\delta_{KF} = -64^\circ$, relative to the reference airfoil chordline.

A commercially available electric ducted fan (EDF) from Schübeler Technologies was selected for the model, which had a diameter of 2.72 in. The dimensional reference chord from the CHEETA configuration was selected to match the diameter-to-chord ratio from the program, while the fan face and fan centerline were placed in accordance with the positions from the reference configuration. A total of 10 EDFs were integrated into the experimental model to provide sufficient space within the model for wire routing and other instrumentation.

The reference nacelle airfoil required modification to accommodate the new EDF duct and alleviate the possibility of inlet separation at high thrust coefficients. For low static thrust cases, the propulsion system capture streamtube area becomes infinitely large and the stagnation point moves to the trailing edge of the nacelle airfoil. While these large values of C_T are not realistic, values of $C_T = 2$ are expected from the experimental configuration (Ref. 44). To alleviate possible inlet separation, the nacelle airfoil was modified to include a larger leading-edge radius.

To maintain the two-dimensionality of the experimental configuration, a mail-slot inlet and rectangular nozzle were selected for the DEP system. However, since 10 discrete EDFs were used for propulsion, a method was devised to smoothly transition the flow from a circular fan duct to a rectangular nozzle exit. The nozzle exit area was calculated by applying principles of quasi-one-dimensional (1D) isentropic flow and compressor matching between the propulsor capture streamtube and nozzle exit. Using a baseline nozzle-to-fan area ratio of 0.8 provided a sufficient range of C_T values, achievable within the operating limits of the selected EDF. Given this ratio and the EDF diameter and data provided by Abbott et al. (Ref. 50), the required length-to-nozzle-diameter ratio for the internal transition duct could be calculated. A value of $l/d = 1.31$ was selected for the experimental configuration, and this value sets the total extension of the main airfoil and nacelle airfoil elements required for the high-lift system integration, increasing the total model chord length to $x/c = 1.33$.

The main airfoil lower surface and nacelle airfoil upper surface were both extended to reach the new total chord length and create the correct nozzle area ratio. The main airfoil upper surface and nacelle airfoil lower surface created the internal transition duct geometry. As the reference airfoils were originally parameterized by Bézier curves, all modifications to the airfoils preserved G^2 continuity and required an increase in airfoil thickness. The rectangular nozzle exit also simplified the design of the thrust vectoring

system to two hinged flaps created from the aft 10% chord sections of the extended main and nacelle airfoil elements.

Whereas the experimental configuration was intended as a quasi-2D system, 3D flow conditioners were designed to separate the propulsive streamtubes of each EDF. The downstream conditioner, located aft of the EDF trailing edge, had a semicircular sidewall tangent to two EDF streamtubes. The two sidewalls were lofted into a cusped trailing edge located just forward of the thrust vectoring flap leading edge. The upstream flow conditioner separated the capture streamtubes of each EDF beginning at 61% chord, just before the inlet region on the main airfoil upper surface. The flow was then smoothly transitioned using G^2 continuous surfaces into semicircular trailing edges at the entrance of each EDF. The leading edge of the conditioner followed a G^2 continuous path from the main airfoil to the nacelle airfoil leading edge. Fig. 20 shows an example of the flow conditioners used between two EDFs to create streamtube separation.

The final 2D airfoil configuration is shown in Fig. 21, overlaid on the reference configuration from Lauer and Ansell (Ref. 49). This configuration was analyzed as a case study by Jois and Ansell using an iterative potential flow scheme (Ref. 44), showing the configuration's potential for use as an integrated high lift system. The following sections of this report describe the adaptation of this configuration into an experimental model for wind tunnel testing and its performance characteristics.

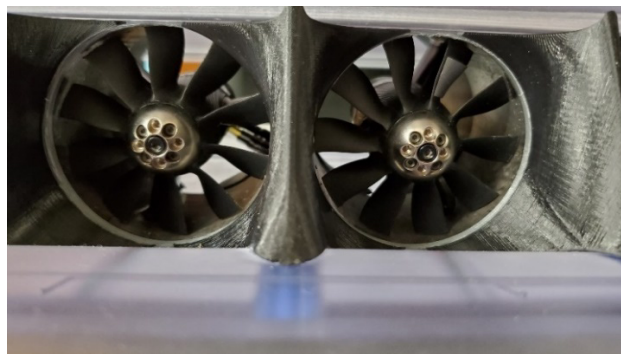


Fig. 20.—Detailed view of upstream flow conditioners between two EDFs, used to separate capture streamtubes.

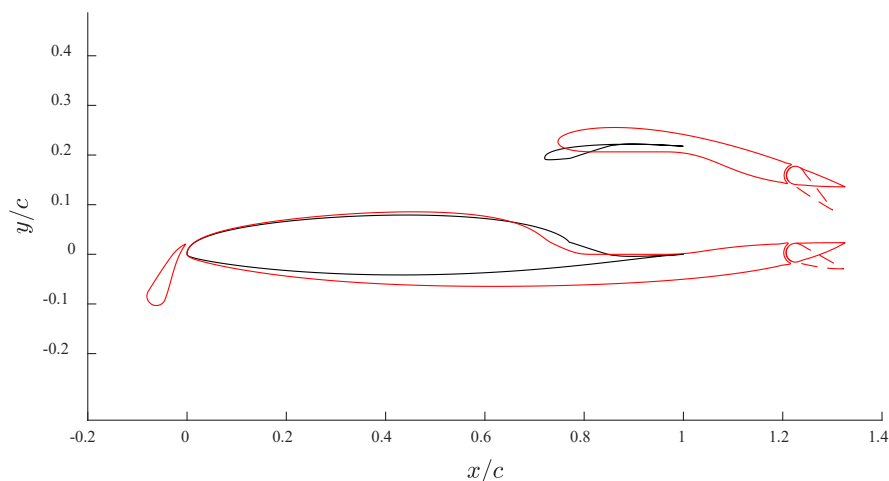


Fig. 21.—Reference geometry in black; modified experimental geometry in red; flaps shown in dashed red at $\delta_f = 30^\circ$.

2.1.2.2.2 Model Construction

The experiments were conducted in an open-return subsonic wind tunnel at the Aerodynamics Research Laboratory, The University of Illinois, Urbana-Champaign (UIUC). Based on the 2D airfoil geometry and 3D internal flow geometry parameters, a set of airfoil elements were drawn and manufactured using a stereolithography (SLA) 3D printing process. A series of 144 surface pressure taps and corresponding internal channels were included in the airfoil model design to facilitate detailed surface pressure measurements. The main element was designed with mounting locations for planar beam load cells and the EDFs. Each EDF mounts to its load cell using a custom hoop bracket 3D printed using a fixed-deposition manufacturing (FDM) technique out of polylactic acid (PLA) plastic. Similarly, the upstream flow conditioners, internal transition duct separators, and EDF motor hub fairings were 3D printed. Steel spars were used to transfer loads from the SLA elements to a three-component force. One separate rectangular spar was used as a non-load-bearing pass through for the load cell wiring. This decision was taken to minimize electromagnetic interference (EMI) between the high-current EDF wires and load cell instrumentation. Due to large anticipated loads on the Krueger flap, only a small portion of the flap was printed using the SLA technique. This portion was placed at the model centerline and attached to two steel spars modeled to the Krueger flap geometry using a wire electric discharge machining (EDM) technique. The model was mounted vertically in the wind tunnel by fixing two structural rectangular spars to the force balance system. Fig. 22 shows the model installed in the wind tunnel facility.



Fig. 22.—Airfoil model installed in wind tunnel facility.

2.1.2.2.3 Control System and Instrumentation

Each EDF was driven by a brushless direct-current (BLDC) motor paired with an electronic speed controller (ESC) and powered by a lithium polymer (LiPo). The selected ESC has built-in telemetry that includes rotational speed, in revolutions per minute (RPM), inferred from back electromagnetic force. These telemetry signals were received by microcontrollers and fed to the data acquisition (DAQ) system to facilitate active, closed-loop control of the EDF RPM. Angle-of-attack adjustment and measurement of the forces and moments acting on the model were both performed by a 3-component external wind tunnel balance from ATE Aerotech. Surface pressure data were acquired from the pressure taps using a DTC Initium Data Acquisition System, which has an accuracy of 0.05% of full scale. Each EDF had its thrust instrumented by a Flintec 3.75-kg planar beam load cell. A set of National Instruments® (National Instruments Corp.) SCXI modules supplied 10 V excitation voltage to all the load cells and amplified and filtered their output signals.

2.1.2.2.4 Testing Conditions

Measurements for this experiment were conducted at $Re = 0.75 \times 10^6$, which provided a certain range of C_T that could be produced by the EDFs. Data were acquired for both Krueger flap removed and deployed configurations, with varying angle-of-attack, EDF tip-speed ratio, and thrust vectoring flap deflection angle. Data were acquired for $-6^\circ \leq \alpha \leq 18^\circ$ for the configuration without the Krueger flap installed, and $8^\circ \leq \alpha \leq 22^\circ$ for when the flap was installed. Acquisition was stopped prior to the given range for each configuration if loads on the force balance system were about to be exceeded. The highest fan tip-speed ratio was not run in conjunction with Krueger flap installation to avoid overloading the balance system as well.

2.1.2.3 Computational Results

Previous work by Jois and Ansell (Ref. 44) describes the results anticipated using the computational framework described previously. The capability of the scheme to generate velocity vector plots and surface pressure distributions was leveraged to predict the performance of the 2D wind tunnel model, and these plots are shown in Fig. 23 and Fig. 24.

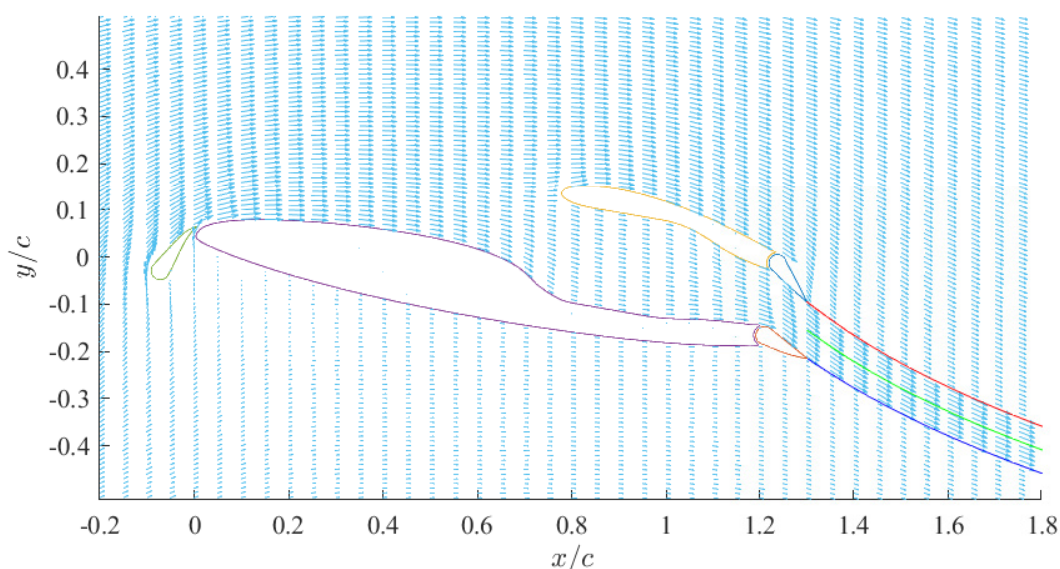


Fig. 23.—Velocity vector plot for thrust-vectoring configuration with $\alpha = 10^\circ$, $C_T = 2$, and $\delta_f = 30^\circ$.

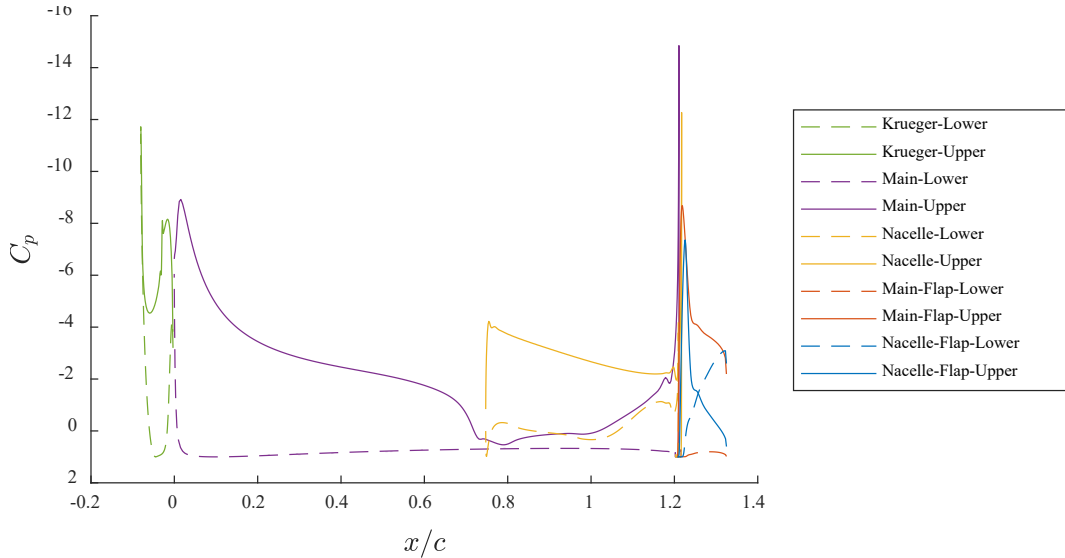


Fig. 24.—Surface pressure plot for thrust-vectoring configuration with $\alpha = 10^\circ$, $C_T = 2$, and $\delta_f = 30^\circ$.

In Fig. 23, the velocity vectors at the surface trailing edges show that the Kutta condition is manifest correctly in the solution. In addition, the discontinuity in the velocity potential function across the powered wake boundary, as prescribed for the scheme, is shown clearly by the change in the velocity vector length between the inner and outer regions of the wake. The supercirculation effect of having a powered thrust vectoring system can be visualized through the large velocity gradients in the stream-normal direction, particularly above and below the main airfoil element and Krueger flap. The surface pressure distribution for the configuration was shown in Fig. 24. The scheme's functionality is limited when attempting to model viscosity-dominated flows, which results in extremely steep inviscid pressure gradients at the leading edges of the thrust vectoring flaps and on the upper surface of the Krueger flap. However, the ability to plot pressure distributions allows an aero-propulsive system designer to better understand the types of pressure gradients expected for a given configuration and to adjust surface curvatures accordingly. The pressure distributions also depend heavily on C_T , giving designers an additional variable to adjust the performance of the system.

2.1.2.4 Experimental Results

Performance data for the experimental model is described in this section, including lift, axial force, and pitching moment coefficients for the airfoil system at varying angle-of-attack α , flap deflection δ_f , and fan tip-speed ratio λ (Ref. 45). Pressure distributions around the system for these independent variables are also shown. Finally, comparisons between the pressure distributions obtained from experiment and from the computational framework are given for some selected configurations.

Fig. 25 shows performance coefficients as a function of α , δ_f , and λ , with curves for subsequent λ values co-plotted on the same graph. A typical linear lift curve as a function of α can be observed in all cases before the stall point. Interestingly, increasing the fan tip-speed ratio tends to increase the overall lift-curve slope as well, regardless of flap deflection. This result has been observed in other related literature as well. Larger λ values also tend to cause the system to stall at a lower angle of attack. This

result could be attributed to the supercirculation effect described earlier, where the circulation manifests at the shear layers between the jet wake and freestream manifest as a much larger leading-edge suction peak, eventually becoming too great for the airfoil upper-surface boundary layer to sustain. Flap deflection shows similar effects to those described in Reference 40, where the zero-lift angle of attack decreases, and the stall point occurs earlier as well. Flap deflection also seems to decrease the lift-curve slope increment when going to higher fan speeds. Addition of the Krueger flap shows similar performance to typical leading-edge treatments, where the lift curve gets extended beyond the pretreatment stall point.

In the second row of figures, the axial force coefficient C_x is plotted on the horizontal axis as a function of the lift coefficient C_l . All configurations tend to produce wide regions of lift coefficients with relatively low axial force coefficient before the axial force increases significantly at stall. All configurations exhibit some unusual curvature in the $\lambda = 0$ case, in that the axial force decreases sharply before the stall point. This phenomenon is also seen a few other cases, namely $\lambda = 4$ at $\delta_f = 10^\circ, 20^\circ$. Future study, as described in Section 2.1.2.5, will be necessary to determine the exact cause of this performance behavior. We theorize that the effect could be caused by an interaction between the main airfoil element and nacelle near system stall or by interaction between the flap deflection, jet thrust force contribution, and aerodynamic drag on the system.

The pitching moment coefficient plots in the third row show that C_m becomes more negative, or more nose-down, with higher flap deflection angles. This echoes descriptions of trailing-edge flap performance in Reference 40. The effect of increased jet thrust is to create a larger nose-down pitching moment as well. The jet wake also creates a larger C_m increment with larger δ_f values. This behavior seems consistent with intuition: as the flap deflection angle increases, the dot product between the jet thrust vector at the nozzle exit and the stream-normal vector increases, thus increasing the moment contribution about the system quarter-chord.

Fig. 26 and Fig. 27 show the pressure distributions around the airfoil system for varying angles of attack for the clean and Krueger-flap-installed configurations, respectively. While increasing λ results in slightly lower pressure values (and thus, higher flow velocities) in the nozzle section of the system, the primary effect observed is a large increase in the leading-edge suction peak. In fact, the clean configuration can sustain peaks of approximately $C_p = -13$ before stall occurs. For all cases of flap deflection, configurations with larger tip-speed ratio tend to stall at lower angles of attack, further evidencing the effect of supercirculation on the airfoil system. The pressure distributions also tend to show peaks near the hinge line of the trailing-edge flaps, which is also consistent with the expectation that the flow will expand around the upper surface of the flaps. The addition of the Krueger flap, as shown in Fig. 27, allows the system to reach even higher leading-edge suction peak values.

Comparisons between the experimental performance of two specific configurations and the solution from the computational framework are shown in Fig. 28. The scheme tends to perform well in modeling the performance of both the main airfoil element and the Krueger flap, though the predictions on the nacelle airfoil tend to vary. While we would expect a typical panel code to overpredict C_p values, especially on the upper surface of airfoils, the underestimation of pressures on the nacelle upper surface in the clean configuration warrants further study. The lower surface of all airfoil elements is modeled well by the scheme, however, including in the nozzle region. At the trailing edge of the system, the computational scheme predicts large exit velocities on the interior of the nozzle and typical pressure recovery to freestream on the outer surfaces. This behavior is not observed in the experimental data. Further study, especially using particle-image velocimetry in the wake region, will allow for better characterization of the flow behavior near the airfoil trailing edges and ensure consistency with the modeling method in the computational framework.

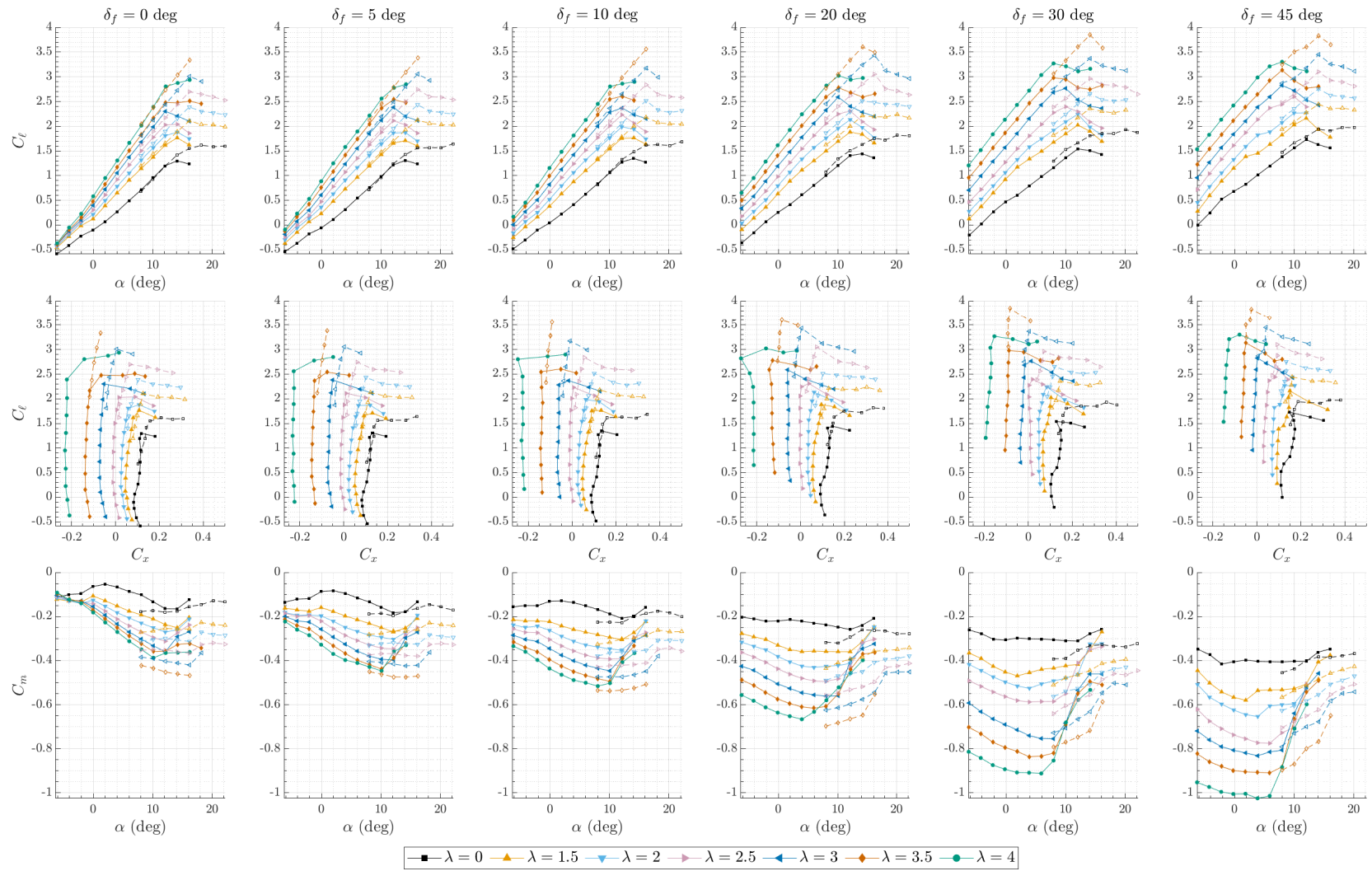


Fig. 25.—Performance coefficients as function of angle of attack, flap deflection, and fan tip-speed ratio.

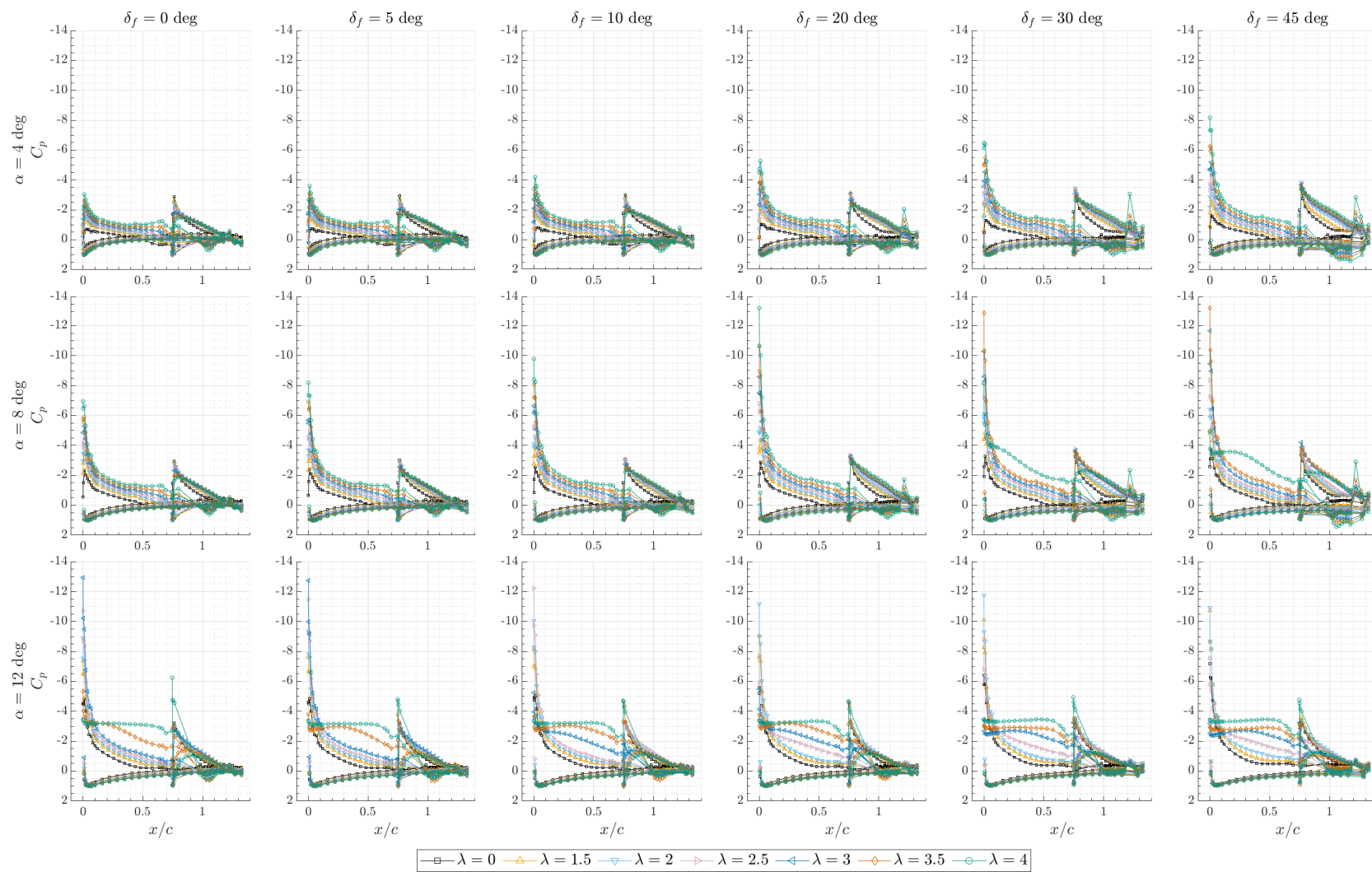


Fig. 26.—Pressure coefficient (C_p) versus axial coordinate (x/c) as function of angle of attack, flap deflection, and fan tip-speed ratio. Clean configuration.

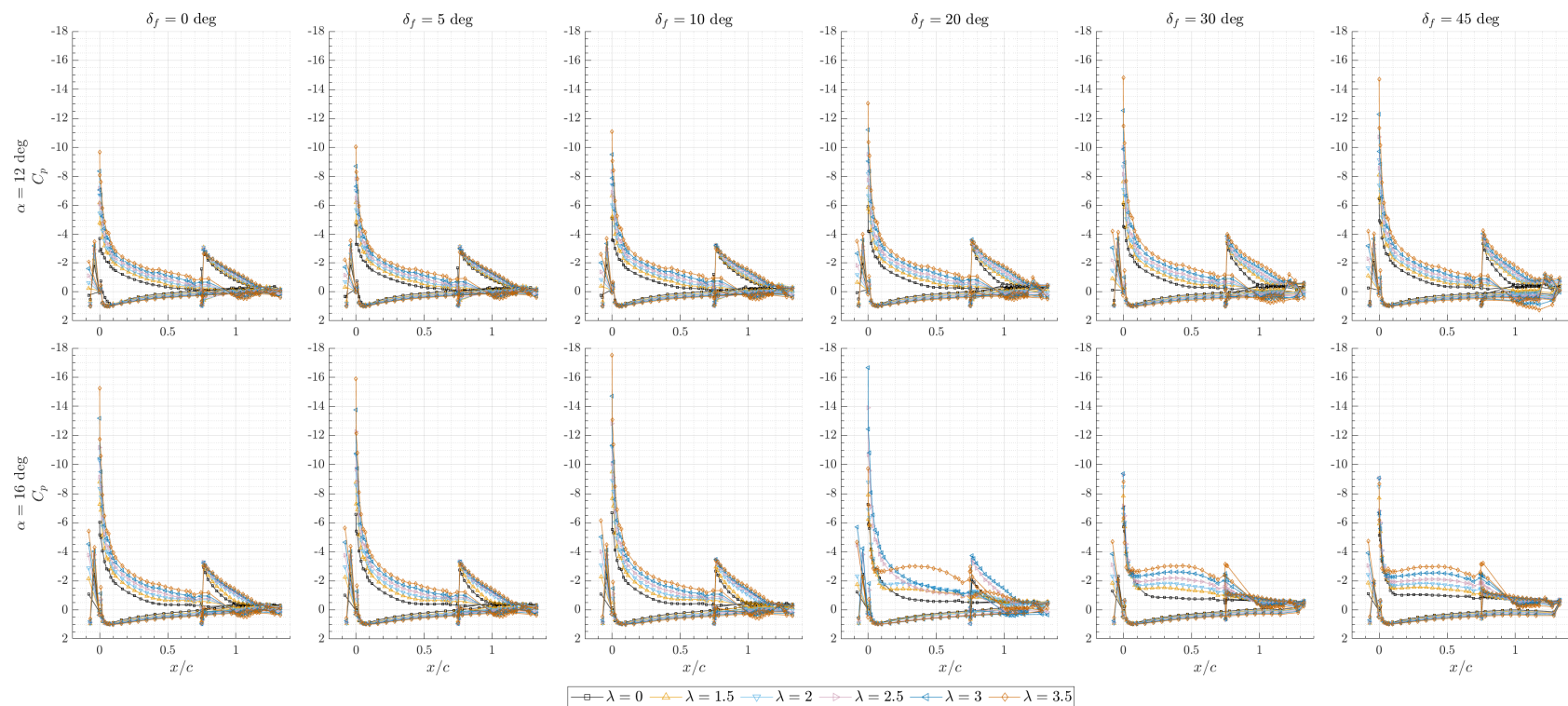


Fig. 27.—Pressure coefficient (C_p) versus axial coordinate (x/c) as function of angle of attack, flap deflection, and fan tip-speed ratio. Krueger flap installed.

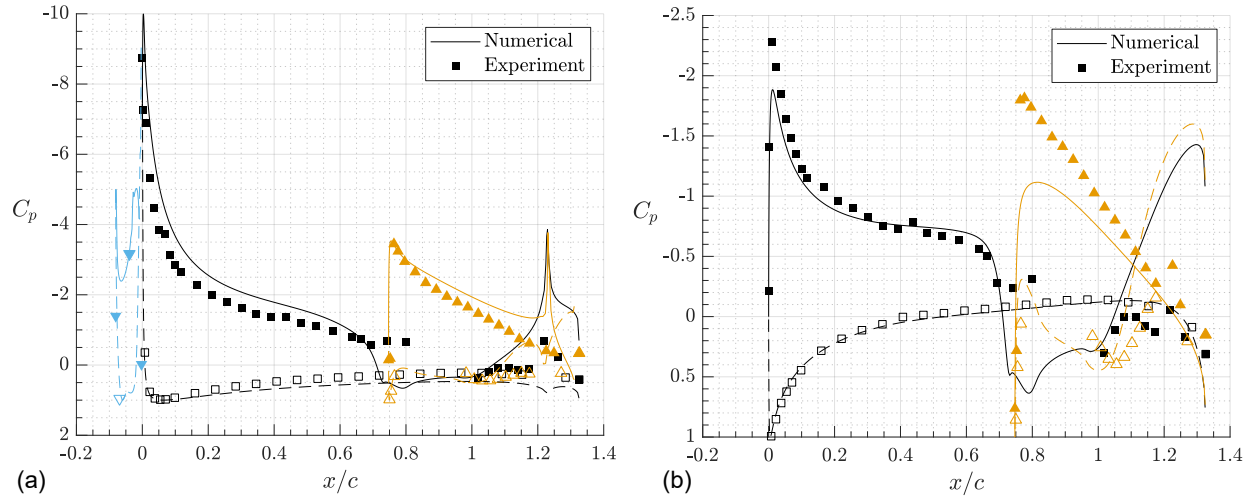


Fig. 28.—Pressure distribution comparison between numerical scheme and experimental data. (a) $\lambda = 3$, $\delta_f = 0^\circ$, clean configuration. (b) $\lambda = 3$, $\delta_f = 20^\circ$, Krueger flap installed.

2.1.2.5 High-Lift Aero-Propulsive Integration: Conclusions

The purpose of this work was to facilitate the design of aero-propulsive geometries, particularly those that leverage DP systems. To that end, the initial validation cases show that the computational scheme presented can reasonably predict the quasi-2D performance of aero-propulsive configurations. In addition, the case study on the adapted aero-propulsive configuration for the CHEETA aircraft concept demonstrates the tool's capability for flow visualization through velocity vector plots and surface pressure distributions. Methods for determining realistic values of C_T provide an additional capability for design practices. Through further experimental validation, we intend to obtain further insight into the performance benefit of DEP and thrust vectoring for high lift. Thrust-drag bookkeeping approaches will help dissect the aerodynamic influence of the propulsive jet from its direct Newtonian force contribution, both of which are key to the overall system performance. Particle-image velocimetry (PIV) experiments are also intended to visualize the evolution of the wake boundaries behind the airfoil system trailing edges, thus providing additional support for the method applied in the computational scheme. Finally, extension of the computational algorithm into three dimensions using a strip-theory approach or vortex ring approach as seen in Reference 51 will broaden the scope of the tool and allow it to capture the spanwise influence of DP systems that are of great interest to the broad aeronautics community.

2.2 Fan Design Strategies

The development of DEP and associated BLI integration strategies introduces new considerations for the design of ducted fan systems. In particular, the use of an electric machine to drive a fan or other compressor results in considerations for machine torque, speed, and thermal performance characteristics that are unique from gas turbine engines. Strategies for distributed power system sizing capable of reconciling the coupling between fan and machine design are further developed in the current study. Furthermore, the design of fan systems capable of accommodating the nonuniform inlet flow of BLI configurations are further developed through the current CHEETA program.

2.2.1 Fan and Motor Co-Optimization for Distributed Electric Propulsion Systems

An integrated modeling approach has been developed for the conceptual design of electric propulsors for the CHEETA aircraft. In this approach, the distributed BLI fan module and the fan-hub embedded SC

electric motors are modeled together using signomial programming (SP). The all-at-once SP optimization eliminates external design iterations between modules and provides sensitivities to design parameters as part of the solution. The approach developed is modular, so component models can readily be added to explore different aircraft architectures. A trade space exploration leads to a conceptual design with nine propulsors powered by 2.4-MW motors, with 14% power savings relative to an equivalent non-BLI twin propulsor configuration. The fan-embedded motor architecture leads to fan hub-to-tip ratios up to 0.5, higher than those typical for fan-alone optimizations.

Under the CHEETA program, supported by NASA, the development of an advanced propulsion system for a fully electric single-aisle transport aircraft powered by cryogenic H_2 fuel cells and fully SC motors was examined. DEP can be used with BLI to increase aerodynamic efficiency and maximize overall system performance (Ref. 9 and 13). A schematic of the proposed CHEETA aircraft concept is shown in Fig. 29(a) along with aircraft mission requirements in Table 4.

The integrated design of the DEP system introduces coupling between the electric machines and the propulsor turbomachinery, with weight and performance trades different from component-only design. For example, rotating electric machines benefit from a large diameter to achieve the lightest, most efficient design for a desired torque, whereas fan weight is minimized for designs with low fan hub-to-tip ratios (0.25 to 0.3). Another key parameter is rotor speed; electric machine performance improves at higher speed (lower torque), whereas a fan design is more efficient at slower rotor speeds, limited by maximum fan blade-tip Mach number. For the CHEETA concept, the electric motors are embedded within the fan-hub in a direct-drive configuration. The optimal rotor speed and machine size requirements for desired torque thus push toward higher-than-typical fan hub-to-tip ratios.

TABLE 4.—CHEETA MISSION REQUIREMENTS

Parameter	Value
Maximum takeoff weight, kg	86,800
Payload, kg (pax)	15,900 (180)
Range, km	5,400
Cruise L/D	18
Cruise Mach number	0.77

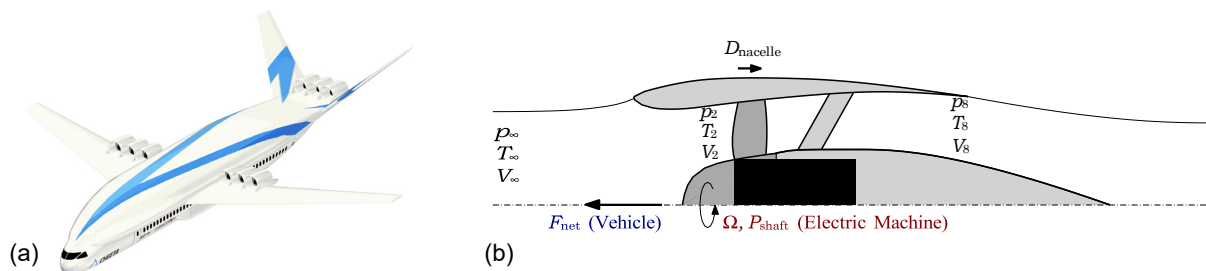


Fig. 29.—CHEETA system configurations. (a) Schematic view of CHEETA aircraft concept. (b) Notional design of ducted fan with embedded electric machine.

To account for these interactions between various components, a co-optimization approach that includes component- and system-level sizing and performance trades is crucial and has been explored by the researchers in References 52 to 54. Hall et al. (Ref. 9) showed about 9% net power savings coming from BLI for a fuselage-mounted twin-propulsor transport aircraft, suggesting that the net power savings can be potentially maximized by combining BLI and DEP. The current authors explored BLI+DEP benefits for an underwing propulsor configuration for the CHEETA mission (Ref. 55) and found DP power savings of about 6% in going from a 2-propulsor configuration to a 32-propulsor configuration. The fan and motor modules were simultaneously optimized for minimum average power consumption, leading to component designs different than if each were designed independently.

In the present research, we develop an integrated modeling approach for electrified aircraft propulsor systems and explore the design space of BLI+DEP electric propulsors to obtain optimal design configurations for the CHEETA aircraft concept. The fan and motor modules are modeled together using a SP optimization framework (Ref. 56), similar to Chandel et al. (Ref. 55), but the model is extended to analyze fuselage and wing distributed propulsors. The formulation ensures fast and robust solutions and provides variable sensitivities to objective function, indicating key design parameters. The unified approach of SP eliminates black-box-type analysis during design iteration based on isolated component design, because the interface constraints emerge as a natural part of the modeling. For maximal BLI benefit, the propulsors are distributed over the fuselage and wing top-surface. The tradeoffs between various sizing and performance parameters determine different sets of propulsors used over the fuselage and wing. The solver indicates a 32-propulsor configuration (28 over-wing and 4 over-fuselage propulsors) being optimal. However, a 9-propulsor configuration is recommended for CHEETA, based on a tradeoff between net power savings, complexity introduced by propulsors of different sizes, and confidence in machine modeling for the selected architecture.

2.2.1.1 Propulsor Design Optimization Model

The DEP system is modeled using a SP toolkit, named GPkit, developed by Burnell et al. (Ref. 57). In this approach, system-governing equations are provided as constraints to optimize a desired objective function. Ensuring the convexity of the optimization problem in log-log space guarantees a global optimum solution; however, the systems constraints must be composed of certain mathematical forms limited to monomials, where a dependent variable $f(x)$ is represented as the product of independent variables raised to constant powers (e.g., $f(x) = cx_1^{a_1}x_2^{a_2} \dots x_n^{a_n}$, $c > 0$), or posynomials, defined as a sum of monomials. When the full-optimization problem can be expressed using only monomials and posynomial expressions, it is called a geometric program (GP) and a global optimal solution is guaranteed. However, some governing equations cannot be expressed using the above forms and must be supplied as signomials where monomials with negative coefficients are allowed. The presence of signomial equations leads to SP formulation, where the optimization problem is solved via sequential solves of its GP approximations. SP solutions are optimized for a local optimum, but global optimality is not guaranteed; therefore, signomial functions to represent a governing equation are used only where necessary. The SP-based formulation is well-suited for rapid optimization and tradespace exploration for aircraft design (Ref. 58) and has been applied to a range of novel aircraft and propulsion concepts (Refs. 59 to 62).

In current modeling, the fan and motor modules of the propulsor system are defined together to formulate a simultaneous optimization problem using SP framework. Governing equations and constraints from the sizing and performance requirements are solved together to minimize the net electric power consumption by the aircraft.

2.2.1.1.1 Ducted Fan Stage

A schematic of the fan meridional view is shown in Fig. 29(b). The propulsor system performance is modeled using one-dimensional compressible flow equations written in SP-compatible form to relate the thermodynamic states between the freestream to fan inlet, fan face, and nozzle exit—an approach similar to aerospace industry-standard cycle performance tools such as Southwest Research Institute’s NPSS[®] (Numerical Propulsion System Simulation) (Ref. 63). More details on fan design parameters can be found in Reference 64.

The main design variables that affect the overall propulsor sizing and performance are fan pressure ratio (FPR), which sets the propulsive efficiency at each operating point, and fan torque, which sets the desired motor performance. The reduction in fan inlet stagnation pressure due to BLI is determined from the kinetic energy defect of the ingested boundary layer, which is estimated based on the fuselage and wing profile drag and the spanwise extent of the ingested boundary layer (Ref. 65).

2.2.1.1.2 Electric Machine

A fully SC, air-core, outer rotor, wound-field synchronous motor is used for the CHEETA aircraft concept. This topology was selected primarily for high power density (critical for aviation applications). Detailed design and optimization of this motor is described in Reference 21. A radial slice of the motor is illustrated in Fig. 30.

The torque generated by this machine is estimated from the electromagnetic shear stress at the rotor surface of a generic radial-flux electric machine (Ref. 66), as described by Eq. (15).

$$T_e = 6\pi \left(\frac{D_{ag}^2 L_{stack}}{4} \right) B_g A \quad (15)$$

The average airgap magnetic flux, B_g , the number of stator slots per pole, and the rotor radial build (i.e., the radial distribution of windings in the rotor) are kept the same as in the 2.5-MW reference machine in Reference 21, optimized with detailed finite-element analysis (FEA). An eight-pole design was selected as the reference machine for balance between low mass and low losses among Pareto-optimal designs.

The strategy of holding select design parameters from this reference machine simplifies the SP optimization while ensuring that the magnetic loading is reasonable for the power level range relevant to the CHEETA mission. The reference machine parameters are included in Reference 64.

While the average airgap flux density, number of slots per pole, and radial build are held constant, the electrical loading (A), stack length, number of poles, and airgap radius vary during the SP optimization.

This architecture has significant alternating-current (AC) losses due to the high-frequency excitation currents and operating temperature threshold (25 K) for the safe operation of the SC coils at selected flux density. The AC loss modeling is based on Reference 67, and the components can be found in the appendix of Reference 64.

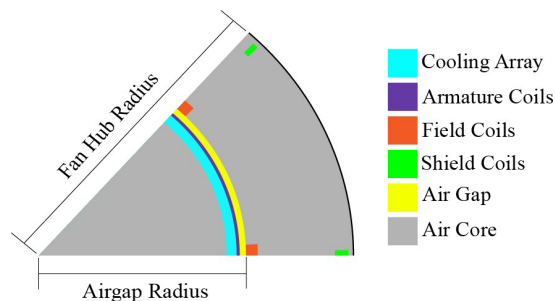


Fig. 30.—Motor radial view for 2.37-MW machine.

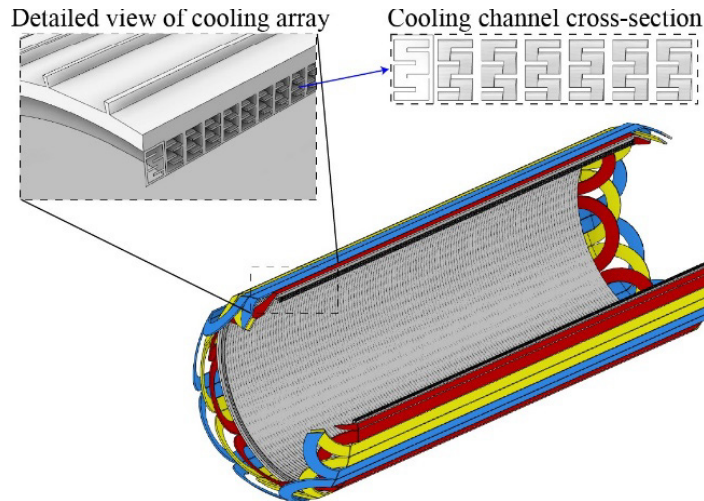


Fig. 31.—Armature windings and interior cooling arrays.

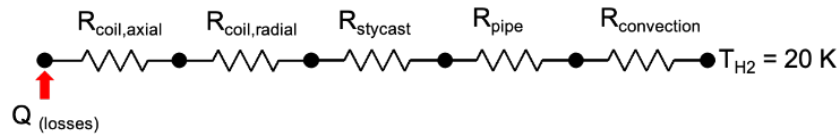


Fig. 32.—One-dimensional motor thermal circuit.

2.2.1.1.3 Thermal Management

The use of a fully SC machine is enabled by free cooling available from the H_2 fuel, that is, the LH_2 is first used to cool the electrical components. It evaporates and warms up during this stage and is then fed to the fuel cells. Cooling for the armature windings is provided by a coil of six parallel channels lining the inside of the winding bundle that carry two-phase (liquid and vapor) H_2 flow from the fuel-cells at an initial temperature of 20 K. Fig. 31 gives an illustration of the cooling system.

The objective of the cooling system is to keep the armature windings temperature below 25 K to maintain superconductivity. This limit is implemented as a constraint in the optimization, using a 1D thermal resistance circuit model for the heat transfer from machine to coolant (Fig. 32). Each resistor in this circuit represents radial heat transfer between two components. The rate of convective heat transfer into the two-phase H_2 is calculated based on the flow speed. The mass flow available for cooling is 0.2 kg/s, approximately 85% of the input to the fuel cell stack. The convective heat transfer coefficient (HTC) is estimated by extrapolating from the data presented in Reference 68.

2.2.1.1.4 System-Level Design Optimization

The ducted fan and electric machine modules are optimized simultaneously to satisfy aircraft thrust requirements during the takeoff, climb, and cruise segments of aircraft mission.

1. Sizing constraints: Distributed propulsor arrays are constrained to fit within the fuselage trailing-edge span and 50% of the wing span, allowing for the remaining wing span to accommodate aerodynamic control and high-lift devices. Nacelle drag is calculated based on the wetted-area approach. With the fan-embedded motor design, motor outer diameter (OD) is bounded by the fan hub diameter. From a motor design perspective, rotating machines with high aspect-ratio (defined as machine axial length divided by machine OD) are more difficult to balance and result in rotor vibrations due to greater twist force experienced along their length (Ref. 69). A conservative limit of two was thus set for the motor aspect ratio pending detailed design and analysis.

2. Performance constraints: Due to the direct-drive design, the required shaft power and rotor speed for each fan is matched with the available power (after losses) and rotor speed of each motor. The fan flow conditions are constrained to meet the aircraft thrust requirements using the power balance method (Ref. 9). For motor thermal management, the coolant flow and maximum thermal loss limits are appropriately supplied for fuselage and wing propulsors.

3. Objective function: The propulsors are sized by minimizing a time-weighted average of the net power (P_{net}) required from the electric machines during takeoff, climb, and cruise segments of the mission.

$$P_{net} = \sum_{fuselage, wing} \left(N_{fans} \frac{\sum t_j P_{I,j}}{\sum t_j} \right) \quad (16)$$

In Eq. (16), index j indicates a mission design point, with t_j being the time spent in the j th segment and $P_{I,j}$ being the net electric power required in that segment. Cruise is the longest mission segment, hence, the objective function is weighted 90% toward the power requirements at cruise conditions.

The low-power segments of the flight (such as descent or ground idle) are not included in this analysis, with an assumption that a propulsor designed to meet the takeoff and cruise power requirements would also be capable of generating the low power required during descent. The contribution of these low-power, short-duration mission segments to the time-weighted overall power requirement is small and can be neglected at this conceptual design stage.

Fig. 33 shows an extended design structure matrix (Ref. 70) for the DEP sizing optimization problem. The system is defined in a modular way, with green blocks representing models for sizing and performance of the individual components; the red box represents a system-level integration model that assesses the objective function as a function of the design variables used in the component models. GPKit employs a simultaneous analysis and design (SAND) optimization architecture, where the individual models do not have inputs and outputs; instead, each component returns values of the monomial and posynomial/signomial constraint functions (represented by f and g , respectively, in the gray boxes in the

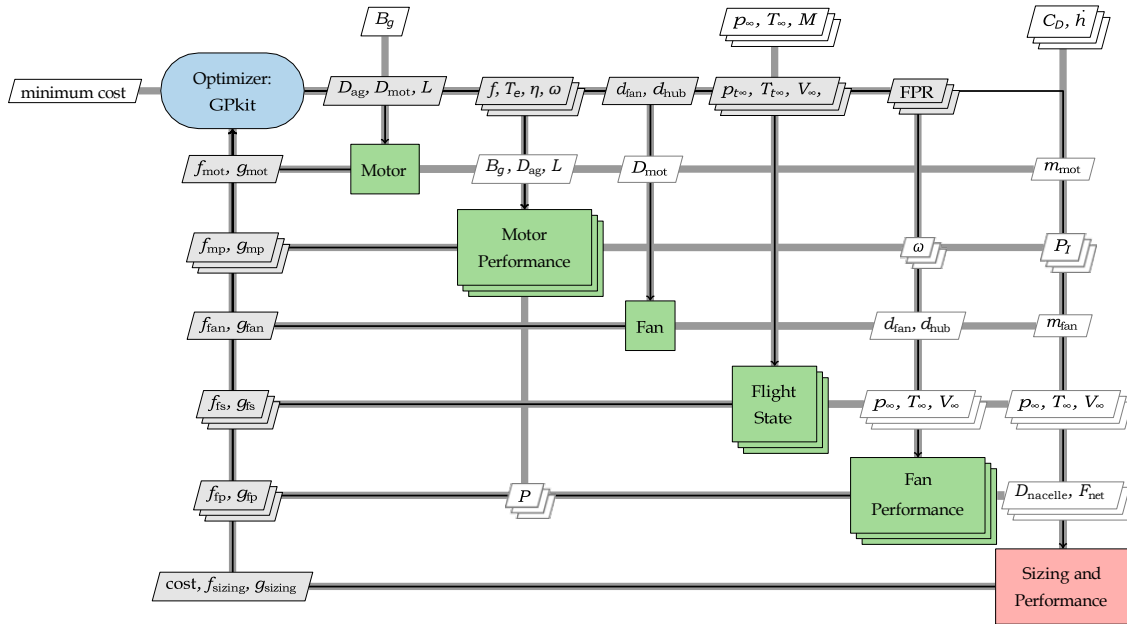


Fig. 33.—Modular flow chart of CHEETA signomial program.

column under the optimizer block) as a function of the design variables (variables in gray boxes in the row to the right of the optimizer block), which the optimizer iteratively updates to approach the minimum cost solution; this information flow is indicated in the figure by the black arrows (see details in Ref. 64).

2.2.1.2 Optimized Propulsor System Designs

Simultaneous optimization of the motor and the fan for minimum average power consumption leads to designs that are different than if each were designed independently. The operating conditions that constrain the sizing of the two components are also different, with the fan sized by the required thrust at top of climb and the motor sized by the required power at takeoff. In this section, the optimized propulsor system configuration for the CHEETA aircraft concept (shown in Fig. 34 to Fig. 36) is analyzed, beginning with a conventional, non-BLI twin propulsor design.

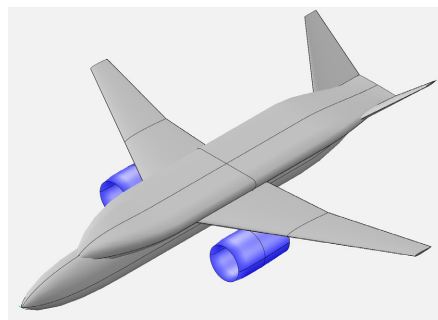


Fig. 34.—CHEETA twin-propulsor configuration without BLI.

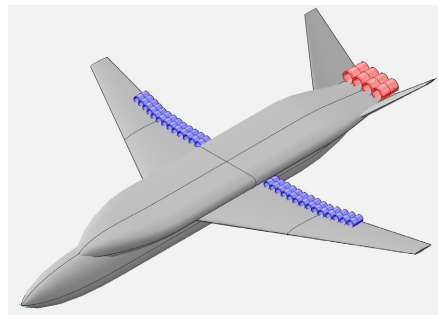


Fig. 35.—32-propulsor configuration for CHEETA aircraft.

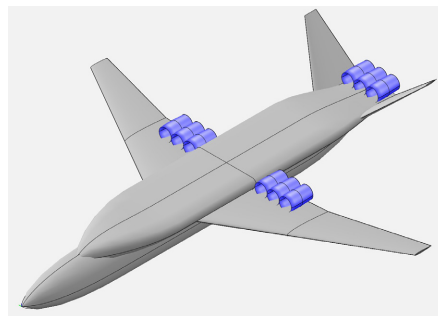


Fig. 36.—9-propulsor configuration for CHEETA aircraft.

2.2.1.2.1 Non-BLI Twin-Propulsor Configuration

This is a reference design for quantifying the power savings of other designs. Due to the fan-embedded motor design and the constraint on maximum motor aspect ratio, a tradeoff between system weight and performance leads to an average fan hub-to-tip ratio of 0.35 at fan inlet, which is higher than the typical range (i.e., 0.25 to 0.3) for advanced civil turbofan design. The maximum power is required at the middle of climb (21 MW), and net cruise power is 16 MW.

2.2.1.2.2 Optimal Design: 32-Propulsor Configuration

The overall system with fuselage and wing propulsors shows the least electric power consumption for a 32-propulsor design (Fig. 35), with 28 wing and 4 fuselage propulsors. Net power savings relative to the non-BLI twin-propulsor configuration is about 16%. Given that the boundary layer growing over the wings is thinner than that on the fuselage trailing edge, the optimized design uses smaller propulsors over the wing for maximal BLI benefit. Due to the distributed arrangement, the wing-motor design pushes the fan hub-to-tip ratios even further (i.e., 0.5). A simplification in design using identical fuselage and wing propulsors leads to a 30-propulsor configuration with 17% net power savings (24 wing and 6 fuselage propulsors, using 0.7-MW motors with a four-pole design).

However, such highly distributed propulsor configurations are undesired due to increased complexity, number of parts, and usage of sub-MW, fully SC motors where technology readiness level (TRL) is significantly lower (less than 3). In addition, the rotor radial-build for the current fully SC motors is kept the same as the 2.5-MW peak power optimized design from Reference 21. Further investigations are required to identify the optimal radial build for sub-MW range, fully SC machines, required by highly distributed configurations. For this reason, the configuration described in the following section uses a prespecified number of propulsors to keep per-motor power closer to 2.5 MW.

2.2.1.2.3 Selected Design: 9-Propulsor Configuration

A 9-propulsor configuration using 2.37-MW motors was analyzed, given that the confidence level on motor modeling is higher for this power range. Design complexity is also lower with identical propulsors and smaller number of parts. Net cruise power is 13.7 MW, about 14% lower than the conventional non-BLI twin-propulsor arrangement. A schematic of this design is shown in Fig. 36.

Net average electric power requirements at five CHEETA mission operating points have been compared between the 2-propulsor and 9-propulsor designs. The conventional twin-propulsor configuration requires 21-MW maximum power during the middle- and top-of-climb (design point with maximum corrected power), whereas the 9-propulsor configuration using BLI+DP requires only about 19-MW maximum power.

The gains in this configuration via design simplification, confidence in motor modeling, and technical readiness level on 2.5-MW range machine architecture outweigh the relatively small penalty on net power savings as compared to the 32-propulsor configuration. Based on this, the 9-propulsor configuration was recommended for the CHEETA aircraft concept.

Design parameters for 2-, 9-, and 32-propulsor configurations are presented in Table 5. All designs converge to a similar level of peak electric loading, indicating an optimal electromagnetic operating point that is scale invariant (though the machine rotor build is kept the same). A key difference lies in the machine airgap diameter and split ratio (defined as the ratio of the machine airgap diameter to the OD), both being more favorable for the larger machines. The electromagnetic torque is produced in the airgap and the split ratio governs the available space within the inner bore for thermal management features, hence higher values of both these parameters are desired. The smaller dimensions and lower split ratio of the low-power motors makes their detailed design more challenging.

As mentioned earlier, the simultaneous optimization of motor with the fan leads to a motor design different than machine-only optimized design. This is further explored for the 2.37-MW motors used for 9-propulsor configuration.

Table 6 presents key motor parameters for a simultaneously optimized design, compared with machine-only optimized designs, using the same pole-count, or same rotor speed, respectively, as the simultaneously optimized design. The machine-only optimization has no constraints on the machine OD or axial length, and motor coolant flow rate and thermal management requirements are kept the same for all designs to have a back-to-back comparison.

TABLE 5.—PROPULSOR DESIGN PARAMETERS
WITH VARIED CONFIGURATION

Parameter	2 propulsors	9 propulsors	32 propulsors
Wing propulsors			
Motor power, MW	10.6	2.1	0.47
Max. speed, rpm	2,210	4,060	8,420
Max. electric loading, A/m	42,200	42,400	42,900
Airgap diameter, m	0.7	0.32	0.13
Motor OD, m	0.89	0.51	0.32
Split ratio	0.79	0.63	0.4
No. of poles and slots	18 and 108	8 and 48	4 and 24
Fan OD, m	2.56	1.35	0.64
Fan hub-to-tip ratio	0.35	0.38	0.4
Per motor mass, kg	95	50	30
Per fan mass, kg	3,950	850	140
Fuselage propulsors			
Motor power, MW		2.37	1.63
Max. speed, rpm		3,900	5,040
Max. electric loading, A/m		42,400	42,500
Airgap diameter, m		0.32	0.24
Motor OD, m		0.51	0.43
Split ratio		0.63	0.56
No. of poles and slots		8 and 48	6 and 36
Fan OD, m		1.35	1.07
Fan hub-to-tip ratio		0.38	0.5
Per motor mass, kg		50	40
Per fan mass, kg		850	480
Propulsor mass, kg	8,090	8,070	6,900
Net cruise power, MW	15.9	13.7	13.3

TABLE 6.—MOTOR DESIGN PARAMETERS FOR SIMULTANEOUS VERSUS MACHINE-ONLY CONFIGURATION

Parameter	Simultaneous optimization	Machine-only optimization	Machine-only (same rpm)
Motor max. power, MW	2.37	2.37	2.37
Max. speed, rpm	4,060	6,970	4,060
Airgap diameter, m	0.32	0.30	0.46
OD, m	0.51	0.49	0.65
Axial length, m	1.01	0.50	0.38
Motor aspect ratio	2	1.02	0.58
Total armature length, km	35	18	21
No. of poles and slots	8 and 48	8 and 48	12 and 72
Per motor mass, kg	50	28	26

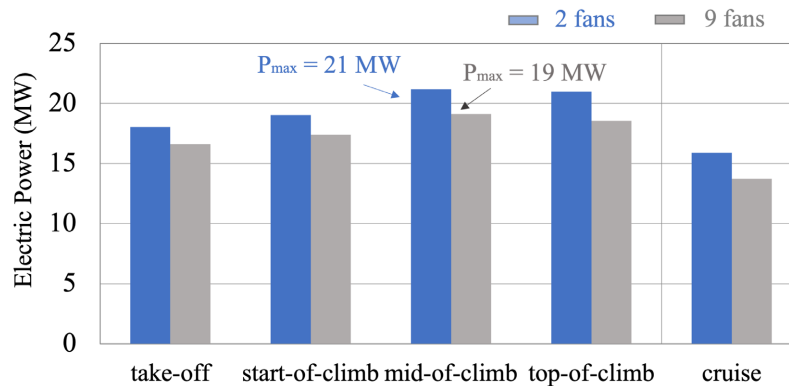


Fig. 37.—Net electric shaft power at CHEETA mission design points for 2-propulsor configuration versus 9-propulsor configuration.

The machine-only optimization with same pole-count (Table 6, column 3) prefers high rotor speed to reduce torque, whereas the presence of the fan pushes the optimal rotor speed lower. A further optimization of just the machine at this optimal speed (column 4), results in a motor design with higher airgap diameter and a corresponding increase in the OD and number of poles, which would require a larger fan hub diameter. A final comparison of the net power requirements for the 2-propulsor and 9-propulsor configurations, across various phases of the aircraft mission, is shown in Fig. 37.

2.2.1.2.4 Motor Characteristics

The estimated efficiency of the SC motors is above 99% at all operating conditions. Fig. 38 shows the effect of the number of propulsors on the optimal motor sizing, in which individual motor torque (on a logarithmic scale) is plotted as a function of angular speed at each mission design point (i.e., takeoff, start-, middle- and top-of-climb, and cruise). The curves indicate contours of constant power per motor. With DP, torque requirement per motor is smaller ($\propto 1/N$), and per motor mass follows a power-law scaling similar to the fan mass scaling mentioned previously. Since the motor dimensions reduce faster with distribution than the power/torque requirements per motor, higher shaft speeds are required to generate the desired torque. The lower power wing- and fuselage-motors in the 9- and 32-fan configurations appear to operate over a broader range of rotor RPM than the 2-fan design; however, the variation relative to the highest RPM within each configuration is similar (i.e., 35% to 40%).

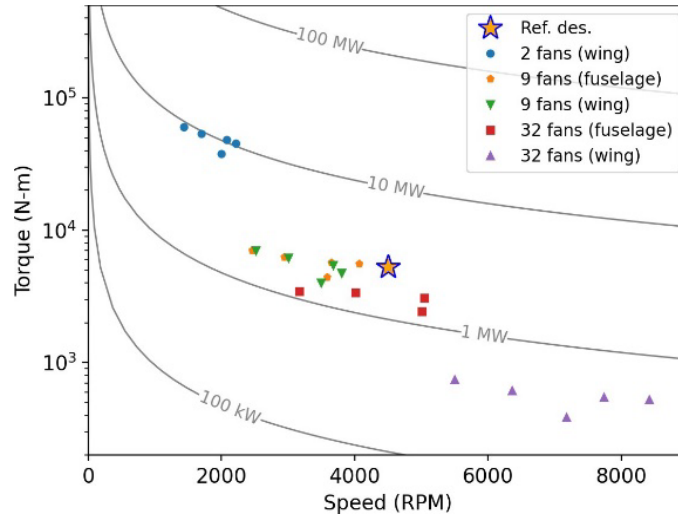


Fig. 38.—Motor torque versus speed at CHEETA mission operating points for wing and fuselage motors in 2-, 9- and 32-propulsor configurations.

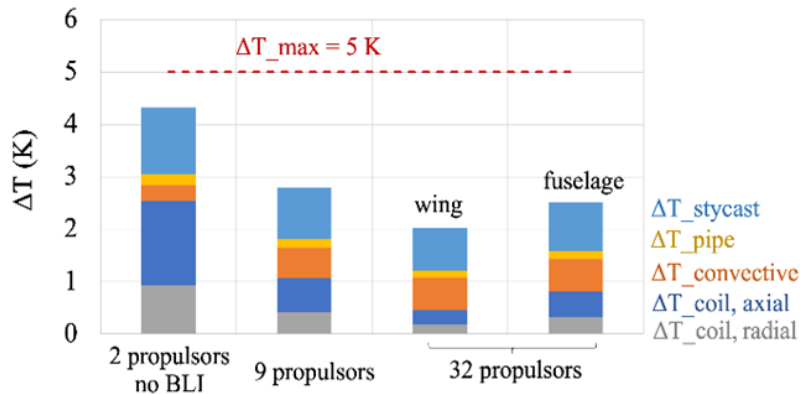


Fig. 39.—Motor temperature rise with varied propulsor configuration.

2.2.1.2.5 Motor Thermal Management

The coolant H_2 enters into the motor heat exchanger (HEX) system at 20 K, hence, the net temperature rise within the motor must be less than 5 K. The temperature rise across various components of the thermal circuit is given in Fig. 39. The twin-propulsor design has the maximum temperature rise. With higher number of propulsors, net cooling surface area increases, resulting in lower motor temperatures. For 9-propulsor and 32-propulsor configurations, motor operating temperatures stay below 23 K.

2.2.1.2.6 Sensitivity Analysis

The SP solver (GPKit) outputs the sensitivity of design variables to the optimized function, indicating the most significant design features. Key motor parameter sensitivities, that is, the relative percent change in the cost function (i.e., P_{net}) due to 1% increase in a selected design parameter, can be found in Reference 64. All these parameters are supplied as a constant, with values based on the reference machine (Ref. 53). Based on the solver output, the difference between the airgap and motor outer-diameter (d_c), peak magnetic flux density (B_g), maximum conductor current density (J_c), and so forth are some of the dominant design features. However, the effect of any uncertainties in supplied parameter values on the optimized aircraft

power consumption is minuscule. For a 1% change in any variable, the impact on net aircraft power is less than 0.01%. This demonstrates the robustness of our design approach and optimized solutions.

2.2.1.3 Fan Design: Conclusions

An integrated modeling approach has been presented to define the trade space of cryogenic-electric aircraft propulsion systems. The electric and propulsive components for NASA's distributed electric BLI CHEETA aircraft propulsors are modeled together using a SP framework, where the overall system is optimized for net energy consumption during the takeoff, climb, and cruise segments of the flight mission.

Relative to the reference non-BLI electric twin wing-propulsors design, the maximum net power savings of 16% are achieved for a 32-propulsor configuration, where the fuselage and wing propulsors are slightly different from each other. A simplified 30-identical-propulsor configuration with 0.7-MW motors uses 0.5% lower power than the 32-propulsor design. However, these highly distributed propulsor designs would require sub-MW, fully SC motor designs, whereas the current motor modeling is best suitable for higher power, for example, ≥ 1 MW, because of the fixed overhead imposed by the cryogenic subsystem. Based on system complexity, performance, and technological readiness levels, a 9-propulsor configuration requiring 2.37 MW per motor with 14% net power savings is recommended as the optimal design for the CHEETA aircraft.

The simultaneously optimized fan-hub embedded motor design for 9-propulsor configuration is much different than the motor-only optimization. The isolated motor optimization attempts to generate the same torque with least machine weight by choosing a smaller axial length and/or higher rotor speed. The difference between designs demonstrates the need for co-optimization. The motor operating temperature is maintained within a specified limit of 25 K in all configurations. With increased distribution, motor thermal management improves because the cooling surface area is increased.

The integrated system-level modeling approach developed here is generic, and component models can readily be added or modified to examine various propulsor architectures with desired level of fidelity. The usage of an SP framework enables fast design iterations, and the in-built sensitivity analysis leads to quick identification of both key design variables and mechanisms.

2.2.2 Nonaxisymmetric Design Strategy for Boundary-Layer Ingesting Fans

A conceptual design approach has been developed to mitigate, or even reduce to zero, the impact of circumferential flow distortion—such as that created by BLI on fan rotor efficiency and vibratory structural forcing. The idea is to create, and tailor, nonaxisymmetric fan exit static-pressure nonuniformities by varying the fan exit guide vane (FEGV) leaving angles that provide an upstream static pressure field to reduce the variations in rotor incidence angle and unsteady rotor forces relative to those with axisymmetric FEGV. It is found that this type of design need not involve a tradeoff between performance and aeromechanics; a nonaxisymmetric aerodynamic design can improve both simultaneously. An analysis of the impact of stage design parameters has been carried out to show how FEGV exit flow angle variations shield the rotor from the adverse effects of circumferential distortions representative of BLI, and the results indicate that inlet distortion effects can be mitigated with realizable nonaxisymmetric FEGVs.

A challenge associated with BLI is inlet distortion (i.e., nonuniformity in velocity and pressure at the inlet to the fan), which reduces fan efficiency and creates unsteady aerodynamic forces on the fan. In this section, we define design strategies for nonaxisymmetric FEGVs or stators to mitigate the adverse effects of inlet distortion. The analysis was published during the ULI program (Ref. 71); we include the relevant parts of that publication here and refer the reader to the original paper for more details on the approach and results.

2.2.2.1 Flow Field Description

We consider a 2D flow on the blade-to-blade domain shown in Fig. 40. The domain is periodic in the circumferential direction over $y = \theta R \in [0, 2\pi R]$ and extends half a circumference, πR , in the axial direction upstream of the rotor and downstream of the stator. The fan rotor and FEGV are represented with two discrete ADs, shown as vertical lines in Fig. 40. The solution has discontinuities in flow variables across the disks that represent the change in velocity, pressure, and temperature across the rotor and stator.

2.2.2.1.1 Equations of Motion

The mechanisms of fan-distortion response away from stall are mainly inviscid, and the flow in the domains outside the blade rows can be described using the Euler Equations,

$$\frac{\partial \mathbf{U}}{\partial t} + \frac{\partial \mathbf{F}}{\partial x} + \frac{\partial \mathbf{G}}{\partial y} = 0 \quad (17)$$

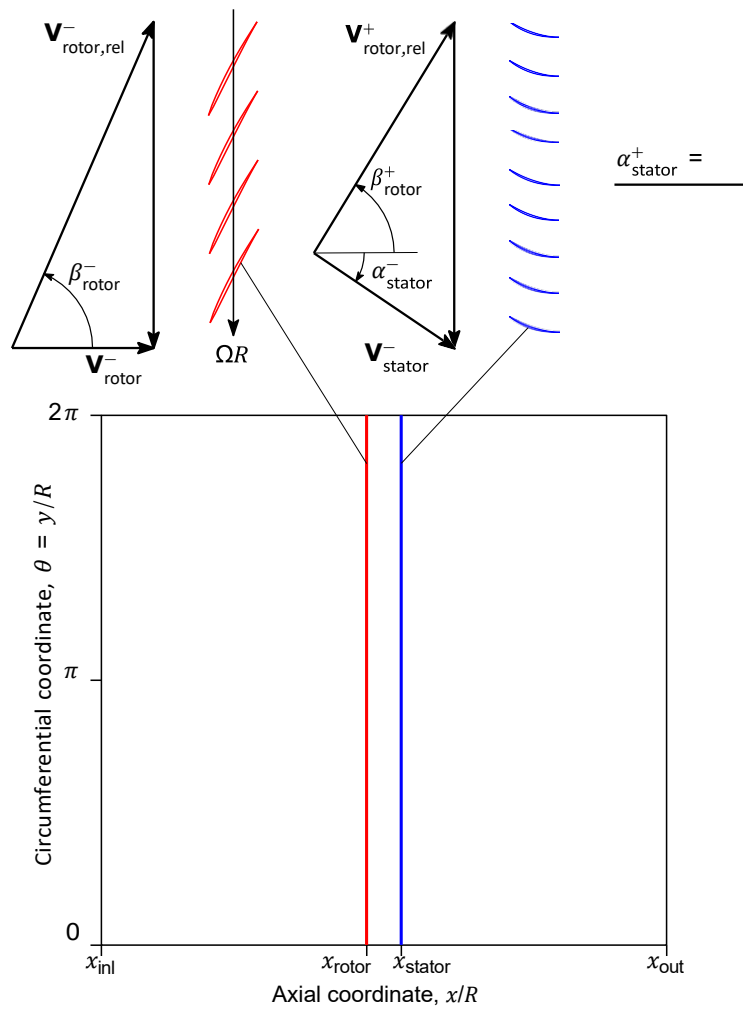


Fig. 40.—Flow domain and mean flow velocity triangles.

The conservation variables, U , and the flux vectors, F and G , are given by

$$\mathbf{U} = \begin{bmatrix} \rho \\ \rho u \\ \rho v \\ \rho e_t \end{bmatrix}, \mathbf{F} = \begin{bmatrix} \rho u \\ \rho u^2 + p \\ \rho uv \\ \rho u h_t \end{bmatrix}, \mathbf{G} = \begin{bmatrix} \rho v \\ \rho uv \\ \rho v^2 + p \\ \rho v h_t \end{bmatrix} \quad (18)$$

where ρ is the density, u and v are the x - and y -components of velocity, p is the static pressure, e_t is the sum of the internal energy and the kinetic energy per unit mass, and h_t is the stagnation enthalpy. The flow is taken as an ideal gas with a constant specific heats.

2.2.2.1.2 Linearized Analysis

We assume the flow can be described as a steady, axisymmetric (i.e., invariant in t and y) mean flow plus a small, periodic perturbation in y and t :

$$\mathbf{U} = \bar{\mathbf{U}} + \delta\mathbf{U}(x)e^{i(ky+\omega t)} \quad (19)$$

where $\delta\mathbf{U}: \mathbb{R} \mapsto \mathbb{C}^4$, $k = 2\pi n/L = n/R$ is the circumferential wavenumber of the periodic variation, and ω is the frequency of the unsteady variation. We restrict the analysis that follows to a single harmonic component but emphasize that a general periodic perturbation can be treated using the sum of perturbations for multiple wavenumbers and frequencies, provided they are small enough relative to the mean flow to be described by a linear combination ignoring their products.

The mean flow, $\bar{\mathbf{U}}$, is constant within each domain. Equation (17) can be expressed in terms of the Jacobian matrices of $\mathbf{F}$ and $\mathbf{G}$ with respect to $\mathbf{U}$ using the chain rule, rearranging, and ignoring second- and third-order perturbation terms, and the equations of motion for the perturbation flow, $\delta\mathbf{U}$, form a linear system of differential equations:

$$\frac{d}{dx}(\delta\mathbf{U}) = \left[-i \left(\frac{\partial \mathbf{F}}{\partial \mathbf{U}} \right)^{-1} \left(k \frac{\partial \mathbf{G}}{\partial \mathbf{U}} + \omega I \right) \right] \delta\mathbf{U} \quad (20)$$

The perturbation flow thus takes the form

$$\delta\mathbf{U} = \sum_{i=1}^4 c_i e^{\lambda_i x} \mathbf{w}_i \quad (21)$$

where $c_i \in \mathbb{C}$, and λ_i and $\mathbf{w}_i$ are the eigenvalues and eigenvectors of the bracketed term in Eq. (20).

We assume the period of the unsteady flow experienced by the rotor in the inlet distortion is much longer than the blade through-flow time, so the reduced frequency $\Omega c/u$ of the (once per revolution) unsteadiness within the fan blade row is small. Further, the AD representation of the fan does not describe the blade-to-blade pressure field, which would appear as higher reduced frequency (blade passing frequency) unsteadiness in the stationary frame. The flow is thus treated as quasi-steady, and we neglect the ωI term in Eq. (20).

2.2.2.1.3 Blade Row Matching and Boundary Conditions

The flow field solution consists of eight unknowns in each region outside the blade rows: the mean flow conservation variables, $\bar{\mathbf{U}}$ and the constants $c_1, \dots, c_4$ in Eq. (21). Solution for the flow domain in Fig. 40 thus requires 12 mean flow and 12 perturbation flow constraints relating quantities in adjacent domains or boundary conditions specified at the upstream and downstream boundaries.

The changes in the flow variables across rotor and stator are defined by relations for continuity, relative flow angle at the trailing edge of each blade row, work input by the rotor (Euler turbine equation), and the efficiency of the work input,

$$(\rho u)^+ = (\rho u)^- \quad (22)$$

$$\Omega R = u^+ \tan \beta^+ - v^+ \quad (23)$$

$$h_t^+ - h_t^- = \Omega R (v^- - v^+) \quad (24)$$

$$\frac{p_t^+}{p_t^-} = \left(\frac{h_t^+}{h_t^-} \right)^{\frac{\gamma}{\gamma-1}} \quad (25)$$

where p_t is the stagnation pressure. In Eq. (25), we have assumed isentropic flow, but a loss model could be included to describe changes in entropy across rotor and stator if desired.

Four additional boundary conditions are required for the mean flow and the perturbation flow, respectively, to define the problem. For the mean flow, we define the inlet static pressure as a reference and specify the Mach number, stagnation pressure, and swirl angle at the upstream boundary. For the perturbation flow, we specify a static pressure perturbation at the downstream boundary and combinations of stagnation pressure, velocity, or swirl at the upstream boundary.

The 2D flow field can thus be described based on a small number of nondimensional input parameters,

$$\left[\frac{\rho}{(\bar{p}_{t,inl} - \bar{p}_{inl})/\bar{a}_{inl}^2}, \frac{u}{\bar{a}_{inl}}, \frac{v}{\bar{a}_{inl}}, \frac{p - \bar{p}_{inl}}{(\bar{p}_{t,inl} - \bar{p}_{inl})/\bar{a}_{inl}^2} \right] \\ = \mathcal{F} \left(\phi, \psi, \frac{\Delta x}{R}, \bar{M}_{inl}, \frac{\delta p_{t,inl}}{(\bar{p}_{t,inl} - \bar{p}_{inl})}, \delta \alpha_{stator}^+, \frac{\delta p_{out}}{(\bar{p}_{t,inl} - \bar{p}_{inl})} \right) \quad (26)$$

In Eq. (26), $\phi = \bar{u}_{inl}/(\Omega R)$ is the stage flow coefficient, $\psi = \Delta \bar{h}_t/(\Omega R)^2$ is the stage stagnation enthalpy rise coefficient, Δx is the axial distance between the rotor and the FEGV, and $\bar{M}_{inl}$ is the mean axial Mach number at the domain inlet. We introduce circumferential nonuniformities through perturbations in inlet stagnation pressure, $\delta p_{t,inl}$, stator exit flow angle, $\delta \alpha_{stator}^+$, or downstream static pressure, δp_{out} .

2.2.2.2 Nonaxisymmetric Fan Exit Guide Vanes

Unlike the fan rotor, a stationary FEGV can have variable geometry around the circumference. To characterize this, we introduce circumferential variations in the FEGV exit absolute flow angle,

$$\alpha_{stator}^+ = |\delta \alpha_{stator}^+| d^{ik(\theta+\varphi)} \quad (27)$$

In Eq. (27), φ is the phase angle between the inlet stagnation pressure distortion and the stator exit flow angle perturbation. The nonaxisymmetric swirl angle creates a nonaxisymmetric static pressure at the stator exit which, modified by the nonaxisymmetric pressure rise across both blade rows, is felt at the rotor inlet. The harmonic variation in FEGV exit flow angle $\delta \alpha_{stator}^+$ is an independent design variable, which introduces the possibility of inverse design allowing specification of any one harmonic flow variable at any one axial location. In particular, we are interested in FEGV designs that can reduce to zero the rotor incidence angle and unsteady force.

2.2.2.2.1 Design for Minimum Rotor Incidence Distortion

Fig. 41(a) shows the rotor incidence distortion, $\delta\beta_{rotor}^-$, for the mean flow stage design parameters, with variations in nonaxisymmetric FEGVs. The horizontal and vertical axes show the two design variables, magnitude and phase of the FEGV exit swirl angle distortion, $|\delta\alpha_{stator}^+|$ and φ , relative to the inlet distortion. The contours are rotor incidence magnitude, normalized by the inlet stagnation pressure coefficient perturbation magnitude, $|\delta p_{t,inl}|/(\bar{p}_{t,inl} - \bar{p}_{inl})$.

There is a zero of incidence distortion for an exit swirl perturbation that is approximately 90° out of phase with the inlet distortion. The magnitude of the stator exit angle perturbation at this zero rotor incidence design is roughly 40% greater than the incidence distortion magnitude for the axisymmetric FEGV case. In other words, a $\pm 5^\circ$ incidence distortion, representative of a BLI inlet distortion (Ref. 72), could be eliminated with a roughly $\pm 7^\circ$ variation in stator exit angle.

The upstream influence and transfer of the stator exit swirl distortion across stator and rotor is complicated, but the end result can be understood in terms of the modification to the static and stagnation pressure fields, depicted in Fig. 42. The nonaxisymmetric FEGV generates a static pressure perturbation downstream of the rotor that is nearly in phase with the inlet stagnation pressure distortion, accelerating the low stagnation pressure flow more strongly than for the stage with axisymmetric FEGVs. This reduces the rotor inlet axial velocity perturbation, increases the circumferential velocity perturbation, and brings them into phase. The rotor inlet velocity components are illustrated in Fig. 43, in which it is seen that the mean and perturbation velocities are aligned. The magnitude of the rotor-relative velocity perturbation has increased relative to the axisymmetric FEGV case, leading to an amplification of the stagnation pressure perturbation across the rotor. The nonaxisymmetric FEGV thus generates a pressure field that shields the upstream rotor from incidence angle variations, albeit at a cost of increased rotor velocity distortion and downstream stagnation pressure distortion.

2.2.2.2.2 Design for Minimum Unsteady Rotor Force

Fig. 41(b) shows the unsteady rotor force, δC_F , with the same nonaxisymmetric FEGV designs as in Fig. 41(a). The trends in unsteady force are similar to those for the incidence distortion, but the phase angle between inlet stagnation pressure distortion and stator exit swirl distortion is different for the zero rotor force design than for the zero rotor incidence design. The stator exit swirl perturbation magnitude, however, is similar between the two cases; in other words, the nonaxisymmetric FEGV design to eliminate incidence distortion for this stage could be rotated approximately 90° about the axis of rotation to yield a design that would nearly eliminate the unsteady rotor force.

Velocity triangles for zero rotor unsteady force are also indicated in Fig. 43. As for the zero incidence design, the axial and circumferential perturbation velocities are in phase. To achieve a constant blade force, however, increases in velocity must be accompanied by decreases in incidence angle. The slope of the locus of perturbation velocities is thus smaller than the slope of the relative velocity vector. An FEGV design for zero unsteady rotor force thus has a nonzero rotor incidence distortion.

The zero rotor force FEGV design increases the stator inlet flow angle variation relative to the axisymmetric design (Ref. 71). The reason is that eliminating the unsteady rotor force eliminates the circumferential variation in work done and hence the distortion attenuation across the rotor, resulting in an increase in the distortion seen by the stator. For a known inlet distortion, however, the stator geometry can be designed to accommodate a nonuniform rotor exit flow (Ref. 73). A nonaxisymmetric FEGV design strategy may thus have the potential to improve the distortion-tolerance of both rotor and stator: a circumferential trailing edge angle distribution can be designed to reduce unsteady variations in rotor blade force or incidence angle, and a different circumferential leading-edge distribution can be designed to match the rotor exit flow angle distortion.

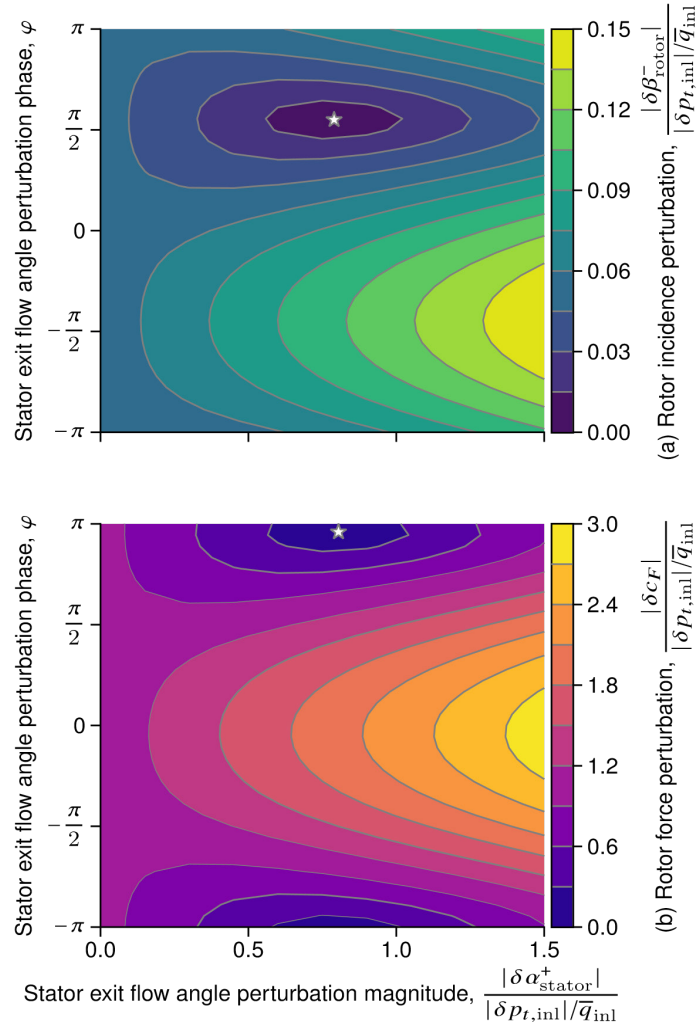


Fig. 41.—Stator exit flow angle perturbation magnitude and phase versus (a) rotor incidence angle perturbation and (b) unsteady rotor force perturbation. Location of zero perturbation indicated by ★; $\phi = 0.43$; $\psi = 0.29$; $\Delta x/R = 0.41$; $M_{\text{inl}} \ll 1$.

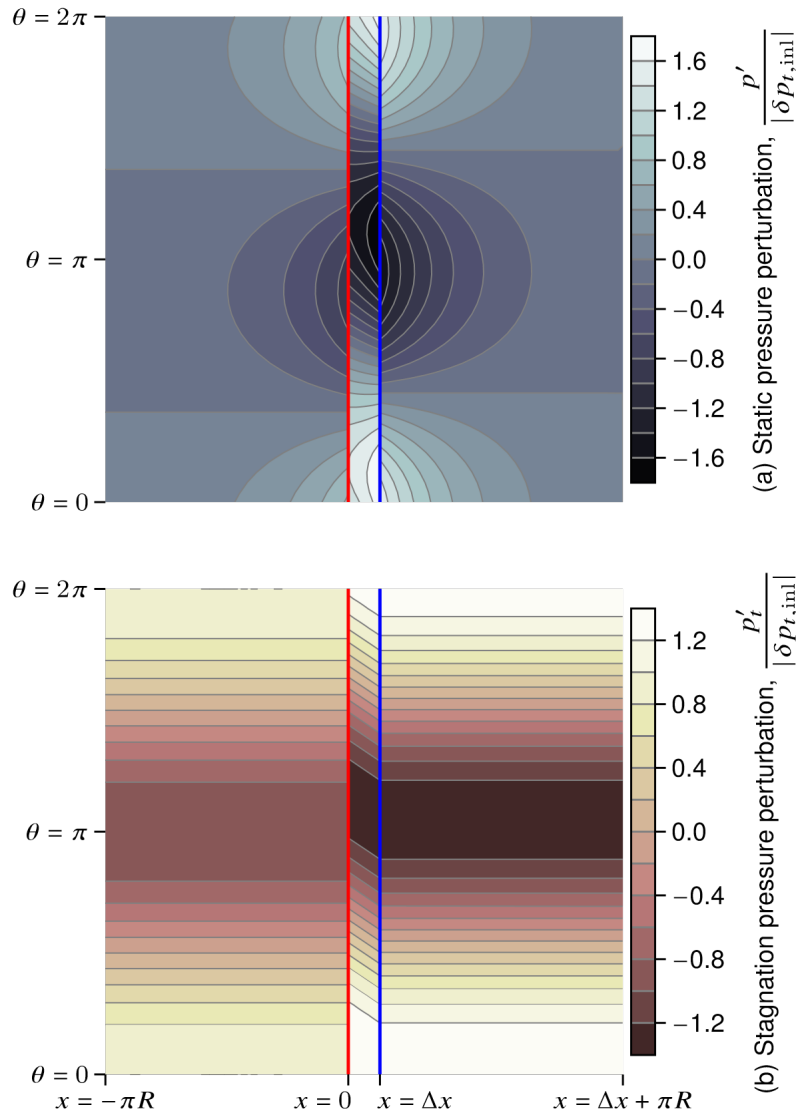


Fig. 42.—(a) Static and (b) stagnation pressure coefficient perturbation fields for fan stage with nonaxisymmetric FEGV for zero rotor incidence with inlet stagnation pressure perturbation; $\phi = 0.43$, $\psi = 0.29$, $\Delta x/R = 0.41$, $M_{\text{inl}} \ll 1$.

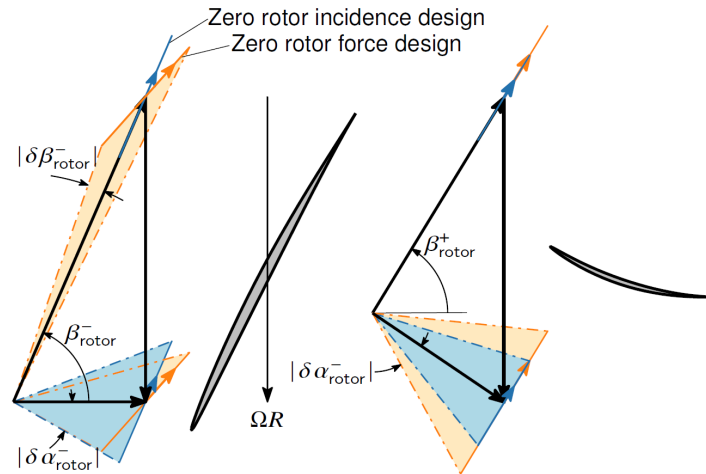


Fig. 43.—Velocity triangles for stage with nonaxisymmetric FEGVs for zero rotor incidence and zero unsteady rotor force with inlet stagnation pressure distortion.

2.2.2.2.3 Multi-Objective FEGV Design

Previously, we saw that the nonaxisymmetric FEGV design for minimum unsteady rotor force has nonzero rotor incidence variation. A corollary is that a design for minimum incidence distortion will have nonzero unsteady force. One would like to be able to eliminate both types of distortions, but because this is not possible, the FEGV design becomes a multi-objective optimization with a balance needing to be struck between incidence distortion (a surrogate for fan efficiency) and unsteady rotor force.

To illustrate the trade between incidence distortion and unsteady force mitigation, Fig. 44 shows the magnitude of both these design metrics in the FEGV design space of Fig. 41. The horizontal axis is the normalized rotor incidence perturbation magnitude, and the vertical axis is the normalized unsteady rotor force magnitude. Each point is shaded by the magnitude of the stator exit swirl angle perturbation for the design, with the scale given on the right-hand side of the figure; the single black point at the center of the concentric rings represents the axisymmetric FEGV. In the figure, the designs for zero incidence distortion and zero unsteady force appear as points on the horizontal and vertical axes, respectively, and there is a Pareto frontier along the straight line between the two. Designs along this line are optimal in the sense that one of the two objective functions cannot be decreased without an increase in the other.

The characteristics of the Pareto frontier in Fig. 44 provide insight into using nonaxisymmetric FEGVs to mitigate inlet distortion. About half the Pareto optimal designs have unsteady force and incidence angle perturbations that are smaller than those for the axisymmetric FEGV design. The implication is that a well-designed nonaxisymmetric FEGV can improve both efficiency and aeromechanics of a fan with a known inlet distortion. In other words, one need not sacrifice efficiency to achieve reductions in vibratory stresses on the rotor. Further, the stator exit angle perturbations along the Pareto frontier are smaller than at the end points for the zero incidence and zero unsteady force designs. This implies that Pareto optimal designs that balance between efficiency and aeromechanics will require smaller FEGV exit angle variations than those that target incidence or unsteady loss alone.

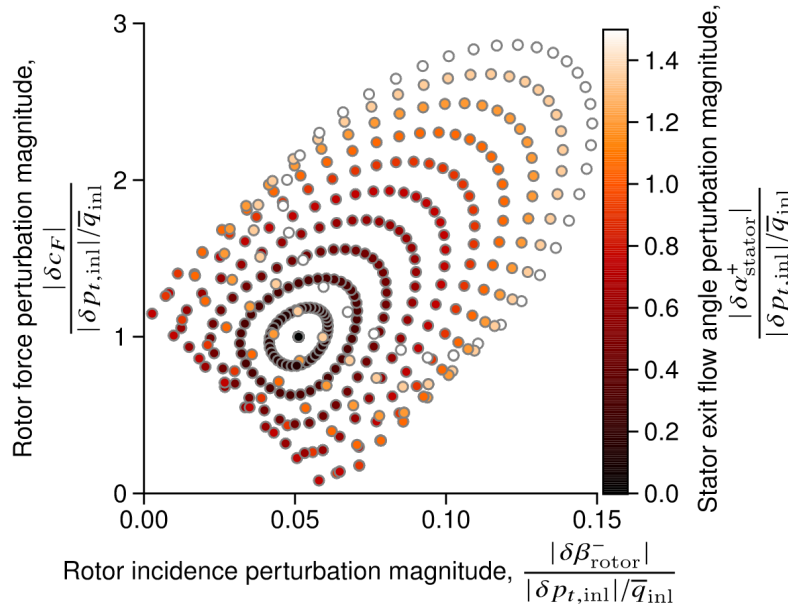


Fig. 44.—Rotor incidence angle perturbation and unsteady force perturbation versus stator exit flow angle perturbation magnitude and phase; $\phi = 0.43$, $\psi = 0.29$, $\Delta x/R = 0.41$, $M_{inl} \ll 1$.

2.2.2.3 Nonaxisymmetric FEGV Design: Conclusions

Based on the design study, we draw the following conclusions about designing for fan response to inlet distortion:

1. There are FEGV exit angle distributions that create static pressure variations that provide substantial reductions in rotor incidence variations (and thus loss perturbations) and in unsteady forces on the upstream rotor of a fan stage. There are, in fact, specific distributions in which distortion-created variations can be reduced to zero.
2. The rotor incidence angle and the rotor unsteady forces cannot be minimized (to zero) simultaneously. Treating the FEGV design as a multi-objective optimization of these two metrics, however, shows a range of Pareto optimal designs with reductions in both parameters relative to an axisymmetric design. Nonaxisymmetric FEGVs may thus have potential to improve both efficiency and aeromechanical operability of fans with inlet distortion, such as that associated with BLI, rather than having to trade between the two.
3. The stator's upstream influence diminishes exponentially with the distance between rotor and stator, and the FEGV must be well within a radius of the rotor to achieve useful modification of the conditions at the rotor (Ref. 71).
4. The estimated FEGV modifications (leading and trailing edge angles) to substantially reduce perturbations in rotor incidence angle or unsteady rotor force are comparable to those currently used to shield civil fans from the pressure field of downstream struts and pylons (Ref. 71).

2.3 Multidisciplinary Design Optimization (MDO)

The optimization of aircraft configurations has a long and storied history wherein analysis tools of varying level of fidelity have been utilized when identifying preferable sizing characteristics that optimally meet aircraft performance metrics, while meeting additional constraints. Optimizing for preferable weight and aerodynamic performance, while subject to load-path, payload, flight envelope, and

stability and control constraints is a highly complex process, but one that is generally approached from a MDO perspective. Throughout the current study, a novel framework was developed to facilitate aeroelastic optimization of wing, body, and configuration designs using high-fidelity Navier-Stokes flow solutions and finite element structural analysis for shape and topology optimization. An example case is also provided, where this framework is applied to the CHEETA concept vision vehicle.

2.3.1 Aeroelastic Design Optimization Framework Development

Modern MDO tools have enhanced the ability for aircraft designers to navigate the multidisciplinary aircraft design space with increasing levels of predictive capability. However, historical approaches to aerodynamic shape optimization with high-fidelity Navier-Stokes flow solutions and structural topology optimization strategies that leverage high-fidelity FEA methods have largely been performed independently of one another. Given the importance of aeroelastic loading for wings of increasingly high aspect ratio seen in modern transport-class aircraft fleets, resolving this interconnection between airframe structures and aerodynamic performance inclusive of aeroelastic deformation becomes important. Such a capability has been developed throughout the current CHEETA program.

Modern aircraft design is a complex and challenging endeavor, as it requires meeting diverse mission requirements while addressing the intricate interplay of multiple aircraft systems, including aerodynamics, propulsion, structure, and controls. For instance, optimizing the structural weight of an aircraft has the potential to reduce the required lift coefficient (C_L) during cruise, thereby mitigating induced drag. However, this pursuit introduces the intricacies of aeroelastic coupling, leading to increased deformations, especially in wing twist, which can cause significant wing washout. Consequently, altering the spanwise pressure gradient at the same C_L may induce shock-induced boundary layer separation, resulting in a detrimental increase in drag. The multifaceted nature of these challenges necessitates careful consideration and integration of various design aspects to achieve efficient and high-performing aircraft configurations.

In the case of a H_2 -electric aircraft design, these challenges in design are further exacerbated. As an example, the low volumetric energy density of H_2 mandates that its storage in tanks be placed within the fuselage, as opposed to storing fuels in the wing box that provide inertial relief (see Fig. 45). Additionally, for the same payload/passenger requirements, placing the tanks within the fuselage leads to an increase in surface area and fuselage mass, thus adding to the structural weight and drag penalty. Hence, to overcome these challenges to enable H_2 -electric aircraft design, an integrated design approach is needed.

In aircraft design, concurrent approaches that leverage the synergistic aeroelastic response fall under MDO. While the fundamental concepts of MDO are widely recognized, their routine application in innovative aircraft configurations is limited. MDO often relies on low to medium fidelity models, neglecting performance penalties from nonlinear physics-based effects like flow separation. The use of high-fidelity CFD is restricted due to computational costs for large-scale configurations and complex multidomain physics. Despite extensive research, there is a scarcity of open-source, high-fidelity multiphysics toolkits and stable MDO strategies, hindering progress in the field.

Throughout this program, a robust framework for large-scale aerodynamic shape and structural topology optimization has been developed, with the primary objective of facilitating the realization of the NASA CHEETA H_2 -electric aircraft concept. This optimization framework has been built upon the pillars of high-fidelity MDO, encompassing essential components that efficient gradient-based optimization, such as high-performance computing, transonic aeroelasticity, topology optimization, and coupled adjoint sensitivity evaluation. Fig. 46 represents the integrated modules constituting the MDO architecture specifically devised for this study.

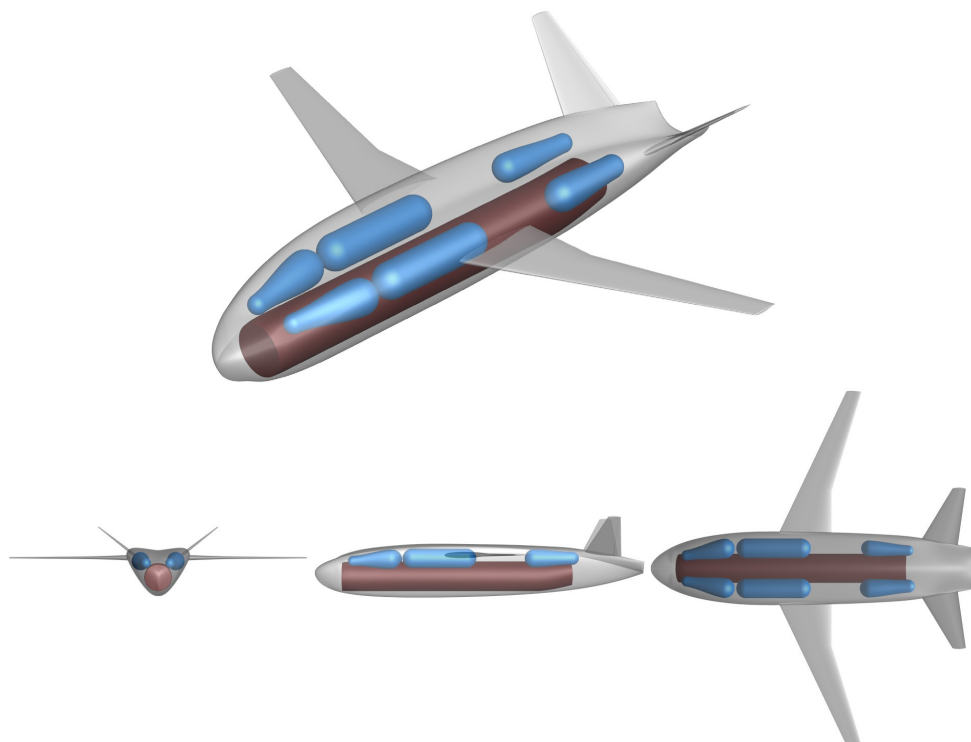


Fig. 45.—Internal component layout for CHEETA concept; LH₂ tanks depicted in blue, cabin in brown.

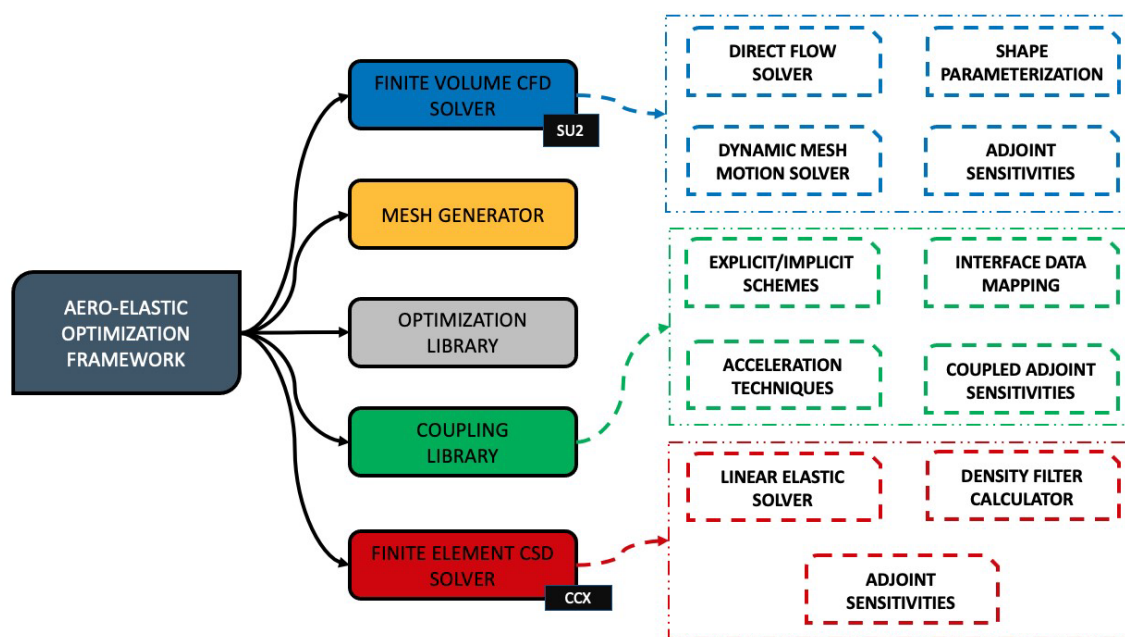


Fig. 46.—MDO and analysis architecture.

2.3.1.1 High-Fidelity Aeroelasticity Framework

One of the critical design attributes of the CHEETA concept is the high aspect ratio wing (see Fig. 47), similar to that of conventional commercial aviation airframes such as the Airbus A350 and Boeing 787.

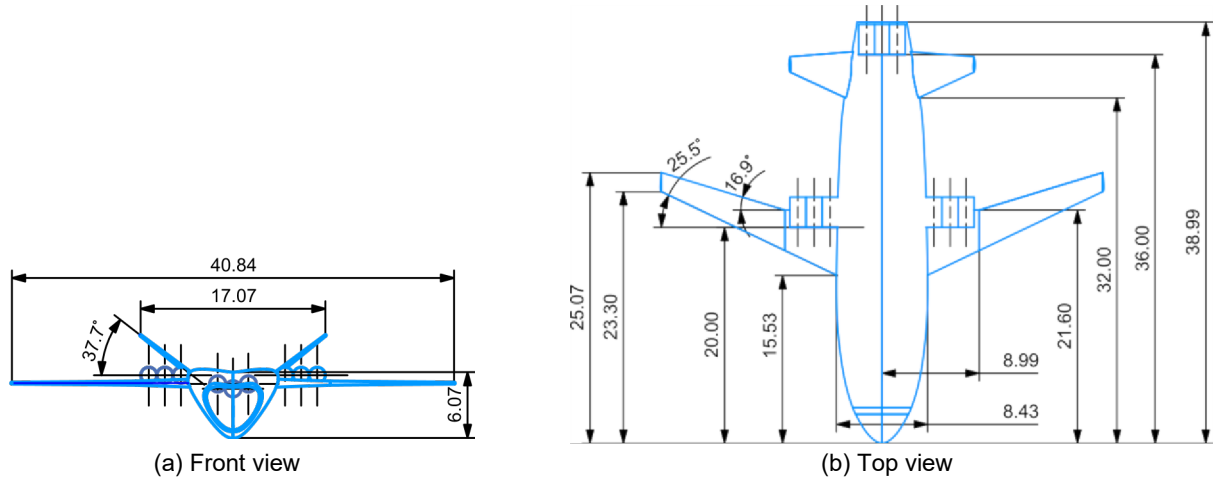


Fig. 47.—Baseline dimensions for CHEETA Mark II (MKII) configuration. (a) Front view. (b) Top view; all dimensions shown in meters.

The primary motivation behind this slender wing configuration is their capacity to reduce induced drag during cruise conditions. Research by Abbas et al. (Ref. 74) indicates that induced drag accounts for approximately 43% of the total drag experienced during flight. High aspect ratio wings have also demonstrated improved L/D performance in off-design conditions, particularly at low speeds (Ref. 75). However, the advantages associated with high aspect ratio designs come at the expense of increased weight penalties and undesired aeroelastic effects. As the wing span increases, the underlying cantilevered wing-box becomes slender, resulting in higher susceptibility to adverse added-mass effect instabilities (Ref. 76), as well as elevated stress and moment at the wing root. The enhanced compliance of high aspect ratio wings is a significant limiting factor for further increases in wing span. This is because, under the same flight conditions, a wing with a higher aspect ratio experiences larger deflections and a higher likelihood of adverse coalescence among bending, torsion, and flap modes of vibration as compared to a wing with a lower aspect ratio. The resulting system exhibits pronounced nonlinear behavior, given that the inputs or excitations to the aeroelastic system do not produce proportional responses. Nonlinearities originate from multiple subdisciplines, including aerodynamics, structural elasticity, and control design, all of which are crucial in aircraft design. Therefore, any endeavor to develop next-generation high aspect ratio wings must incorporate comprehensive considerations of nonlinear aeroelasticity.

The lack of well-established open-source aeroelastic simulation tools for large-scale aircraft aeroelasticity led to the development of a novel aeroelastic framework. In the current section, the mathematical building blocks that constitute the multiphysics simulation framework are presented and summarized. The large-scale multiphysics framework is developed for evaluating the static and dynamic aeroelastic state $\mathcal{G}$, expressed as

$$\mathcal{G} = \begin{Bmatrix} S(\mathcal{U}, \mathcal{X}, \mathcal{F}) \\ R(\mathcal{F}, \mathcal{X}) \\ M(\mathcal{X}, \mathcal{U}) \end{Bmatrix} = 0 \quad (28)$$

where S , R , and M are the elastic, fluid and fluid-mesh residual, respectively. For a more detailed explanation of the underlying numerics of each domain, the reader is referred to References 77 and 78. The three subsystems are solved using an implicit partitioned coupling approach wherein each residual is solved in sequence, subject to dynamic and kinematic boundary conditions at the aeroelastic interface Γ .

Validating and verifying numerically predicted aeroelastic behavior is critical to ensure the accuracy of the computed coupled states within an optimization process. The inclusion of nonlinear flow phenomena, particularly shock-boundary layer interactions over an elastic surface in transonic flow regimes, further complicates the numerical modeling process. Consequently, meticulous attention must be devoted to ensuring the correctness of the numerical framework. To thoroughly investigate the mathematical and numerical accuracy of the aforementioned aeroelastic framework, the High Reynolds Number Aero-Structural Dynamics (HIRENASD) wing (Refs. 79, 80, and 81) was selected as a benchmark case. This choice allows for an accurate assessment of the framework's performance and fidelity.

A complete representation of the domains, especially the elastic domain, is crucial for an aeroelastic topology optimization framework that requires a fully discretized elastic domain. Therefore, the lessons learned from this validation and verification process facilitate the successful implementation of coupled aeroelastic analysis as a forward analysis during the design cycle. Moreover, the results presented in this report surpass the static aeroelastic characterization reported in the literature, examining the effects of resulting aeroelastic states on both the fluid on the elastic surface and the elastic fields within the wing box using high-resolution quantitative and qualitative visualizations.

Fig. 48 illustrates the HIRENASD wing surface and underlying wing box. The elastic domain includes the clamping flange and nose, preserving the original wind tunnel model's structural properties. While the clamping flange is fixed, the section between the flange and wing root is elastic but not subjected to aerodynamic tractions. Fig. 48(b) displays the internal wing-box structure. The wing geometry, including its sweep and transonic flow conditions, aligns with that of civil transport aircraft wings. The well-defined wing-box layout with predetermined rib and skin thickness enables a focused study of the structure's aeroelastic behavior. Previous investigations on aeroelastic coupling and verification and validation studies have commonly used the Advisory Group for Aerospace Research and Development (AGARD) wing as a benchmark, often representing its structure as a beam or flat plate rather than a complete wing-box configuration.

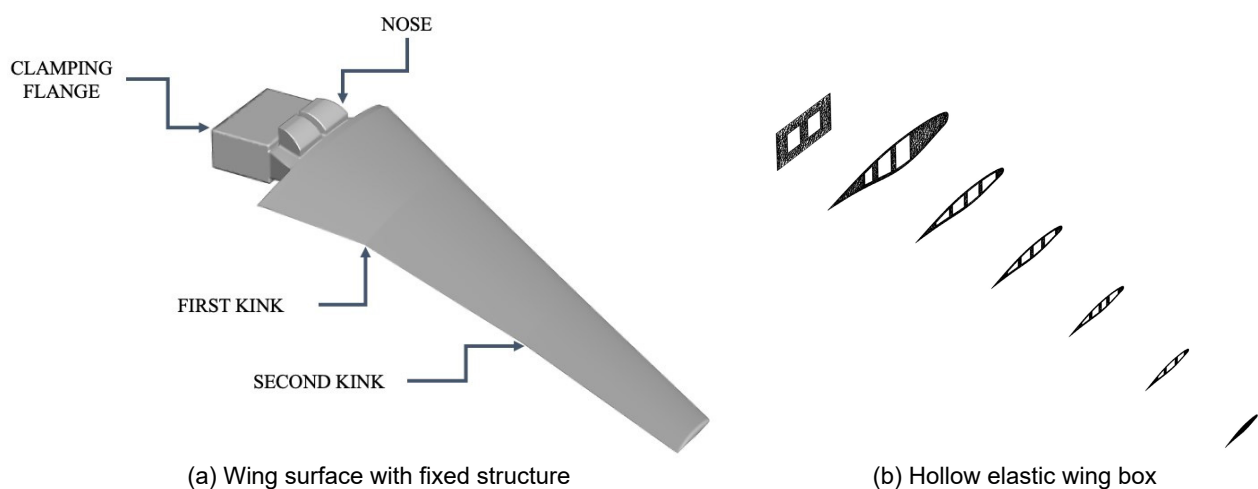


Fig. 48.—HIRENASD wing surface and wing-box layout.

In the test matrix shown in Fig. 49, an α sweep is conducted for three loading conditions (q/E) at each Mach number. This setup allows for the examination of Mach number effects on the aeroelastic state as well as the impact of increasing loading factor. To investigate the influence of Reynolds number, α sweeps are performed at $Re_c = 23.5, 35$, and 50 million, with a loading factor of $q/E = 0.48 \times 10^{-6}$. For a comprehensive discussion of the aeroelastic states at these flow conditions, the reader is referred to Reference 78.

Fig. 50 compares the predicted and measured tip displacements of the HIRENASD wing at $M_\infty = 0.80$ and a Reynolds number of 35 million, varying with α . The wing-tip displacements were obtained by Reimer et al. (Ref. 82) to validate the SOFIA model and were measured using a stereo pattern tracking (SPT) system for two loading factors.

The predicted wing-tip displacement shows good agreement with the SPT data up to $\alpha = 3.5^\circ$, but beyond this point, the predicted response gradually deviates from the measured elastic state. To ensure small displacements, all computations were restricted to $\alpha = 5^\circ$. Fig. 51 presents the aerodynamic characteristics at $M_\infty = 0.80$ and Reynolds number of 35 million, under aeroelastic equilibrium. In Fig. 51(a), an increase in wing-tip displacement is observed as the loading factor rises for a fixed α . This behavior is expected, given that an increased loading factor corresponds to higher freestream dynamic pressure, assuming constant material properties. Fig. 51(b) and Fig. 51(c) indicate a decrease in C_L and C_D with increasing loading factor.

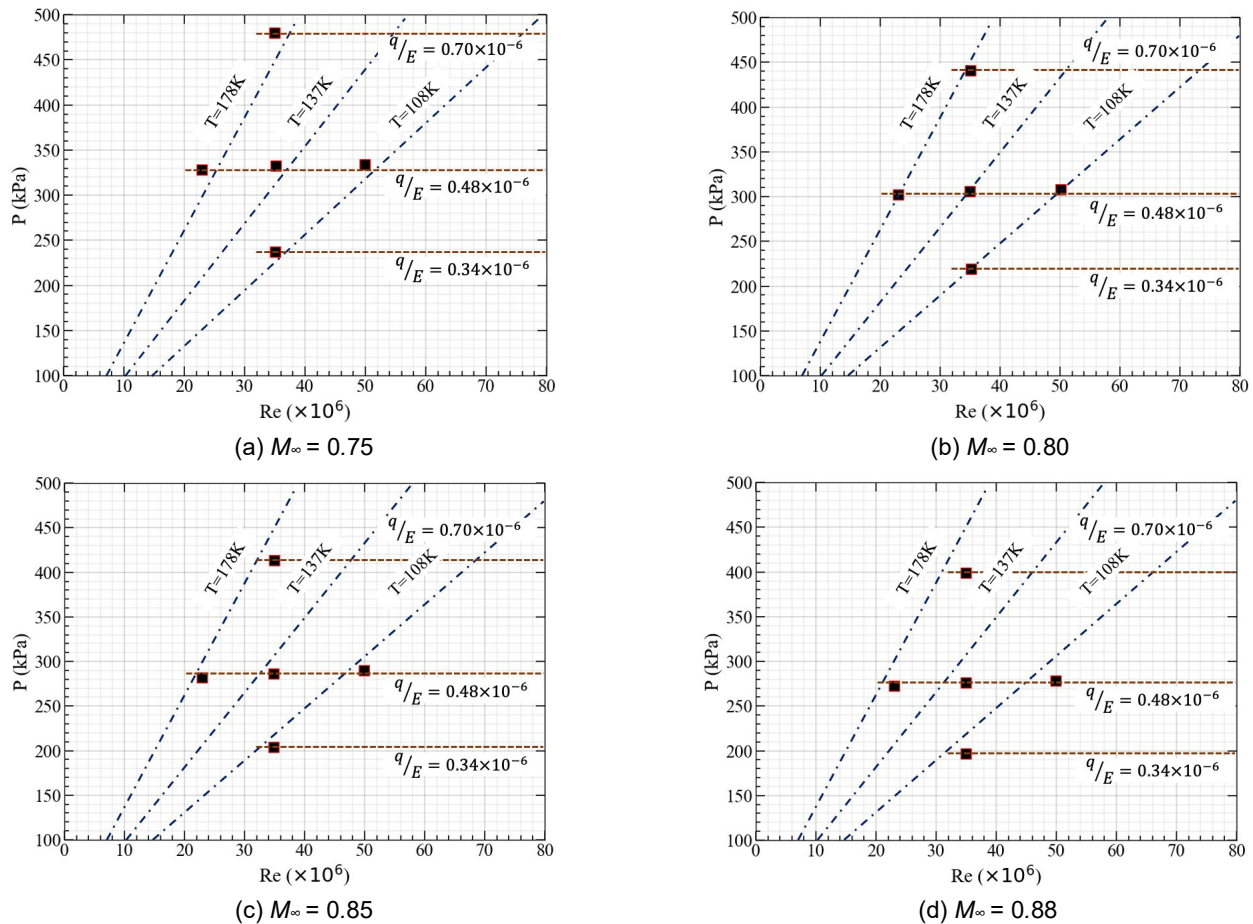


Fig. 49.—High Reynolds number flow conditions for HIRENASD aero-elastic analysis.

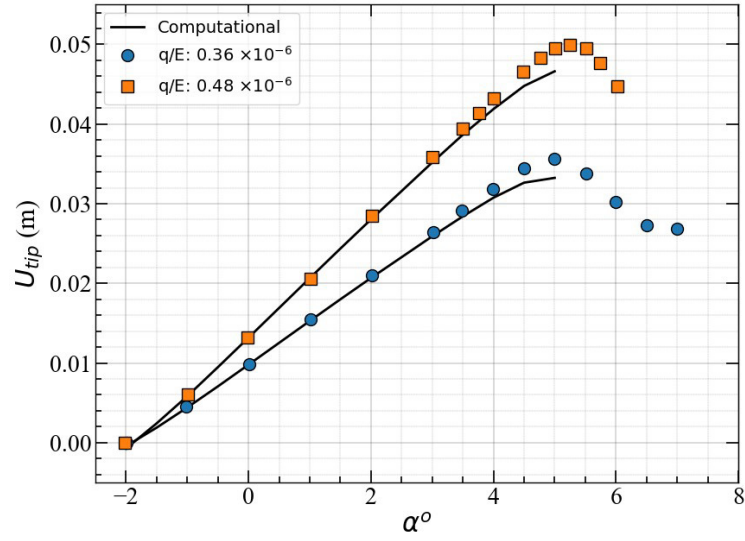


Fig. 50.—Experiment versus numerical prediction of wing-tip displacement at $M_\infty = 0.8$ and $Re_c = 35 \times 10^6$.

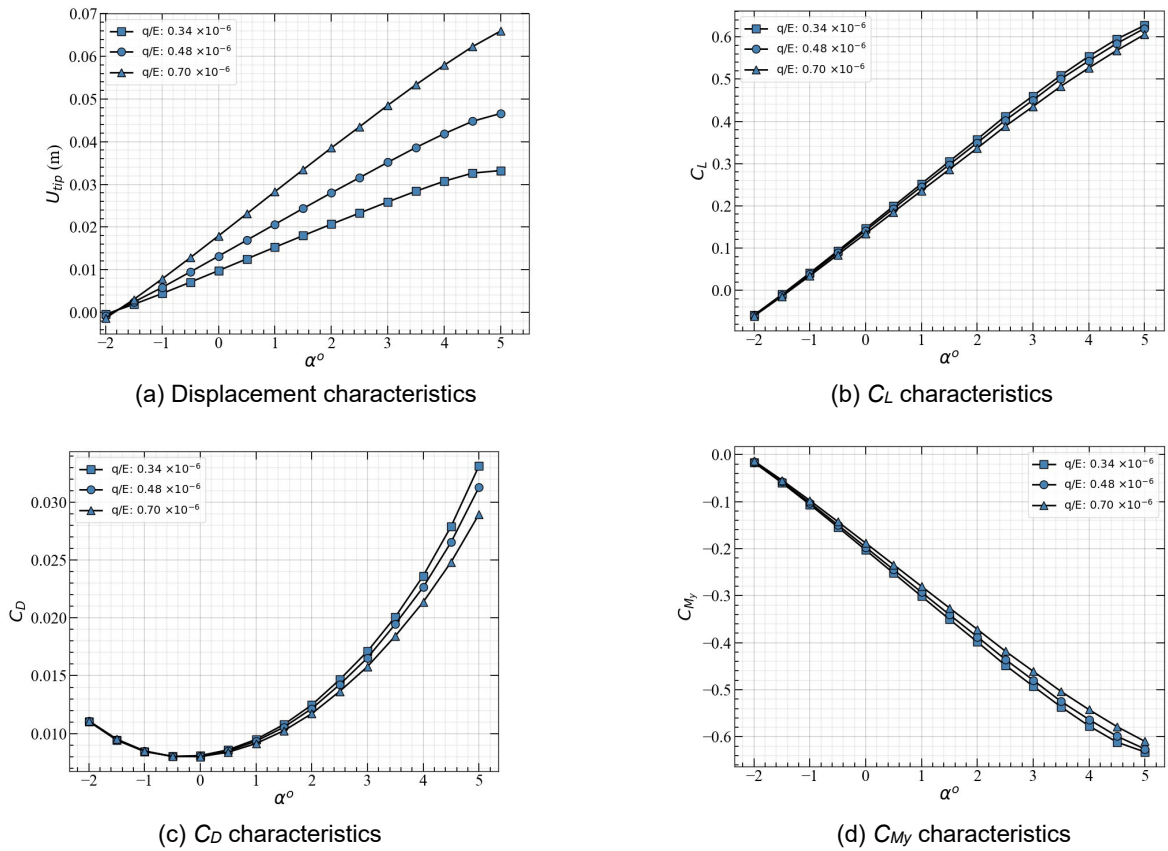


Fig. 51.—Aero-elastic performance characteristics with varying loading factor (q/E).

The observed decrease in aerodynamic performance can be attributed to the increase in washout as the loading factor increases, resulting in a higher nose-up pitching moment of the wing (see Fig. 51(d)). Furthermore, a closer examination of Fig. 51(a) and Fig. 51(b) reveals that in the absence of additional loading ($C_L = 0$), the aerodynamic twist of the wing increases with the loading factor. To gain insight into the decrement of aerodynamic performance with increasing α , the aeroelastic states at two angles of attack were studied. Fig. 52 and Fig. 53 depict the distribution of wing-skin von Mises stress along with Mach isocontours at $M_\infty = 1.1$ and $M_\infty = 1.2$ for $\alpha = 2^\circ$ and $\alpha = 4^\circ$ respectively. Although isocontours at $M_\infty = 1.0$ would ideally represent the entire supersonic bubble over the surface, the representation was limited to clearly show the corresponding surface stress distribution. At $\alpha = 2^\circ$ and $M_\infty = 0.75$, there are no shock features (Mach isocontours) present for both the rigid and deformed geometries, which is consistent with the sectional C_p distribution reported by Reimer et al. (Ref. 82). Upon increasing α to 4° , shock features emerge on the outboard section of the rigid configuration ($y \geq 0.5$ m). However, this leading-edge feature appears to diminish in the deformed configuration. Furthermore, the surface stress distribution from Fig. 52(a) and Fig. 52(b) indicates a decrement at aeroelastic equilibrium, which can be attributed to the overall decrease in additional loading.

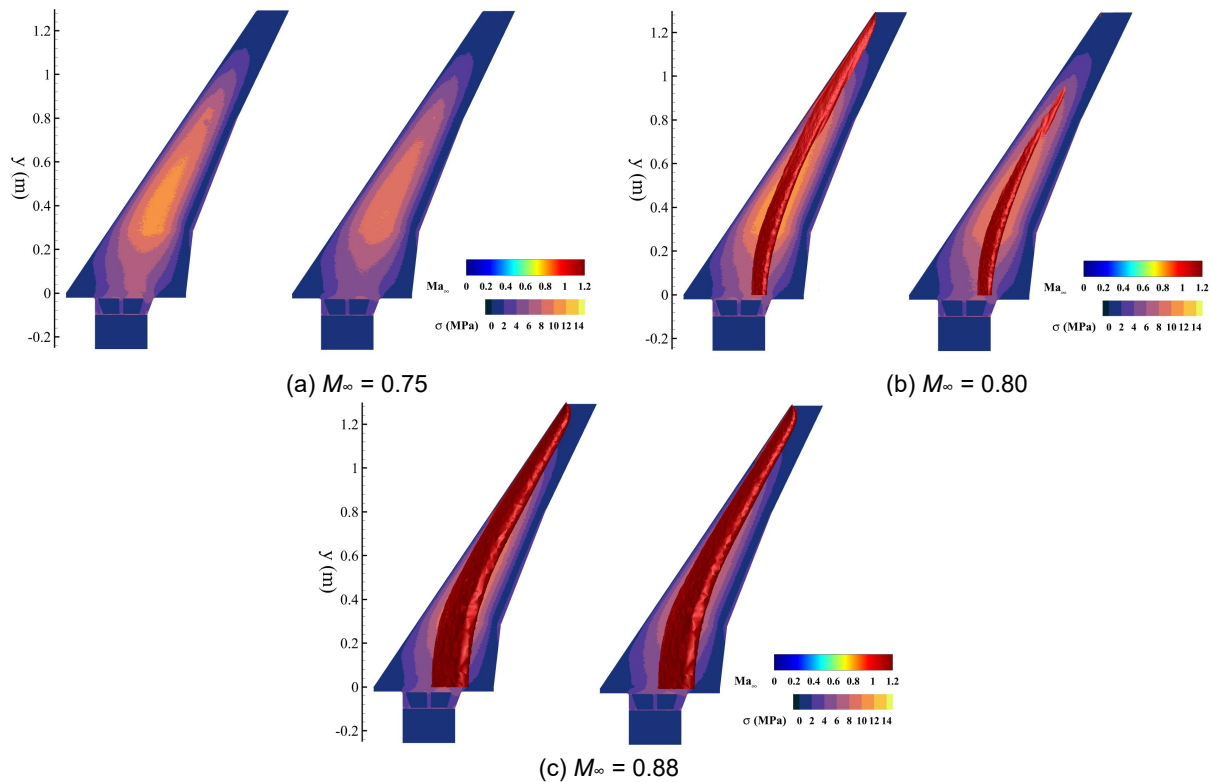


Fig. 52.—Rigid (left) versus aero-elastic (right) states with isocontours at Mach = 1.1 and $Re_c = 35 \times 10^6$, $\alpha = 2.0^\circ$ and $q/E = 0.48 \times 10^{-6}$.

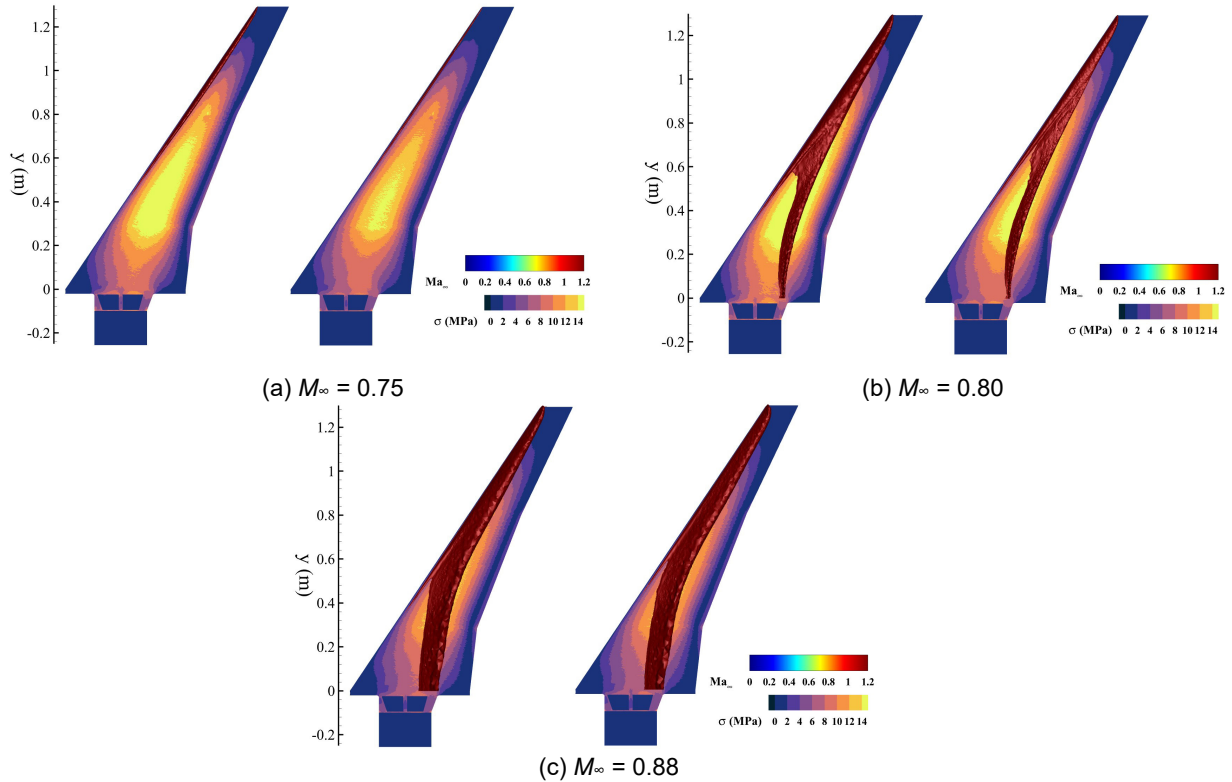
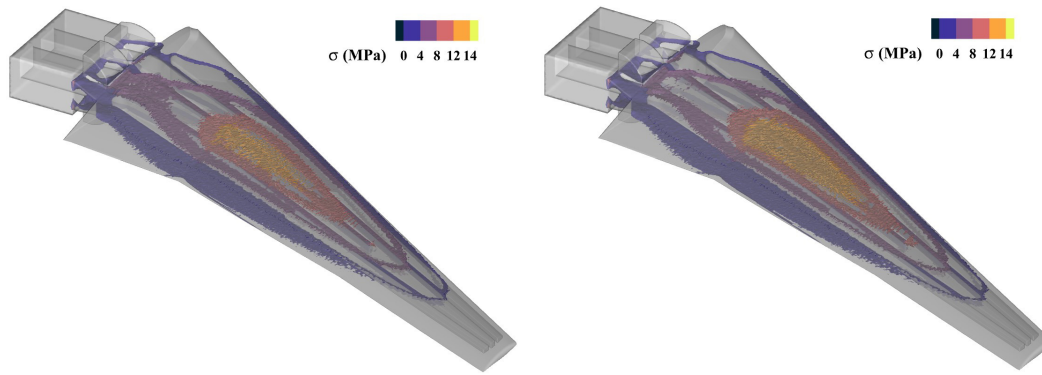


Fig. 53.—Rigid (left) versus aeroelastic (right) states with isocontours at Mach = 1.2 and $Re_c = 35 \times 10^6$, $\alpha = 4.0^\circ$ and $q/E = 0.48 \times 10^{-6}$.

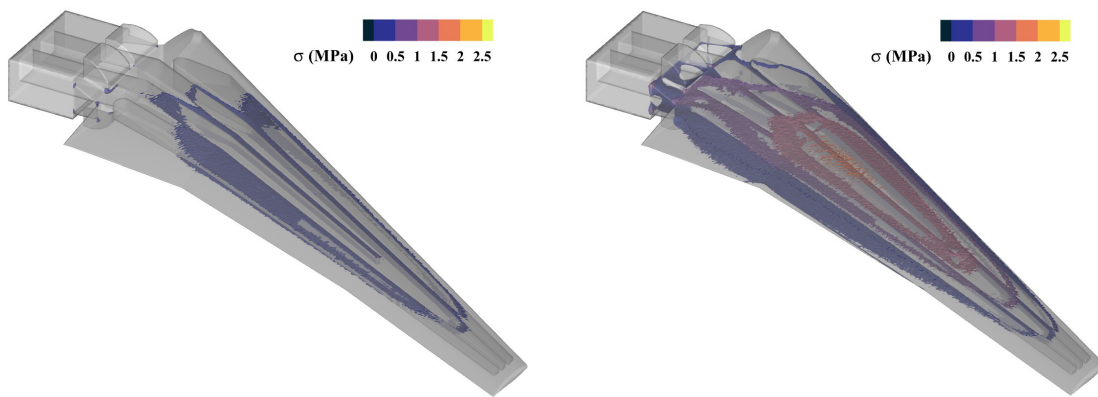
Fig. 54 depicts the internal stress states of the deformed elastic wing at $M_\infty = 0.75$ and 0.88 in (a) and (b) respectively. Interestingly, the wing skin and the front and rear spars appear to be the primary load-bearing members of the wing structure. The ribs play a role in transmitting loads from the skin to the spars in shear, but they do not seem to contribute significantly to the load transmission to the wing root at the current contour levels. As the Mach number increases, there is an observed increase in skin stress distribution at the highest contour level. However, for $M_\infty = 0.88$, the stress at lower contour levels exhibits a different path compared to $M_\infty = 0.75$. When comparing the stress distributions in the root region, a higher concentration is observed for $M_\infty = 0.75$ and 0.88 .

Fig. 55 provides insight into the internal stress state under basic loading conditions ($C_L = 0$). The wing exhibits aerodynamic tractions resulting from aerodynamic twist. At $M_\infty = 0.75$, the spars are identified as the primary load-carrying members in the presence of aerodynamic twist. However, at $M_\infty = 0.88$, the skin also contributes to the load-carrying and transmitting characteristics of the wing box. Notably, distinct stress isocontours are observed with varying loading factors. For instance, at a low loading factor of $q/E = 0.34 \times 10^{-6}$, the intermediate isocontours appear coalesced near the root. As the loading factor increases, the intermediate isocontours become more widely distributed. At the highest contour levels, the stress distribution over the skin follows a trend consistent with previous observations: a decrease in loading due to increasing washout. Conversely, lower contour levels indicate only marginal differences in the spanwise distribution of stress toward the fixed wing root.

The comprehensive test and evaluation conducted on the aeroelastic solver, using a benchmark case aligned with the original design problem, showcases its robustness and high level of accuracy. These favorable characteristics make the developed aeroelasticity solver well-suited for seamless integration into the large-scale aerostructural optimization framework.



(a) $M_\infty = 0.75$ (b) $M_\infty = 0.88$
 Fig. 54.—von Mises stress contours at $Re_c = 35 \times 10^6$, $\alpha = 4.0^\circ$, and $q/E = 0.48 \times 10^{-6}$.



(a) $M_\infty = 0.75$ (b) $M_\infty = 0.88$
 Fig. 55.—von Mises stress contours at $Re_c = 35 \times 10^6$, $C_L = 0.0$, and $q/E = 0.48 \times 10^{-6}$.

2.3.1.2 Structural Topology Optimization

Topology optimization has been extensively explored in the last two decades. However, its application to aerostructural design has remained limited. Previous studies by Chin et al. (Ref. 83), Krog et al. (Ref. 84), and Zhu et al. (Ref. 85) have primarily focused on isolated components of aircraft structures, often with fixed external tractions. However, with the advancements in computing paradigms, such as large-scale distributed and shared memory architectures, the aerostructural design community has shifted its attention toward aeroelastic topology optimization of larger aircraft structures. Prominent works by James et al. (Ref. 86), Dunning et al. (Ref. 87), Maute et al. (Ref. 88), and Aage et al. (Ref. 89) exemplify this emerging trend.

To achieve lightweight structures for the CHEETA airframe, an aeroelastic topology optimization approach is employed using the NASA Common Research Model (CRM) as a canonical test case (Ref. 90). The computational domain consists of the fuselage and the wing, with the wing being the elastic domain subjected to aerodynamic tractions at specified flow conditions outlined in Table 7. The normalization of the full-scale CRM configuration is based on the MAC (mean aerodynamic chord) along all three dimensions, excluding the horizontal and vertical stabilizers. The cantilevered wing is responsible for supporting the weight of the wing structure, as well as the fuselage, fuel, and prescribed payload. To avoid large deformations during the design process, the wing structure is relieved from the added weight. Detailed information regarding the structural characteristics can be found in Table 8, while Table 9 provides geometric specifications for the CRM configuration.

TABLE 7.—SCALED NASA CRM
FLOW CHARACTERISTICS

Parameter	Value
M_∞	0.85
Re_c	5.0×10^6
ρ_f , kg/m ³	0.27

TABLE 8.—SCALED NASA CRM
ELASTIC CHARACTERISTICS

Parameter	Value
E , GPa	73.1
ν	0.3
ρ_s , kg/m ³	2,780

TABLE 9.—SCALED NASA CRM
GEOMETRIC CHARACTERISTICS

Parameter	Value
Aspect ratio	9.0
Span, m	8.38
Yehudi chord, m	1.035
MAC, m	1
Tip chord, m	0.390
Taper ratio	0.275

The conventional approach to wing structure design involves several iterative steps. Initially, the geometry of the outer mold line (OML) is defined based on desired aircraft performance characteristics. Then, a fixed amount of structural material is allocated along the wing span to meet the required structural strength during flight. Subsequently, the preliminary wing structure undergoes evaluation for aeroelastic effects, and necessary modifications are made as needed. These modifications are further assessed for static and dynamic aeroelastic performance metrics and their compliance with stiffness and strength requirements. This iterative tailoring process is repeated until a wing structure that meets the design objectives is achieved.

In the current study, this traditional design approach is bypassed by directly solving the compliance minimization problem

$$\begin{aligned}
 &\min_{\rho_e, \alpha} C_{elastic} \\
 &s. t. L = \rho_s \times (V_F \times V) \times g' \\
 &V_F = V_F^* \leq \epsilon_1 \\
 &C_L = C_L^* \leq \epsilon_2 \\
 &[\mathbf{K}(\rho_e)]\{\mathbf{U}\} = F(\rho_e) + F_g(\rho_e) \\
 &0 < \rho_{e_{min}} \leq \rho_e \leq 1
 \end{aligned} \tag{29}$$

The problem formulation aims to determine the optimal values of the design variables, which include the angle of attack α and the structural density variables ρ_e . Several constraints are imposed to ensure the feasibility and performance of the design. The first constraint enforces the condition that the lift (L)

should equal the weight (W) during cruise. The weight of the structure is computed based on the structural density, volume fraction (representing the amount of material available to the optimizer), the volume enclosed by the wing OML, and a modified acceleration due to gravity (g') that accounts for the weight of the fuselage, payload, and fuel. The second constraint restricts the amount of material available to the optimizer, setting a limit on its distribution to minimize compliance. In the presented results, the volume fraction varies from 27% to 30%, allowing for different amounts of material allocation. The third constraint ensures that the design achieves a specific lift coefficient (C_L) during cruise conditions. A design C_L value of 0.5 is prescribed and enforced in the forward aeroelastic analysis by updating the angle of attack (α).

Fig. 56 represents the optimal material topologies obtained for the wing structure at two different volume fractions: 30% and 27%. The allocation of material by the optimizer reveals interesting patterns and load-bearing characteristics of the wing. In the case of V_F : 0.30, the optimizer assigns a higher concentration of material to the mid-span region, effectively making it the load-bearing section of the wing box. Additionally, a substantial amount of material is allocated to the root section to provide necessary support at the fixed end. Notably, material is also assigned to the trailing edge region, indicating a load transfer in the spanwise direction, resembling a trailing edge spar. Moving toward the wing tip, the optimizer distributes material predominantly in the trailing edge region, incorporating vertical truss-like members for additional support. This distribution pattern suggests that the center of pressure is positioned aft along the chordwise direction in the outboard section. In contrast, for the V_F : 0.27 case depicted in Fig. 56(b), the effect of reduced material availability is evident. The root and mid-span regions receive a higher concentration of bulk material allocation. However, due to the limited material resources, the vertical truss-like members observed in the outboard region are absent from the material distribution. Consequently, with reduced volume fraction, the wing skin assumes a greater load in terms of bending and torsion compared to the V_F : 0.30 case.

Fig. 57 provides a representation of the spanwise aeroelastic dihedral for both the baseline (left half) and tailored (right half) configurations. The bend-twist coupling effect, a common aeroelastic characteristic of swept wings, is clearly observed in the tailored configurations. Notably, the spanwise twist distribution, influenced by the variation in volume fraction (V_F), is depicted through the Mach isocontours. In the case of V_F : 0.30, the corresponding spanwise twist results in outboard regions where $M_\infty \geq 1.1$. This twist distribution pattern leads to a specific Mach number distribution across the span. Conversely, for the V_F : 0.27 case, the Mach isocontours appear more compact toward the inboard region, indicating a different distribution of spanwise twist. This change in twist distribution is attributed to the increased washout observed in the V_F : 0.27 configuration. As a consequence, the decrease in lift in the outboard region compared to the V_F : 0.30 case results in a lower wing-tip displacement. Further insights into the changes in spanwise lift distribution for the two tailored configurations are presented in Fig. 58. This figure allows for a clear comparison of the lift distribution patterns, highlighting the impact of the tailored configuration on the aerodynamic performance along the span.

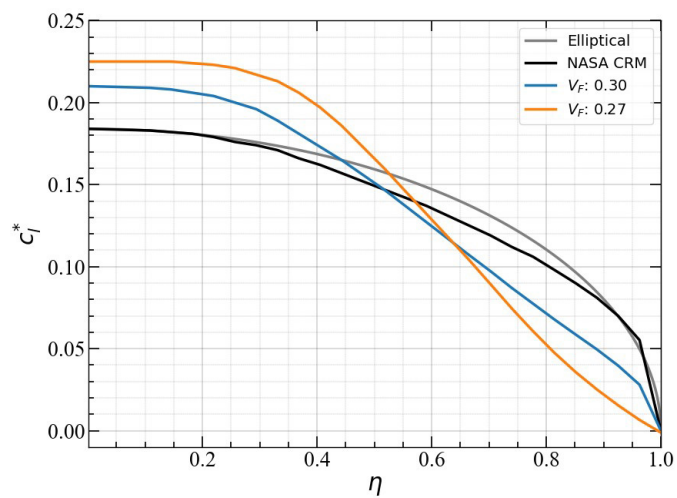
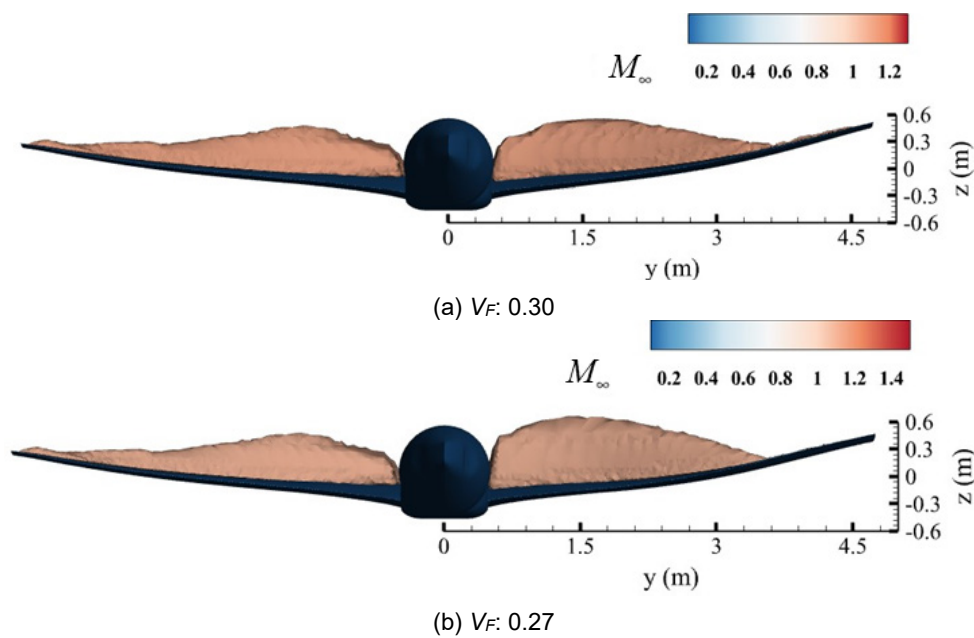
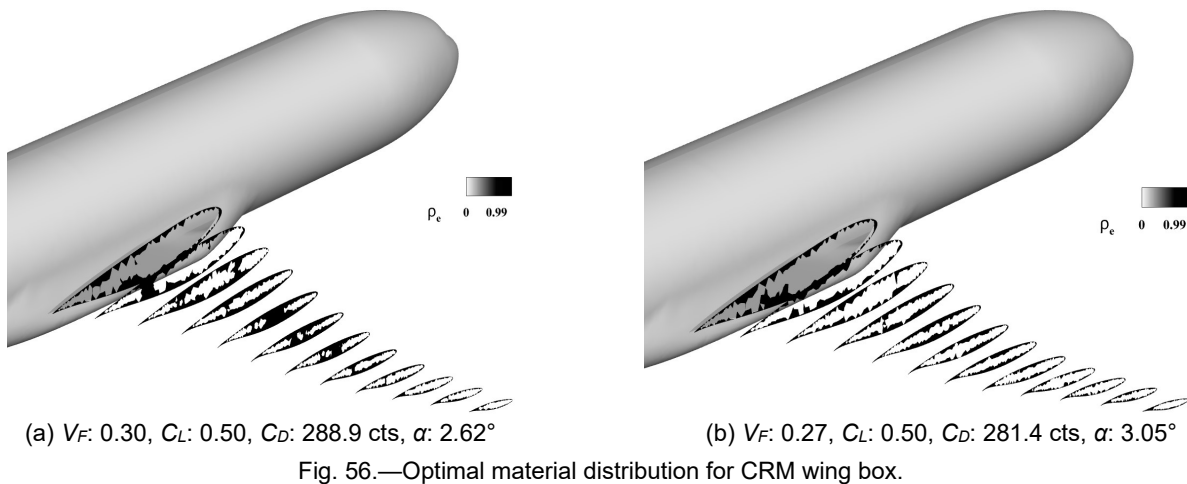


Fig. 59 presents the aerodynamic performance characteristics of the baseline, $V_F: 0.30$, and $V_F: 0.27$ configurations, computed at aeroelastic equilibrium. It is important to note that the aerodynamic center and the moment arm length e_1 are influenced by the angle of attack (α), leading to changes in the accumulated torque and subsequent spanwise twist distribution. Additionally, variations in the sectional Φ result in a shift in the center of pressure, affecting the spanwise camber. This coupled system response manifests as a decrease in lift coefficient (C_L), an increase in drag coefficient (C_D), and an increase in nose-up pitching moment.

Fig. 59(b) reveals nearly identical C_D characteristics for the two tailored configurations. Interestingly, both tailored configurations exhibit similar, albeit distinct, behaviors for C_L and C_D in the nonlinear regime, and they even converge within the range of $7^\circ \leq \alpha \leq 9^\circ$. The higher C_D characteristics observed in the tailored configurations are to be expected, as the problem formulation in Eq. (29) only includes structural and flow design variables, lacking shape design variables that could be manipulated by the optimizer to control the shape and mitigate drag penalties.

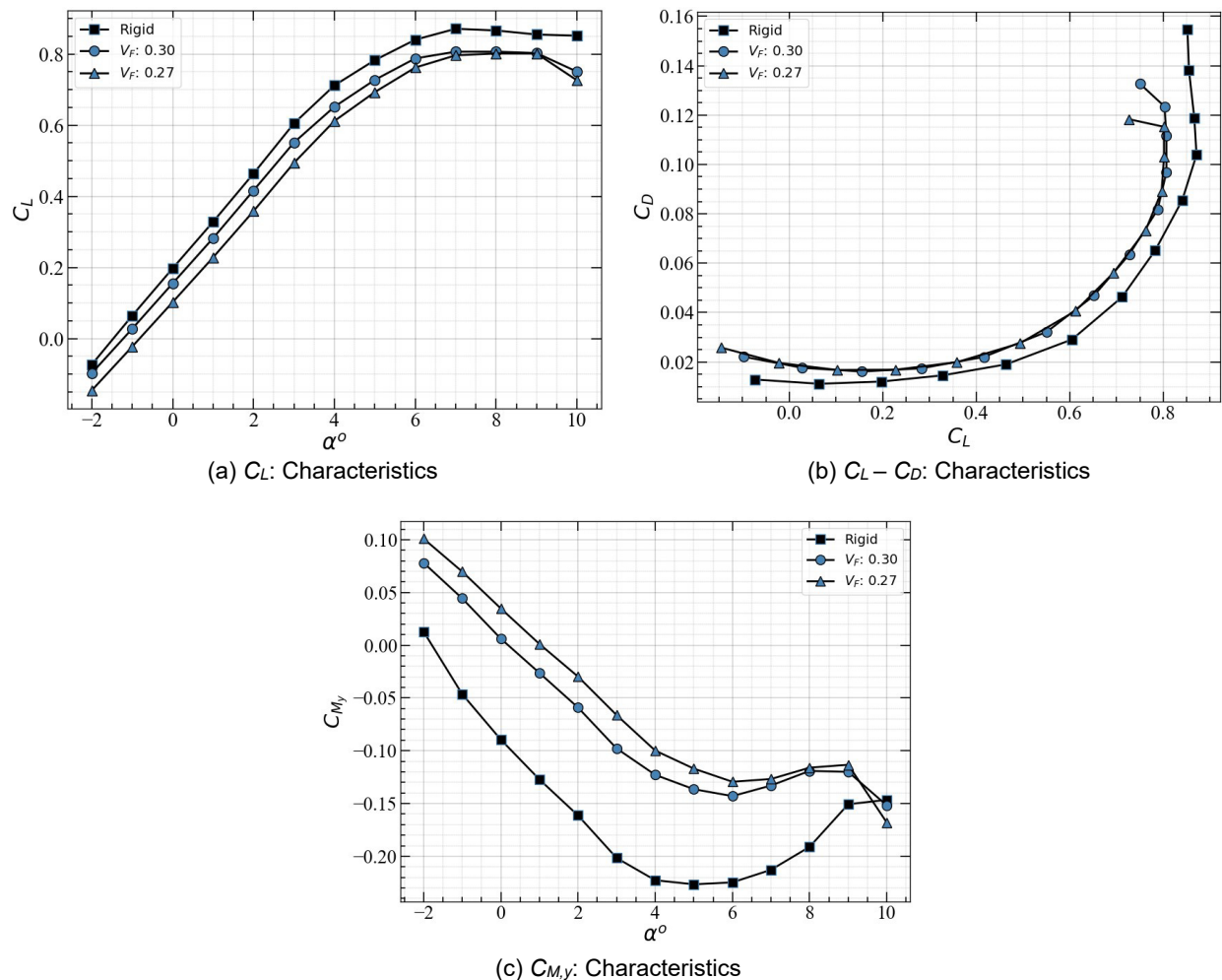


Fig. 59.—Aeroelastic performance characteristics.

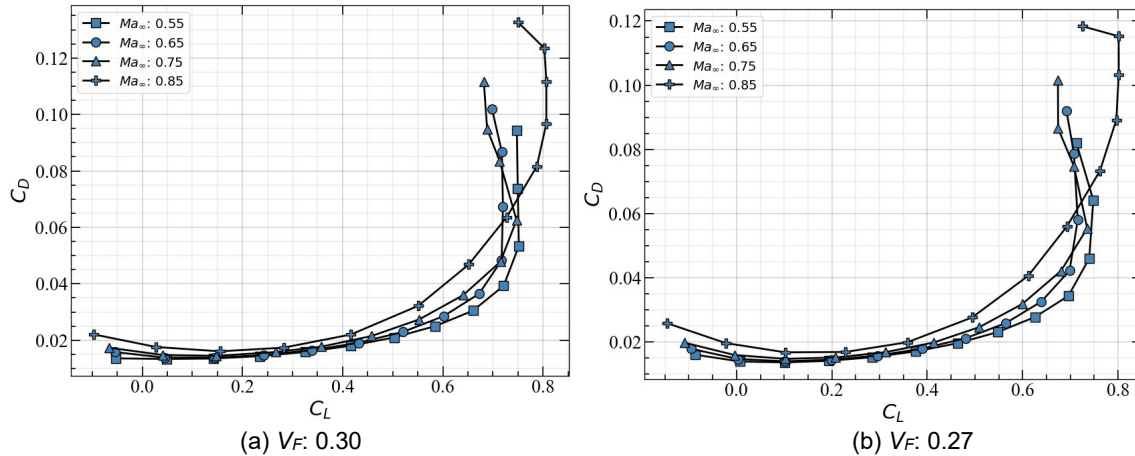


Fig. 60.—Aeroelastic performance characteristics of off-design Mach numbers.

Fig. 60 provides insights into the aerodynamic performance characteristics at off-design Mach numbers. Understanding the behavior of a wing configuration in off-design conditions is valuable for selecting design points in multipoint design optimization problems (not discussed in this context). The observed $C_L - C_D$ characteristics exemplify the aerodynamic performance of an elastic wing at aeroelastic equilibrium. As the Mach number (M_∞) increases, the dynamic pressure also increases, and the performance trends depicted here are reminiscent of those observed in the HIRENASD wing when subjected to increasing loading factors (q/E). At higher angles of attack, the two wings exhibit distinctive performance attributes.

2.3.1.3 Topology Optimization Using Shell-Based Reduced Design Space

The wing structural design obtained through topology optimization differs significantly from a conventional wing-box design comprising ribs and spars. Instead of a repetitive arrangement of thin plates in orthogonal arrays, topology optimization produces a complex material distribution pattern that poses challenges for manufacturing. To overcome these challenges, we investigated an alternative formulation where the solid domain of the wing box is populated with numerous ribs and spars. Through topology optimization, we then determined the optimal material distribution within each thin-walled plate compared with full 3D topology optimization of the solid domain defined by the wing OML. This comparison allowed us to evaluate the effectiveness of the approach, highlighting the effect of the reduction in design space. Further details about this study can be found in Reference 91.

The compliance minimization problem outlined in Reference 75 is employed to address two topology optimization scenarios. In the first scenario, the baseline design domain comprises 43 equally spaced ribs parallel to the root chord, supplemented by 15 spars spanning between the leading and trailing edges. The wing is modeled using shell finite elements of equal thickness for simplicity. To facilitate the optimizer's ability to determine the optimal number of components, a considerable number of ribs and spars are included. This allows the optimizer to completely eliminate some of them to achieve the desired optimal component count. In contrast to the first scenario, the second scenario utilizes tetrahedral finite elements to discretize the entire solid domain within the wing's OML. The edge length of these tetrahedral elements is approximately equal to the shell finite element employed in the first case. To maintain consistency in comparing the results obtained from shell finite elements and 3D tetrahedral elements, the 2D finite element types are extended to three dimensions. This extension ensures consistency in the analysis conducted for both cases. Consequently, in the second scenario, the solid domain has the ability to transform into a wing-box design with thin, plate-like components, just like in the first scenario. The design domains are shown in Fig. 61. The target volume fraction corresponds to 80% of the baseline shell model with thickness of 3 mm. The skin is included in the design space.

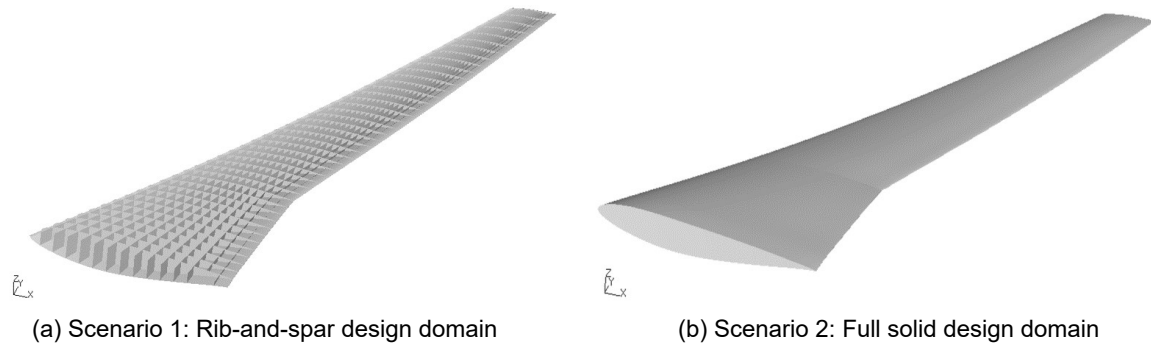


Fig. 61.—Design domains for structural design of NASA CRM wing.

To quantitatively compare the stiffness of the two designs, the compliance ratio θ is introduced, representing the ratio between the optimal compliance of the shell structure and the solid structure of equal weight. Fig. 62 demonstrates that as the shell thickness in the first scenario approaches the maximum edge length of the solid tetrahedra in the second scenario (10 mm), the shell wing-box achieves a stiffness comparable to the solid domain. This suggests that, for designs with equal weight and finite element edge lengths, a wing-box design obtained through reduced space topology optimization can achieve similar stiffness to that attained through full domain topology optimization.

Fig. 63 showcases the optimal material distribution pattern for both the shell and solid models. In both cases, a trapezoidal wing-box-like structure can be observed, indicating a significant similarity in the material distribution pattern. The solid model, despite having the ability to explore the complete design space, exhibits distinct patterns of ribs and longitudinal spar-like structures that function as stiffener to the skin in similar locations where the shell design retains its ribs and spars. This observation suggests that the conventional wing-box layout, when properly designed, can be an effective approach to achieve an optimal design configuration that is comparable to what can be obtained through density-based topology optimization. The similarity in material distribution patterns and overall structural features between the conventional wing-box layout and the topology optimization approach suggests that the former can serve as a viable alternative for wing topology optimization, offering the advantages of reduced computational effort. By combining the existing knowledge and manufacturing techniques associated with wing-box structures with the weight-saving opportunities provided by topology optimization, engineers can create efficient and structurally sound wing designs. This approach allows for the optimization of wing configurations while avoiding the complexities and challenges typically associated with intricate designs obtained via full-scale topology optimization.

Fig. 64 provides insight into the material distribution patterns of the optimal solid design and their comparison with the shell-based design. The section-wise material distribution extracted from the solid design at the same locations as the ribs in the shell-based design reveals a distinctive pattern of multicellular ribs, indicating the resemblance between the conventional wing-box layout and the optimal design. To compensate for the fixed shell thickness in the shell model, thicker truss-like vertical members are retained, whereas the solid model incorporates multicellular rib and spar-like arrangements. However, the intricate local design features of the solid model pose challenges for manufacturing and assembly due to their complexity. In the shell model, the material alignment along thin plates simplifies the manufacturing process, despite the complexities introduced by topology optimization. The shell model's design, consisting of thin plates, can be easily manufactured and assembled, whereas the solid model's intricate features may require more specialized manufacturing techniques.

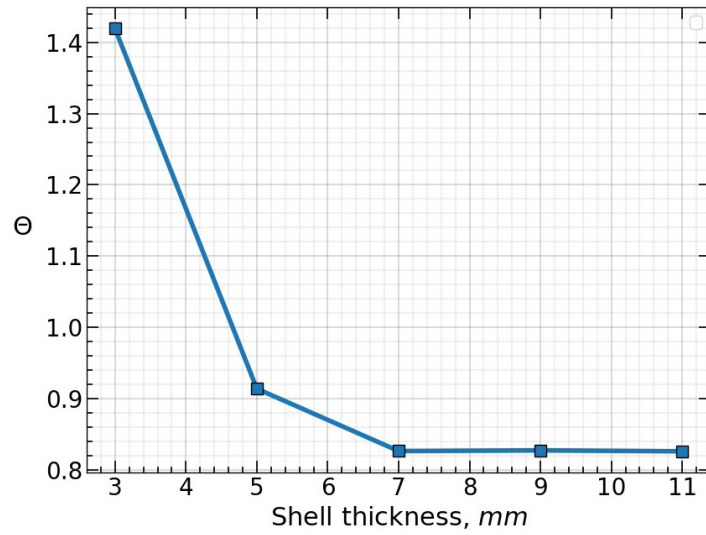


Fig. 62.—By including skin in the design space and ensuring appropriate thickness, the compliance ratio of the wing-box design can be comparable to that of a 3D-topology-based solid design.

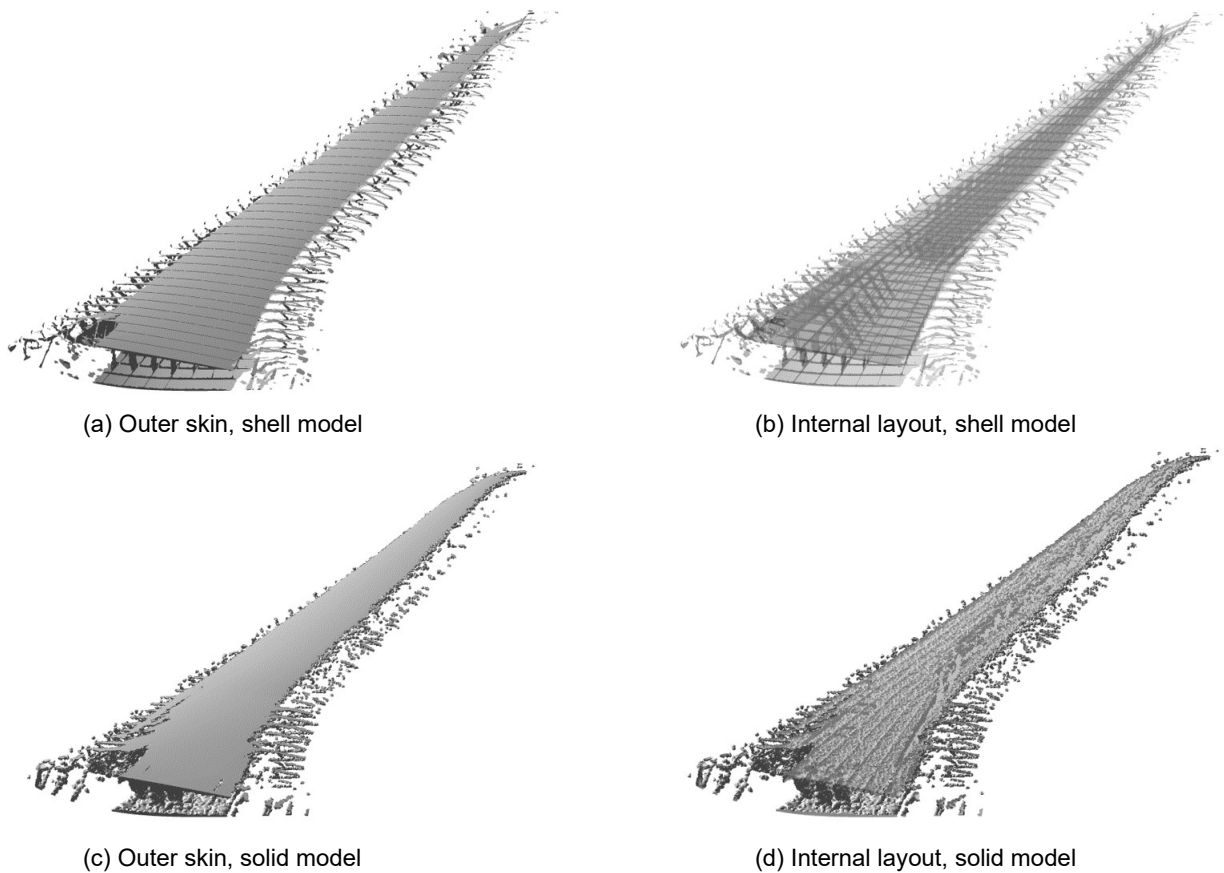


Fig. 63.—Optimal wing structure for minimum compliance obtained with shell-based rib-and-spar and full solid topology optimization.

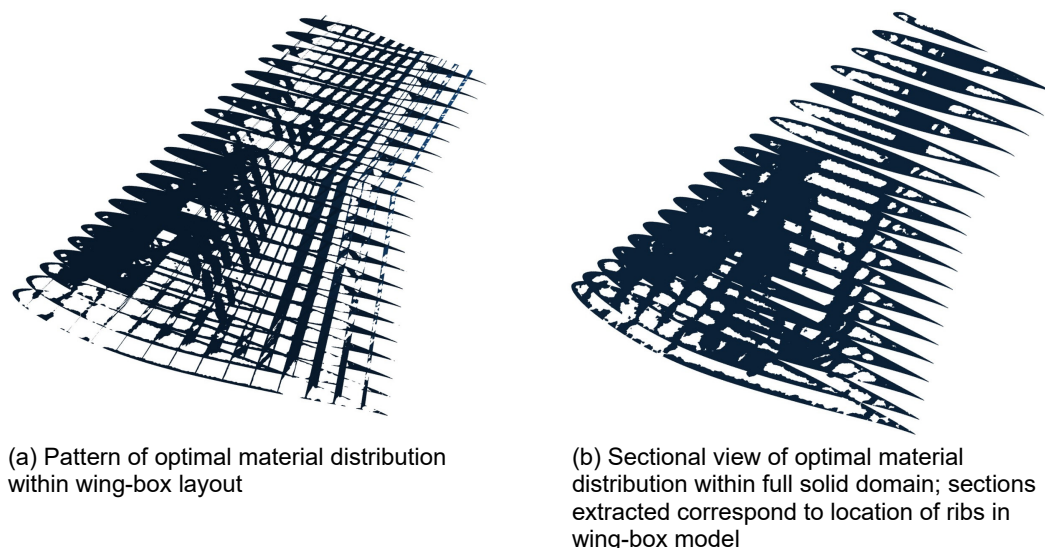


Fig. 64.—Similar clustering pattern between (a) reduced design space search and (b) full design space search. Only inboard portion of wing is shown here.

2.3.2 Configuration Design Optimization

Based on the development of the aeroelastic high-fidelity MDO strategy developed in Section 2.3.1, a preliminary optimization of the CHEETA vision vehicle was performed. This optimization study required multiple iterations of the seed aircraft geometry, grid generation, and optimization parameters.

Utilizing LH₂ as a fuel source for commercial aviation offers compelling advantages, notably its high specific energy and the potential for significantly reduced greenhouse gas emissions. However, the adoption of LH₂ is not without challenges, which include the need for cryogenic storage as well as LH₂'s low volumetric energy density, associated safety risks, specialized infrastructure requirements, and elevated costs in comparison to traditional aviation fuels. Overcoming these obstacles is crucial for establishing LH₂ as a viable fuel option for commercial aviation and necessitates a comprehensive redesign of aircraft, starting at the conceptual level.

In this context, the present study introduces the CHEETA aircraft concept—a narrow-body, H₂-electric aircraft—designed with special attention to cryogenic storage, safety considerations, and associated impacts on overall aircraft performance. This report delves into the design philosophy underpinning the CHEETA concept, tracing its evolution into a baseline configuration suitable for high-fidelity analysis. Further discussion is also provided on the flight performance characteristics of this baseline configuration, employing high-fidelity Reynolds-averaged Navier-Stokes (RANS) based CFD. A preliminary single-point optimized configuration is provided, subject to operational and design constraints.

2.3.2.1 Design Conceptualization and Baseline Configuration

To establish the design criteria for the CHEETA aircraft, the principal objective was to achieve zero emissions of both CO₂ and NO_x during flight operations. This goal was set with the intention of matching the payload, range, cruise altitude, and speed of existing single-aisle aircraft. While it is acknowledged that the production of LH₂ can itself result in emissions, this aspect was considered beyond the scope of the aircraft design project and, as such, was not incorporated into the set of requirements.

An assumed entry-into-service date of 2050 was factored into the design, which allows for the integration of advanced technologies like SC powertrain systems. Given the cryogenic nature of the

onboard LH₂, this enables the employment of cutting-edge drive electronics and motors, along with high-temperature proton-exchange membrane fuel cells (HT-PEMFCs). These fuel cells facilitate a noncombustive conversion of H₂ into electricity, emitting only water vapor as a byproduct. Although contrail formation can have notable climate forcing effects, such considerations were not included in the design parameters for this project (Ref. 1).

Specifications for the CHEETA aircraft, drawn from the Boeing 737-800, are detailed in Table 10 (Ref. 92).

Building on these requirements, an initial concept was formulated, as depicted in Fig. 65. The aircraft configuration incorporates a cabin with dimensions identical to those of the Boeing 737-800, situated at the lower section of the design. This design choice allows for the utilization of the same cabin layout, thereby providing ample space for the requisite number of passengers and cargo. To accommodate the large fuel tanks required for storing the voluminous LH₂, a substantial lifting body section was added above the cabin.

To exploit the expansive surface area of this upper section, a bank of three propulsors was strategically positioned to enable BLI along the entire length of the fuselage. Additionally, two other sets of three propulsors were mounted over each wing, further leveraging the potential benefits of BLI. It should be noted that the anticipated gains from BLI have not been factored into the sizing or configurations presented in this initial concept.

TABLE 10.—PERFORMANCE REQUIREMENTS OF CHEETA
CONCEPT, BASED ON SELECT BOEING 737-800 MISSION (REF. 92)

Parameter	Value
Passengers (single-class)	180 pax
Cabin	Single-aisle
Span (at airport gate)	Within Group III (79 to 118 ft)
Takeoff field length (standard day).....	8,200 ft
Range.....	2,935 NM at end of life (EOL)
Cruise Mach.....	0.78
Cruise altitude.....	37,000 ft
Reserves.....	200 NM divert, 30 min loiter, 5% remaining fuel after landing
Regulation.....	FAR 25 compliant
CO ₂ emissions	0.0 lb/pmi (0.2 lb/pmi for B737-800)
NO _x emissions	0.0 lb/LTO-cycle (27 lb/LTO-cycle for B737-800)

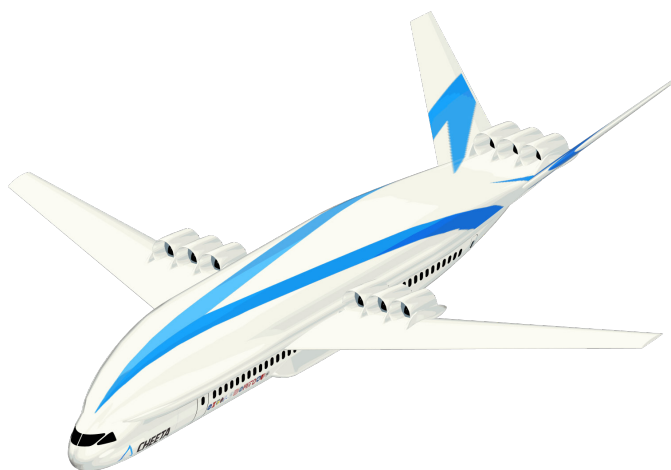


Fig. 65.—Isometric view of CHEETA aircraft concept.

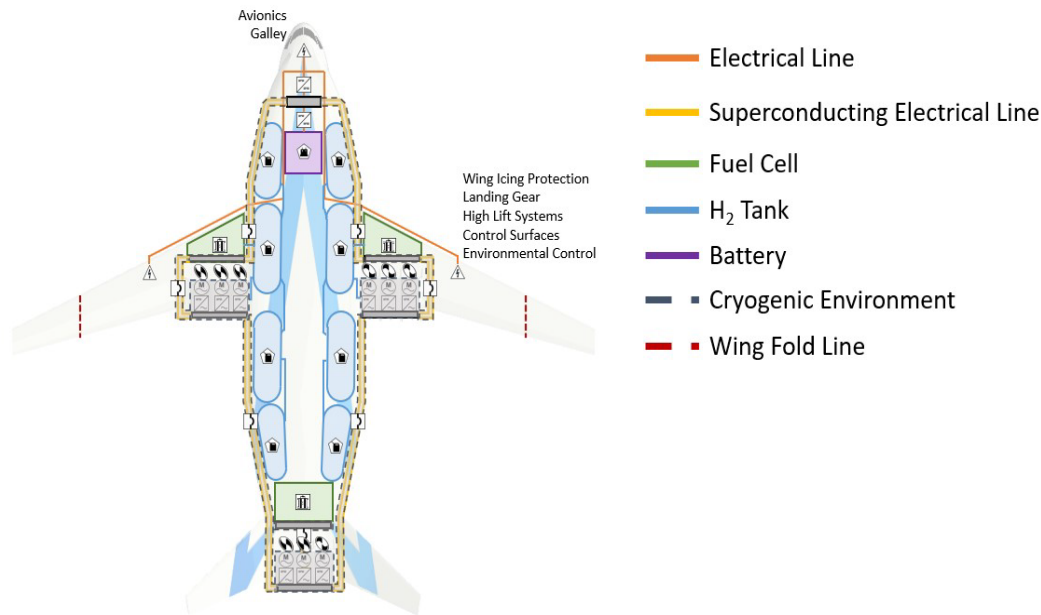


Fig. 66.—System overview of CHEETA aircraft concept.

The internal systems architecture of the aircraft is illustrated in Fig. 66. The aircraft's power is supplied by fuel cells, represented in green, which channel electricity through SC electrical lines, shown in yellow, to the propulsors, depicted in gray. Additionally, a portion of this SC electrical system is diverted to lower-power subsystems responsible for various aircraft operations, including environmental control and anti-icing mechanisms.

The fuel is stored in tanks, depicted in blue, situated above the passenger compartment. These tanks are strategically positioned both above and external to the passenger area to mitigate the risk of H₂ boiloff seeping into the cabin. As a result, a sizable central body structure is incorporated into the design specifically to accommodate these fuel tanks. An overview of baseline design characteristics of the geometries studied here are shown in Table 11.

To accommodate both the passenger cabin and the overhead LH₂ fuel tanks, an unconventional fuselage shape was deemed necessary. Three iterations of the OMLs for the aircraft are presented in Fig. 67. The MKI version represents the original OML configuration, whereas several design modifications led to the development of the MKII and MKIII versions; the rationale behind these changes will be elaborated upon throughout this section.

The wireframe curve and individual surfaces that make up the fuselage's OML are detailed in Fig. 68. Primary curves, which are aligned in the u parametric direction, are shown in shades of orange and green. The green curve was specifically introduced to facilitate the stretching of the fuselage skin to envelop the fuel tanks. Meanwhile, cross curves, aligned in the v parametric direction and displayed in blue and magenta, were designed in a heart shape to effectively encase both the passenger cabin and the fuel tanks.

In the original MKI configuration, a single through-curve mesh surface was used to shape the entire fuselage, including a recessed area intended for the cockpit window. However, this feature led to the formation of a significant stagnation zone, which in turn increased computational demands during both the forward and adjoint solution processes. To address this issue, the geometry was streamlined in the MKII design by eliminating this recessed cockpit area.

The method employed for surface generation in the MKII design is illustrated in Fig. 68. A fill surface, indicated in yellow, was utilized for crafting the nose, while a surface bridge, depicted in purple,

was established between the fill surface and the through-curve mesh surface. This arrangement was designed to maintain C^0 (position), C^1 (tangency), and C^2 (curvature) continuity throughout the fuselage. Each surface was designated as a B-surface, with each constituent wireframe curve being modeled as a Bézier curve. The technique outlined in Lauer and Ansell (Ref. 49) was employed to ensure G^0 , G^1 , and G^2 continuity across the connected Bézier curves.

In the MKIII iteration, the surface generation was simplified back to a single through-curve mesh, but with the recessed cockpit area still removed. Additionally, for MKIII, the fuselage surface near the fuselage-tail juncture was flattened (see Fig. 69) to prepare for tail rotations in future optimization studies. Also in MKIII, a twist distribution similar to the CRM's twist distribution (Ref. 90) was added to improve the wing aerodynamic characteristics. This also provided a better starting point for flow across the wing root section, as the free-form deformation control points there were held fixed in the optimization to avoid warping the mesh at the wing-fuselage juncture.

TABLE 11.—BASELINE DESIGN CHARACTERISTICS

Parameter	CHEETA-MKI	CHEETA-MKII	CHEETA-MKIII
Capacity			
Pax + crew	180	180	180
Max payload, lb	45,000	45,000	45,000
Mission			
Altitude, ft	37,000	37,000	37,000
Cruise Mach	0.773	0.773	0.773
Range, nm	2,935	2,935	2,935
Cruise Reynolds number	26.6×10^6	27.6×10^6	27.6×10^6
Design C_L	0.5	0.5	0.5
CG x -location, m	18.378	18.972	18.972
Wing geometry			
Aspect ratio	9.75	9.87	9.87
MAC, m	4.74	4.93	4.93
Root chord, m	7.756	8.036	8.036
Taper ratio	0.263	0.220	0.220
Total span, m	40.843	40.843	40.843
Wetted area, m^2	241.2	227.4	231.2
Volume, m^3	119.64	115.31	115.31
Fuselage geometry			
Chord, m	36.98	39.00	39.00
Span, m	7.74	8.40	8.40
Volume, m^3	1,755.0	2,101.6	2,101.6
Tail geometry			
Root chord, m	4.81	4.81	4.81
Aspect ratio	6.86	6.49	6.49
Taper ratio	0.3	0.3	0.3
Wetted area, m^2	97.25	82.16	81.26

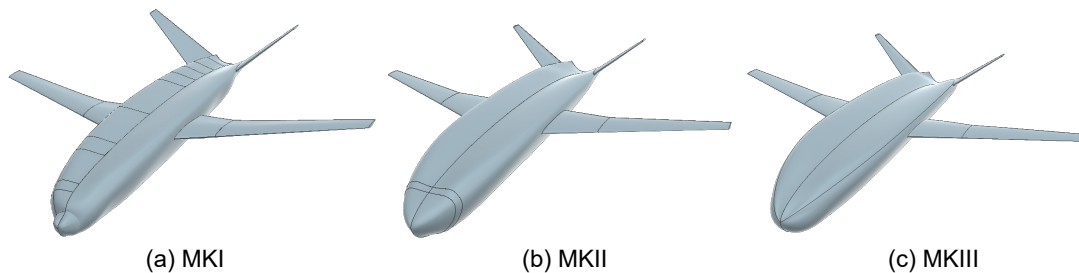


Fig. 67.—CAD models for the CHEETA MKI, MKII, and MKIII configurations.

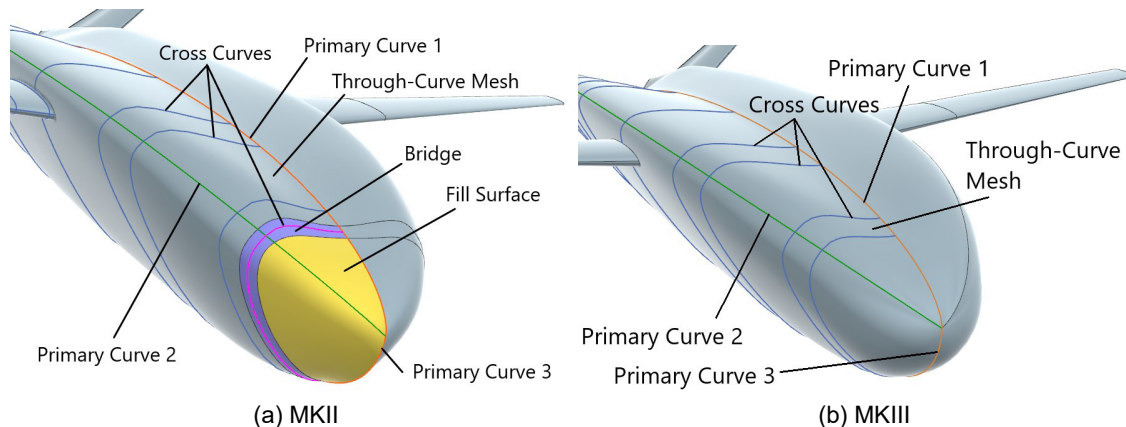


Fig. 68.—Curves and surfaces used to generate MKII and MKIII fuselage bodies.

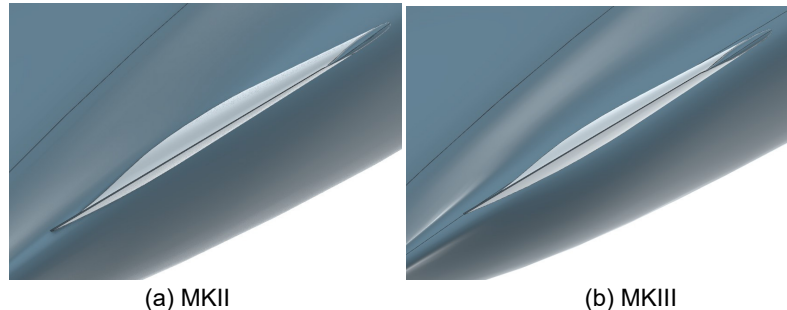


Fig. 69.—Tail-fuselage junction for MKII and MKIII.

A comparative analysis of the planforms for CHEETA MKI, MKII, and MKIII is provided in Fig. 70. It is worth noting that the V-tail depicted represents its vertical projection. The geometric parameters for the V-tail, as tabulated in Table 11, have been computed based on a flattened representation of this feature.

Several factors motivated the modifications in the planform design. Firstly, the fuselage was elongated by approximately 2 m to facilitate a smoother pressure recovery, a necessity arising from the increased internal height required for the sizable LH₂ fuel tanks. Secondly, the fuselage's tail section was broadened to better suit the future incorporation of BLI propulsors. Thirdly, the Yehudi break was extended to make room for a Yehudi-mounted DEP system. Additionally, a minor widening of the fuselage across its entire chord length was executed to raise the minimum material thickness from 100 to 150 mm. This adjustment is anticipated to be beneficial for future aerostructural optimization studies. The updated minimum thickness specification is also illustrated in Fig. 71.

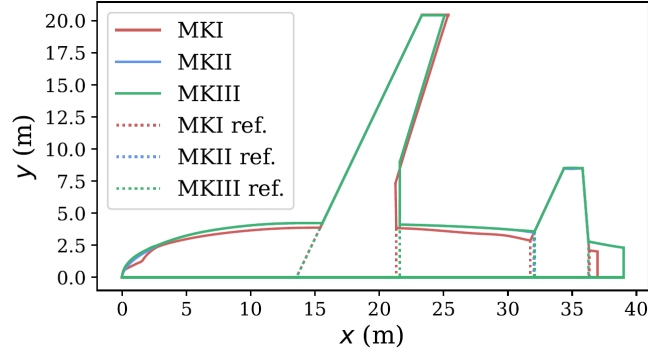


Fig. 70.—Planform comparison for CHEETA MKI, MKII, and MKIII configurations; dotted lines show extension of leading edge and trailing edge for wing and V-tail, used to compute geometric parameters displayed in Table 11.

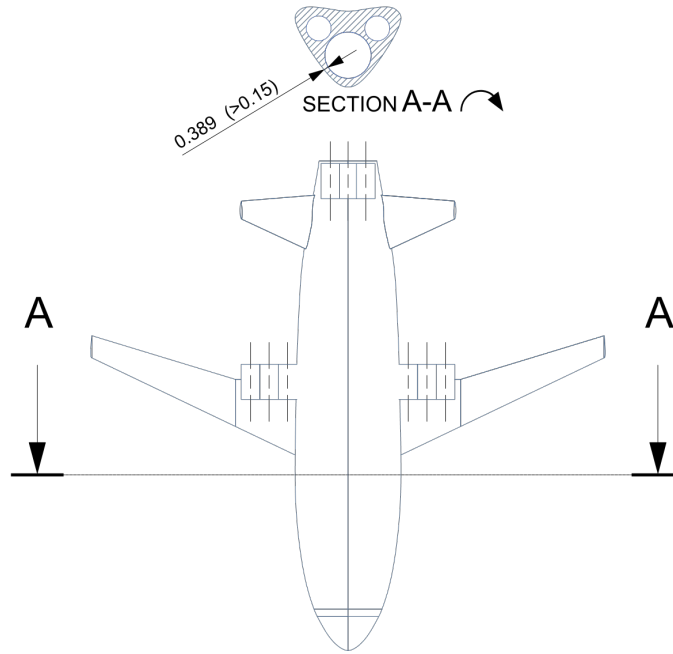


Fig. 71.—Fuselage section showing internal fuel tanks and passenger cabin; all dimensions in meters.

2.3.2.2 Numerical Framework

To evaluate the aerodynamic performance of the CHEETA configuration, a 3D RANS framework is employed wherein the fluid residual

$$\mathcal{R}(\mathcal{F}) = 0 \quad (30)$$

is solved to evaluate the vector of state variables

$$\mathcal{F} = \begin{pmatrix} \rho \\ \rho \mathbf{V} \\ \rho E \\ \rho v \end{pmatrix} \quad (31)$$

Spatial discretization is carried out using the finite volume method, as outlined in Jameson (Ref. 93), Barth (Ref. 94), and Hirsch (Ref. 95), tailored for an unstructured mesh. This method incorporates a dual vertex scheme paired with a standard edge-based structure, as described in Economou (Ref. 96) and Sanchez et al. (Ref. 97). This approach aligns with the efficient data structuring principles within SU2. Following this discretization, the unsteady Navier-Stokes equation

$$\mathcal{R}(\mathcal{F}) = \frac{\partial \mathcal{F}}{\partial t} + \nabla \cdot \mathbf{F}^c - \nabla \cdot \mathbf{F}^v - Q = 0 \quad (32)$$

is reformulated as:

$$\frac{d}{dt} \int_{\Omega_{F_i}} \mathcal{F} d\Omega_F + R_i(\mathcal{F}) = 0 \quad (33)$$

Given that the scope of the shape optimization study is limited to steady-state cruise conditions, Eq. (33) is expressed in terms of time-invariant cell volume $|\Omega|_i$ as

$$|\Omega|_i \frac{\Delta \mathcal{F}_i}{\Delta t_i} = -\mathcal{R}(\mathcal{F}^{n+1}) \quad (34)$$

A first-order linearization of Eq. (34) and its substitution in Eq. (33) yields in the linear system:

$$\mathcal{R}(\mathcal{F}) : \left(\frac{|\Omega|_i}{\Delta t_i} + \sum_{j \in N(i)} \frac{\partial \mathcal{R}_i(\mathcal{F}^n)}{\partial \mathcal{F}_j} \right) \Delta \mathcal{F}_j = \mathcal{R}_i(\mathcal{F}^n) \quad (35)$$

where the Jacobian is approximate due to the various simplifying assumptions made in the calculation of the partial terms over the domain. An implicit solution procedure is used to solve the linear system shown in Eq. (35). Even though the Jacobian is first-order accurate, the implicit method allows for a higher Courant number C that assists in numerical stability of a stiff problem.

In an effort to aeroelastically tailor the CHEETA configuration, the native SU2 framework is extended to the aeroelastic domain by defining an interface Γ and coupling it with a linear finite volume solver in space and time, thus augmenting the fluid F and fluid mesh residual M with a structure residual S . More information on the large-scale aeroelastic framework can be found in Ranjan (Ref. 78).

All design optimization experiments reported in this manuscript use gradient-based optimization techniques. Accuracy of objective function sensitivities and constraint Jacobian is crucial to achieve converged optimized designs. A lagged, coupled adjoint framework, consisting of sensitivities derived using closed-form analytical expressions and iterative methods, obtained using SU2's reverse-algorithmic differentiation toolkit (Albring et al., Ref. 98) is developed and employed. For a generic optimization problem:

$$\begin{aligned} \min_{\alpha} J(\mathcal{U}, \mathcal{F}, \mathcal{X}, \alpha) \\ s. t. \mathcal{R}(\mathcal{F}, \mathcal{X}, \alpha) - \mathcal{F} &= 0 \\ S(\mathcal{U}, \mathcal{F}, \mathcal{X}, \alpha) &= 0 \\ M(\mathcal{U}, \alpha) - \mathcal{X} &= 0 \end{aligned} \quad (36)$$

the coupled adjoint equations for fluid, fluid mesh and structure discipline are derived as:

$$\begin{aligned}\frac{\partial J}{\partial \mathbf{u}} &= -\lambda_M^T \frac{\partial M}{\partial \mathbf{u}} - \lambda_S^T \frac{\partial S}{\partial \mathbf{u}} \\ \lambda_F^T &= \frac{\partial J}{\partial \mathcal{F}} + \lambda_F^T \frac{\partial \mathcal{R}}{\partial \mathcal{F}} + \lambda_S^T \frac{\partial S}{\partial \mathcal{F}} \\ \lambda_M^T &= \frac{\partial J}{\partial \mathcal{X}} + \lambda_F^T \frac{\partial \mathcal{R}}{\partial \mathcal{X}} + \lambda_S^T \frac{\partial S}{\partial \mathcal{X}}\end{aligned}\tag{37}$$

Since the scope of the current report is limited to shape optimization, all cross-coupling terms in Eq. (58) are set to zero. For mesh deformations, the 3D surface mesh is deformed using free-form deformation with both Bézier and second-order uniform B-splines as interpolation functions. The choice of fast fluid dynamics (FFD) interpolation/blending functions is observed to influence convergence and numerical stability to a great extent. Fig. 72 illustrates the baseline CRM (Ref. 90) and random surface perturbations induced using Latin hypercube sampling at the same magnitude. The second-order B-spline results in local deformation (owing to its local support) as compared to Bézier curves with global support.

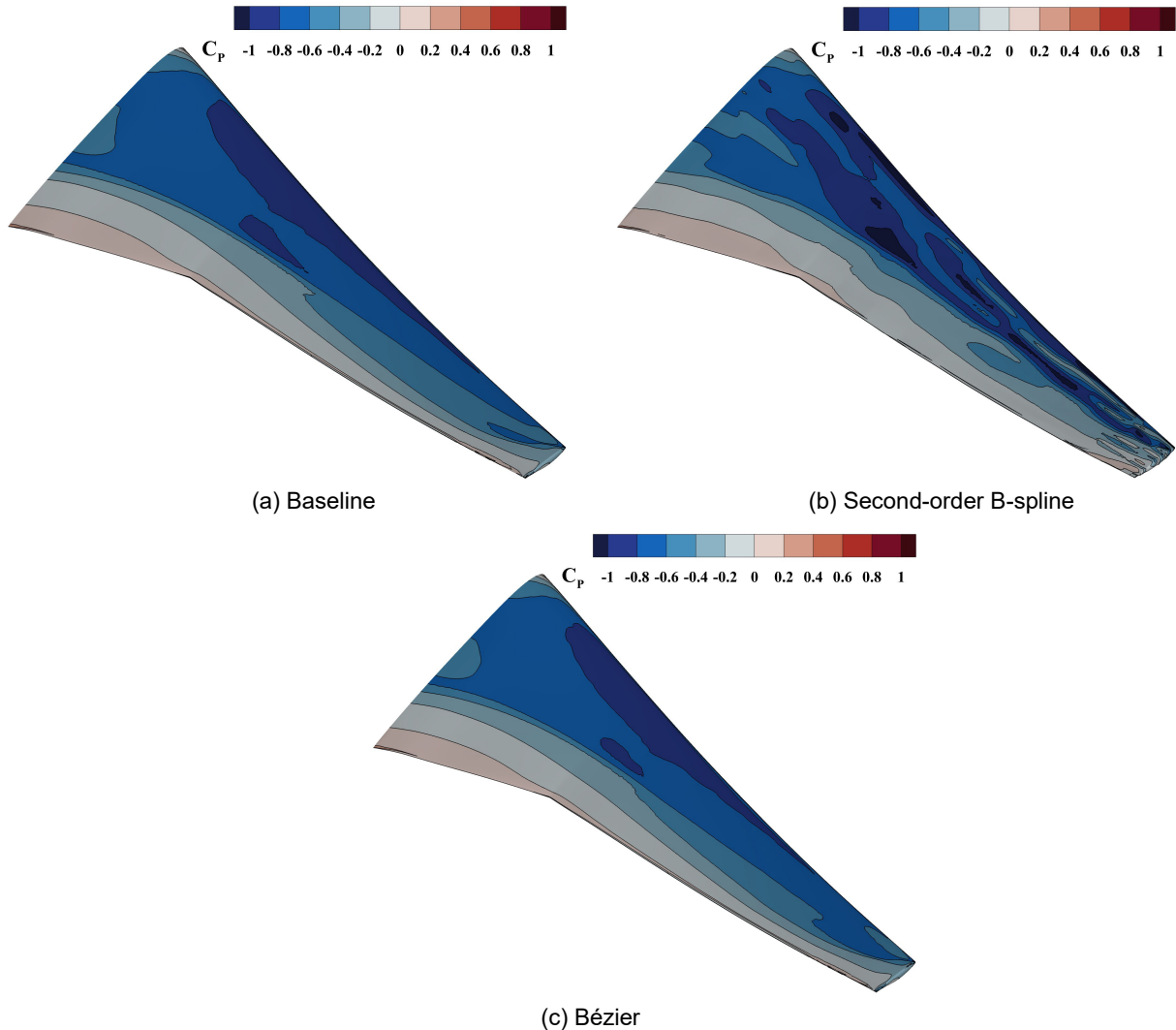


Fig. 72.—Effect of FFD blending functions on surface perturbation of NASA CRM (Ref. 90).

2.3.2.3 Single-Point Aerodynamic Shape Information

2.3.2.3.1 Grid Convergence Study

A comprehensive grid convergence study was undertaken to guarantee mesh-independent optimization outcomes. Two computational domains were evaluated: (1) an inviscid domain with a predominant quadrilateral surface tessellation and (2) a viscous domain with triangle-dominant surface tessellation, complemented by prism elements to address boundary layer physics. Fig. 73 illustrates the results from the grid convergence study conducted at C_{LW} : 0.50. Both drag and α convergence trends suggest mesh independence with decreasing grid refinement factor H . A similar study for the Euler domain was conducted but is not shown here.

Table 12 lists the mesh generated for grid convergence study along two dimensions. In the first dimension, the surface element edge length is consistently scaled down while maintaining the average wall spacing constraint. Meshes M00 to M04, M05 to M09, and so forth are results of such refinements. As a second dimension of refinement, the average y^+ constraint is changed. Meshes M00, M05, M10, and so forth are refined for constant edge length but with decreasing wall spacing. Meshes M25 to M30 represent the quad-dominant cases that are used for the inviscid shape optimization study discussed in this report.

Fig. 74 shows the quad-dominant M25 and the viscous M12 surface domains. These were chosen for shape optimization studies, balancing computational expenses with precision. Initial findings for M25 are explored in this report, whereas results for M12 shape optimization will be addressed in a forthcoming study.

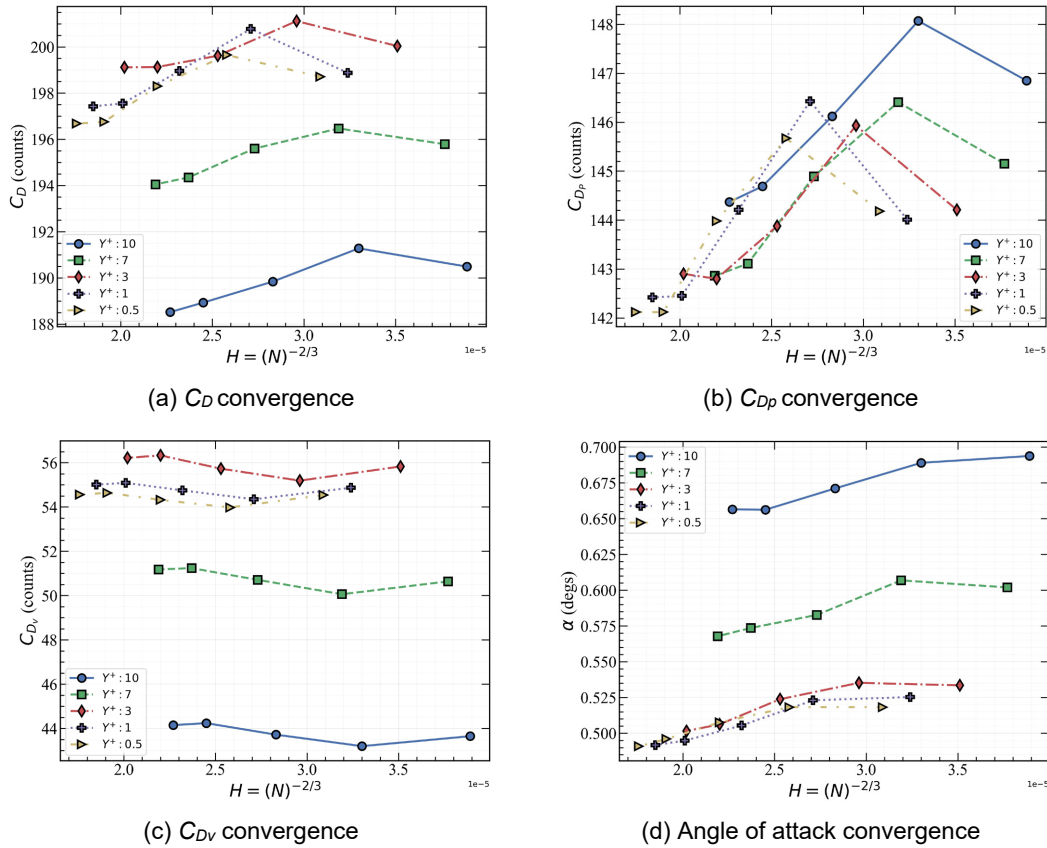
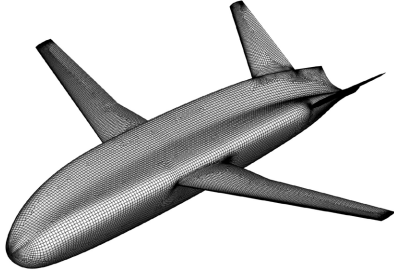


Fig. 73.—Drag convergence characteristics.

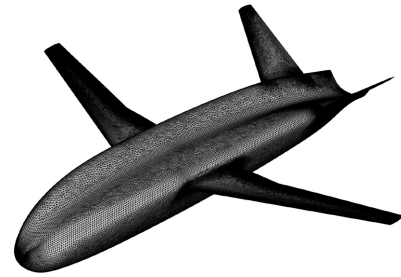
TABLE 12.—INVISCID AND VISCOUS GRID SUMMARY

Mesh	Average, y^+	S , m	Nodes, N	Elements, E
M00	10	5.27256×10^{-5}	4,121,590	8,848,294
M01	10	5.27256×10^{-5}	5,278,143	11,339,642
M02	10	5.27256×10^{-5}	6,658,435	14,311,373
M03	10	5.27256×10^{-5}	8,225,619	17,681,614
M04	10	5.27256×10^{-5}	9,270,107	19,933,397
M05	7	3.71716×10^{-5}	4,323,072	9,254,468
M06	7	3.71716×10^{-5}	5,552,229	11,888,364
M07	7	3.71716×10^{-5}	7,005,256	15,002,077
M08	7	3.71716×10^{-5}	8,665,532	18,567,265
M09	7	3.71716×10^{-5}	9,773,914	20,934,569
M10	3	1.59306×10^{-5}	4,806,608	10,224,424
M11	3	1.59306×10^{-5}	6,220,531	13,225,361
M12	3	1.59306×10^{-5}	7,856,303	16,700,210
M13	3	1.59306×10^{-5}	9,721,508	20,682,865
M14	3	1.59306×10^{-5}	11,006,757	23,404,124
M15	1	5.27256×10^{-6}	5,433,704	11,471,597
M16	1	5.27256×10^{-6}	7,079,855	14,941,105
M17	1	5.27256×10^{-6}	8,963,464	18,918,971
M18	1	5.27256×10^{-6}	11,106,456	23,451,679
M19	1	5.27256×10^{-6}	12,600,594	26,596,581
M20	0.5	2.63628×10^{-6}	5,830,758	12,275,961
M21	0.5	2.63628×10^{-6}	7,624,749	16,036,896
M22	0.5	2.63628×10^{-6}	9,661,574	20,319,989
M23	0.5	2.63628×10^{-6}	11,970,498	25,179,034
M24	0.5	2.63628×10^{-6}	13,600,746	28,606,290
M25	1,000	5.27256×10^{-3}	965,632	1,797,942
M26	1,000	5.27256×10^{-3}	1,505,193	2,752,146
M27	1,000	5.27256×10^{-3}	1,608,056	2,895,636
M28	1,000	5.27256×10^{-3}	2,493,625	4,506,150
M29	1,000	5.27256×10^{-3}	2,584,868	4,234,833
M30	1,000	5.27256×10^{-3}	2,696,337	4,220,599

Fig. 75 represents the multipoint aerodynamic performance attributes for a rigid structure. For a range of payload values (aircraft C_L), Fig. 75(a) indicates the fuselage to be the primary drag contributor, followed by the wing. Contributions from the tail are ≈ 20 drag counts for the shown ranges of aircraft C_L . A closer look into Fig. 75(c) and Fig. 75(d) suggests that these high drag contributions are driven by viscous and pressure drag, owing to the large fuselage wetted area and high suction pressure on the wing, respectively.

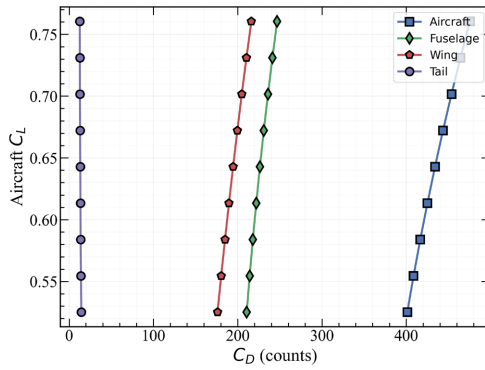


(a) M25: Inviscid quad-dominant surface mesh

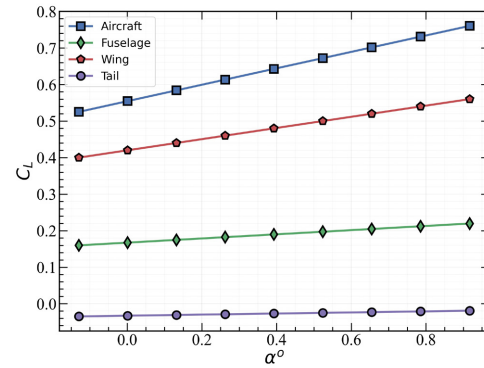


(b) M12: Viscous triangle-dominant surface mesh

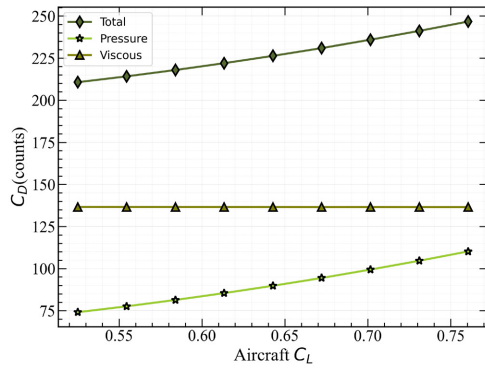
Fig. 74.—Surface mesh definition for inviscid and viscous analysis.



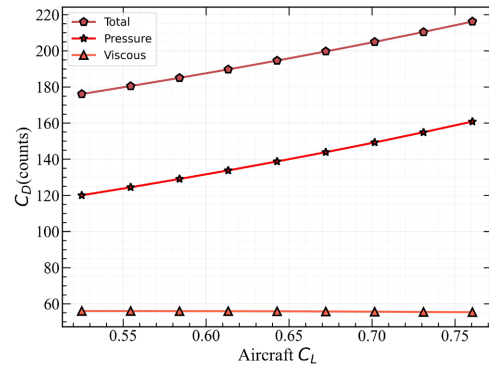
(a) Aircraft drag polar



(b) Aircraft lift decomposition



(c) Fuselage drag decomposition



(d) Wing drag decomposition

Fig. 75.—Aircraft multipoint performance in viscous domain.

2.3.2.4 Inviscid shape optimization

Table 13 presents the problem formulation, wherein C_{D_p} is minimized at a single design point, subject to flow and geometric constraints. The surface mesh is encapsulated by multiple free-form deformation boxes with the displacement of control points as design variables. These displacements are mapped to the surface mesh using Bézier curves as the mapping polynomial. Three vectors of design variables are defined:

1. ξ —Defined over the wing and constrained to move along the z-normal direction
2. ζ —Defined over the fuselage and allow deformation along the y-normal (spanwise) direction
3. ϑ —Encapsulate the fuselage and constrained to move along the y-normal direction

Fig. 76 illustrates the FFD configuration used in this study. In addition to the FFD control points, the flow angle of attack (α) is treated as a design variable, primarily to satisfy the lift condition. The aircraft pitching moment is constrained to be greater or equal to that at design condition and is computed about the neutral point (revised from CHEETA MK2), set at $x = 19.62$ m from the nose. The tail rotation angle (ε) is introduced to trim the aircraft. A fuselage volume constraint is enforced for scalar β_1 . Similarly, the wing volume is constrained based on the initial wing volume and scalar β_2 . Lastly, 10 spanwise thickness constraints are enforced based on the initial value and a spanwise constant scalar β_3 . Lower values for β_* allow for larger feasible area for the optimizer. All FFD-based design variables are collectively defined as global design vector Ξ .

TABLE 13.—SHAPE OPTIMIZATION PROBLEM FORMULATION

	Function	Description	Quantity
Minimize	C_{Dp}	Single-point drag	
With respect to	ξ	Vertical perturbations of wing FFD control points	368
	Z	Lateral perturbations of fuselage FFD control points	109
	ϑ	Vertical perturbations of fuselage FFD control points	122
	α	Angle of attack	1
	ε	Tail rotation angle	1
Subject to	$C_L = 0.617975$	Lift constraint	1
	$C_{My} \geq -0.006543$	Pitching moment constraint	1
	$\mathcal{V}_F \geq \beta_1 \times \mathcal{V}_{Finitial}$	Fuselage volume constraint	1
	$\mathcal{V}_W \geq \beta_2 \times \mathcal{V}_{Winitial}$	Wing volume constraint	1
	$(t/c)_W \geq \beta_3 \times (t/c)_{Winitial}$	Wing spanwise thickness constraint	10

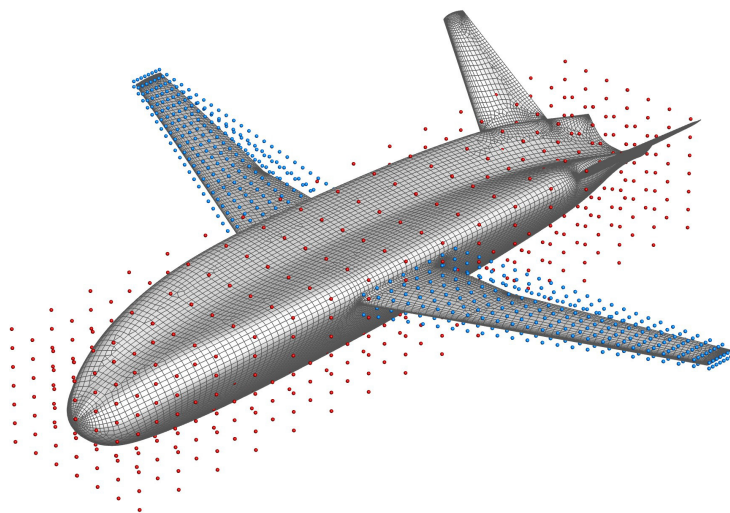


Fig. 76—Free-form deformation design space definition.

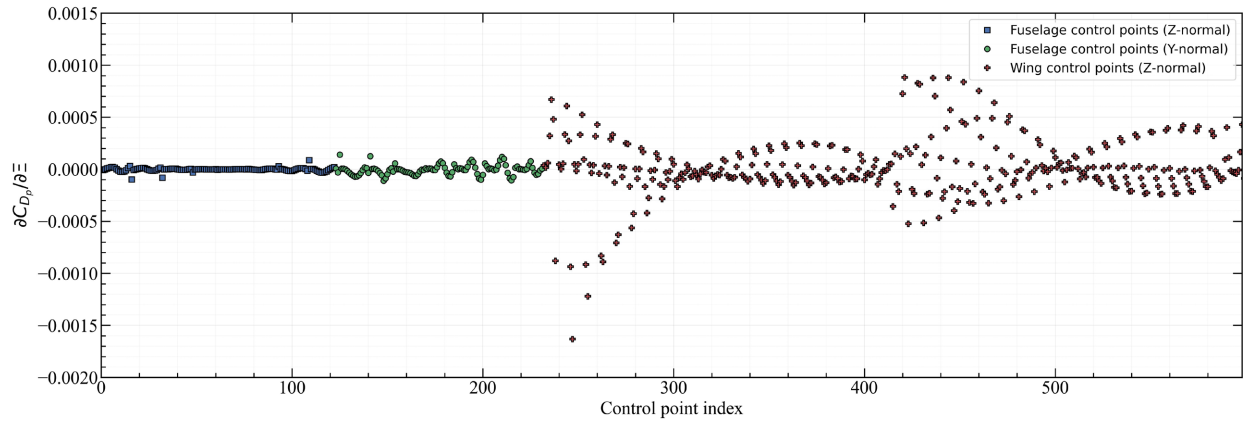


Fig. 77.—Initial C_{Dp} sensitivity.

Fig. 77 depicts the C_{Dp} sensitivities in relation to the global design variable vector Ξ for the design space shown in Fig. 76. The initial 122 control point indices are in ϑ . Subsequently, 109 control points are defined in ζ , while the final 368 control points, denoted as ξ , encapsulate the wing surface and have constraints to shift solely along the z-axis. The FFD control point indices are sequentially identified, starting from the lower surface and progressing counterclockwise. A perusal of Fig. 77 reveals pronounced drag sensitivity on the wing surface. Given the minimal pressure drag contributions from the fuselage, as previously observed in Fig. 75(c), sensitivities $\partial C_{Dp}/\partial\vartheta$ and $\partial C_{Dp}/\partial\zeta$ are nearly negligible. For a viscous flow domain, both $\partial C_{Dp}/\partial\vartheta$ and $\partial C_{Dp}/\partial\zeta$ are anticipated to exhibit greater magnitudes.

The gradient-space illustrations of Fig. 77 offer valuable insight into how the optimizer may deform the geometry to minimize the objective function. For the problem formulation described in Table 13, one can expect significant changes to the wing in an effort to reduce C_{Dp} . However, these representations are only an instance of the continuously evolving gradient space.

The design problem described in Table 13 is solved using IPOPT (Ref. 99), a primal-dual interior point method. At each design iteration, the objective function is evaluated from Eq. (55). Fig. 78(a) depicts the design evolution wherein a 9.91% reduction in C_{Dp} is shown. Several optimization scenarios, each with varying scaling parameters, were examined to pinpoint a step size conducive to stable design convergence. For all scenarios, a function convergence tolerance was set at 10^{-6} . Further specifications included a barrier function tolerance of 10^{-8} and a dual infeasibility tolerance at 10^{-6} . Fig. 78(b) reveals a significant reduction—up to five orders of magnitude—in the Lagrangian gradient, suggesting a nearly optimal solution. Meanwhile, Fig. 78(c) underscores a steady decline in the barrier function, signifying the optimizer’s aptitude in addressing the constrained optimization challenge while maintaining solution feasibility.

In all scenarios depicted in Fig. 78, the fuselage volume constraint parameter β_1 was fixed at 0.99. Simultaneously, the wing volume constraint parameter β_2 was established at 0.95, permitting alterations to the wing volume by the optimizer. This decision to opt for a reduced lower boundary for the wing volume was influenced by the wing’s lack of fuel tanks. The wing thickness constraint parameter β_3 was held constant at 0.25, thus increasing the feasible region in the design space. Fig. 79 contrasts the baseline CHEETA MKIII with the optimized wing, focusing on the spanwise twist and thickness distribution. Aiming to reduce drag, the optimizer reduces the inboard twist and introduces a washin for $\eta \geq 0.60$. As shown in Fig. 79(b), there is an uptick in spanwise thickness near the wing root, followed by a steady reduction in t/c to 7.2% at the wing tip.

Fig. 80 illustrates the gradient space at the optimal solution. As expected, the optimizer yields a design that is less sensitive to C_{Dp} , especially over the wing surface. The existent nonzero sensitivities at

the root on the pressure and suction sides of the wing are attributed to FFD control points held fixed during optimization. This was done to prevent the volume and surface cells from being distorted due to FFD perturbation.

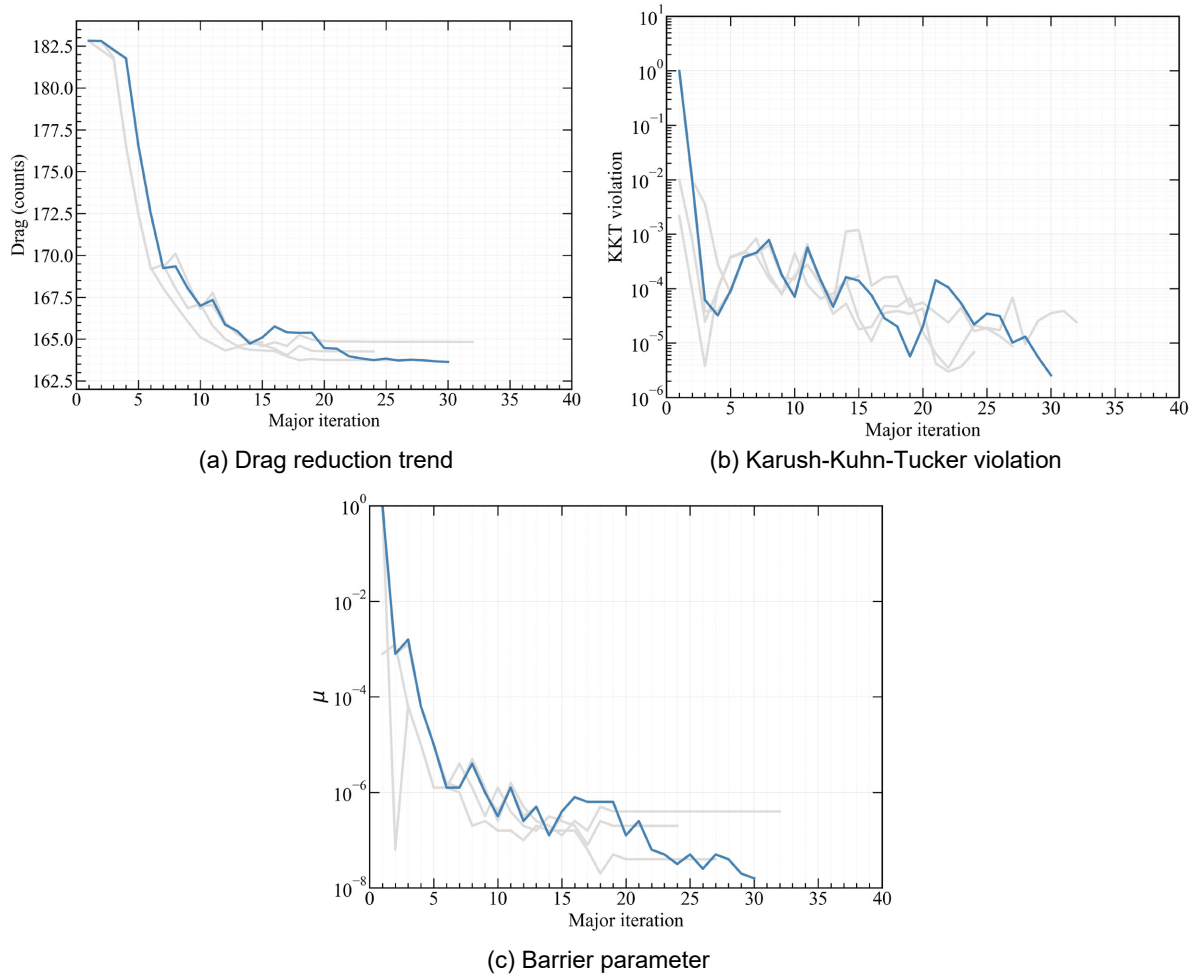


Fig. 78.—Optimization convergence characteristics.

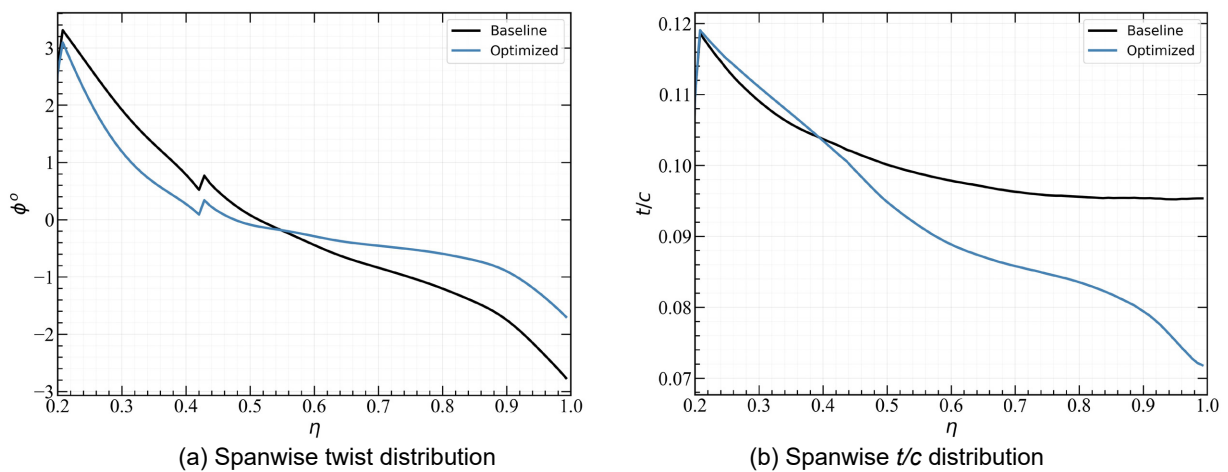


Fig. 79.—Baseline versus optimized geometric characteristics.

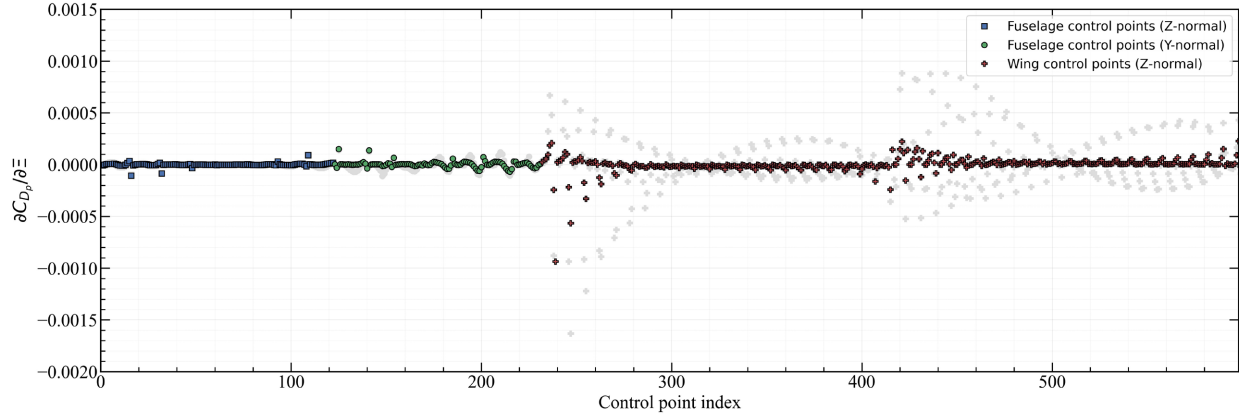


Fig. 80.—Initial versus final C_{Dp} sensitivity.

The span load for the aircraft from fuselage center plane to wing tip is shown in Fig. 81 for the baseline configuration, the MKIII optimized configuration, and an ideal elliptical span loading. The elliptical span loading was chosen such that the resultant C_L was the same as the target C_L of the full aircraft configuration. That is, the elliptical span loading satisfies Eq. (59) by setting the loading at the center plane according to Eq. (60).

$$\frac{b\bar{c}}{S} \int_{-1}^1 \frac{c(\eta)C_l(\eta)}{\bar{c}} d\eta = C_L = 0.618 \quad (38)$$

$$\left[\frac{cC_l}{\bar{c}} \right]_{y=0} = \frac{4SC_L}{\pi b\bar{c}} \quad (39)$$

$$cC_l = \int_{x_{le}}^{x_{te}} [C_{p,l}(x) - C_{p,u}(x)] dx \quad (40)$$

The loading was calculated by summing the values produced by Eq. (61) for all surfaces at each spanwise slice, which means that the effects of fuselage, wing, and tail are all shown. Both the baseline and optimized extracted spanload distributions were found to numerically integrate to the target C_L according to Eq. (59) with an error of less than 0.04%. It is evident that the optimized case more closely resembles the ideal elliptical spanloading than the baseline case for the majority of the outer wing section ($\eta \gtrsim 0.45$). The associated decrease in induced drag is likely where a significant portion of the C_d improvement occurs. In addition, because of the increased outboard loading in the optimized case, the optimized case also has a smaller loading on the wing section inboard of the Yehudi break to compensate, which could potentially lead to small savings in viscous drag. Note that the sharp decrease in loading between $\eta = 0.40$ and $\eta = 0.45$ is caused by the accounting for the negative lift generated by the tail.

Surface pressure coefficient distributions extracted at several spanwise stations along the wing are shown in Fig. 82. Note that the nondimensional span locations are relative to the exposed wing root y -location ($\neq 0$). In each of the first four subplots, the optimized wing sections have much more gradual pressure recovery regions compared to the baseline wing sections. In addition, the optimized wing sections are much less aft-loaded than the baseline wing sections, which leads to a less aggressive pitching moment. This decrease in pitching moment magnitude will be advantageous in limiting trim drag in further optimization studies where a $C_{My} = 0$ constraint is imposed by adjusting the tail angle of attack.

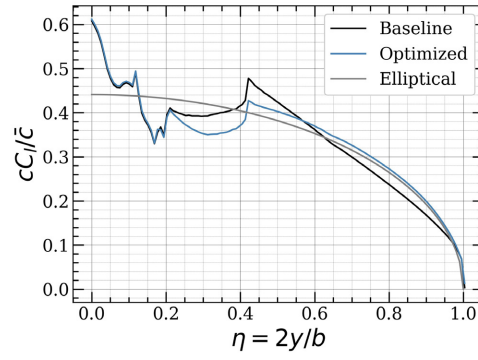
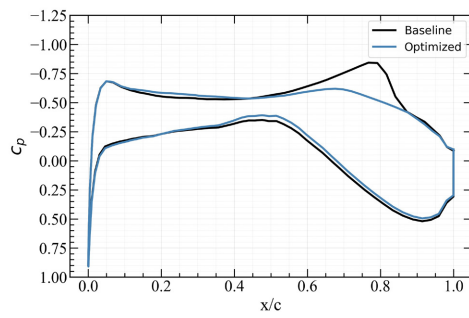
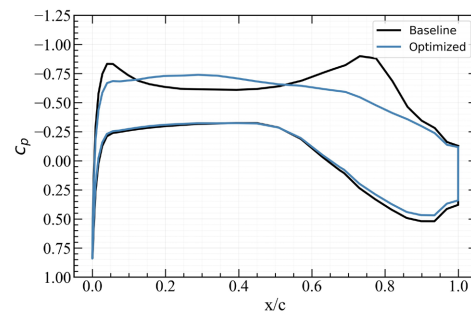


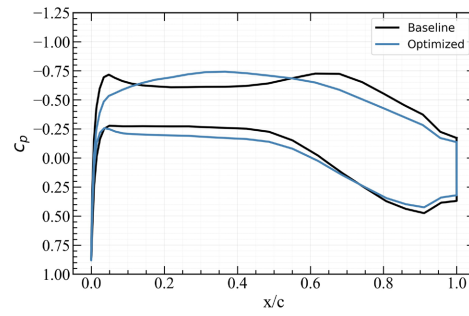
Fig. 81.—Spanwise lift distribution.



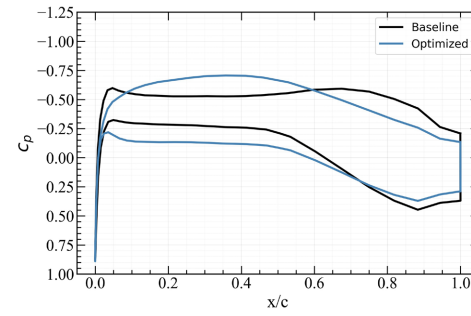
(a) $\eta_w: 0.05$



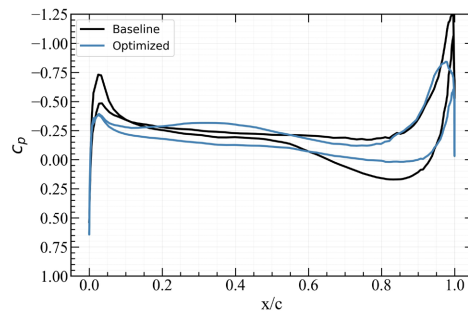
(b) $\eta_w: 0.25$



(c) $\eta_w: 0.50$



(d) $\eta_w: 0.75$



(e) $\eta_w: 0.99$

Fig. 82.—Spanwise C_p distribution.

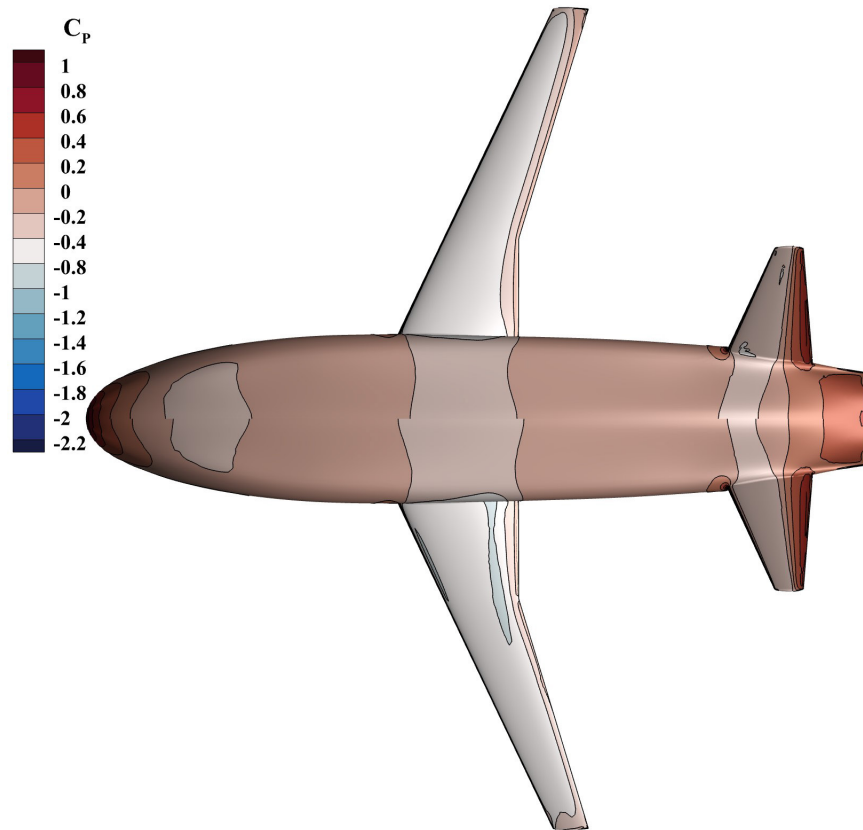


Fig. 83.—Baseline (lower half) versus optimized (upper half) surface C_p distribution.

In the final subplot, the surface pressure coefficient decreases dramatically close to the trailing edge. This is due to the highly 3D flow at the tip, where a significant portion of the local flow velocity is in the spanwise direction. The suction peak here, as in most of the other C_p distributions shown, has been reduced in strength by the decrease in geometric twist shown in Fig. 79. The surface C_p distribution for the baseline and optimized configuration is shown in Fig. 83. The shift in the fuselage surface C_p isobars across the symmetry plane is attributed to change in aircraft α by $+0.1212^\circ$.

3.0 Electrical Component Technology

Throughout the ULI research program, advancements in the design of several cryogenic electrical power system components were made. Relevant technologies that were matured through the current study include advanced, high-power-density cryogenic electric machines, low-loss cryogenic power electronics, and advanced SC power transmission and protection strategies.

3.1 Cryogenic Electric Machine Design

The design of a new class of cryogenic electric machines served as a cornerstone to the technology development effort of the CHEETA research program. In addition to identifying the topological trades inherent to the design of a fully SC electric machine, a wealth of analysis was necessary to characterize mechanical, electrical, and thermal considerations for this class of machine. Additional means of motor cryogenic cooling and quench protection were studied in detail in development of a 2.5-MW machine conceptual design.

3.1.1 Motor Design Topology Trade Studies

Throughout the current study, a Pareto front optimization strategy was leveraged to identify desirable characteristics inherent to the 2.5-MW SC machine design for aircraft applications. This process focused on the thermal and mechanical design components of the machine, as well as selection and sizing of active components that resulted in preferable trades in weight and efficiency of the machine. Additional motor topologies were also studied, allowing the anticipated outcomes of a fully SC machine to be compared to alternate architectures.

The CHEETA concept airplane is propelled by fully SC electric motors, which are cooled by the plane's LH₂ fuel supply (Fig. 84). UIUC was tasked with designing a fully SC aircraft propulsion and performing experiments to design risk. The following report details efforts over the course of the CHEETA project.

This report describes the electromagnetic and thermal design of the CHEETA fully SC aircraft propulsion motor. It details the design constraints, describes the selected machine topology, and summarizes experiments aimed at reducing design risk.

A 2.5-MW electrical propulsion machine designed for commercial electric propulsion requires high-power-density (>25 kW/kg) and high-efficiency (>99%) electrical motors. The high electrical frequencies in high-speed fully SC machines result in significant AC losses. Conventional SC machine designs use iron in their stator and rotor to reduce the AC losses and conductor usage at the expense of power density. These losses and power densities must be improved to surpass conventional electric-machine power densities and meet the specifications required for commercial electric propulsion (Refs. 100 and 101). H₂-powered all-electric aircraft provide a design space where AC losses can be optimized by leveraging the available free cooling of the cryogenic fuel. Several design studies have been conducted on high-speed fully SC machines for electric propulsion. A compilation and analysis of previously proposed high-speed SC machines are reported in Reference 102. Machine topology significantly affects AC losses, and optimization must be performed to identify the best topology for minimizing these losses. Most previous design studies demonstrate that fully SC machines have the potential to meet the power density required for electric propulsion. However, minimizing AC losses to reduce required cryogenic power and weight is still a roadblock for propulsion applications. Table 14 shows some of the example machine design studies conducted for electric propulsion. While these provide crucial weight and AC-loss information, an optimization to minimize AC losses was never carried out. Since AC losses of a given SC wire and machine weight are influenced by several geometric and operational motor parameters, the multi-objective optimization discussed in Section 3.1.1.2 is used to obtain the Pareto-optimal front.

This section details efforts to design and analyze a fully SC machine optimized for a H₂-powered regional all-electric aircraft for the CHEETA aircraft concept. A conceptual aircraft with eight under-wing distributed ducted fan systems is shown in Fig. 85. A proposed electric power distribution system with a cooling circuit is shown in Fig. 86. In addition to SC electric motors, SC cables and cryogenic-cooled electric drive drains will be employed to achieve higher system efficiency. A hybrid-design topology is optimized utilizing low AC-loss MgB₂ wires. The optimization is targeted to achieve 50 kW/kg specific power while limiting the cryogenic heat rejection to less than 3 kW. Various machine topologies are explored for this application. Optimization targets minimizing weight and AC losses in the armature coils. Tradeoffs between iron-shield and active-shield coil designs are explained.

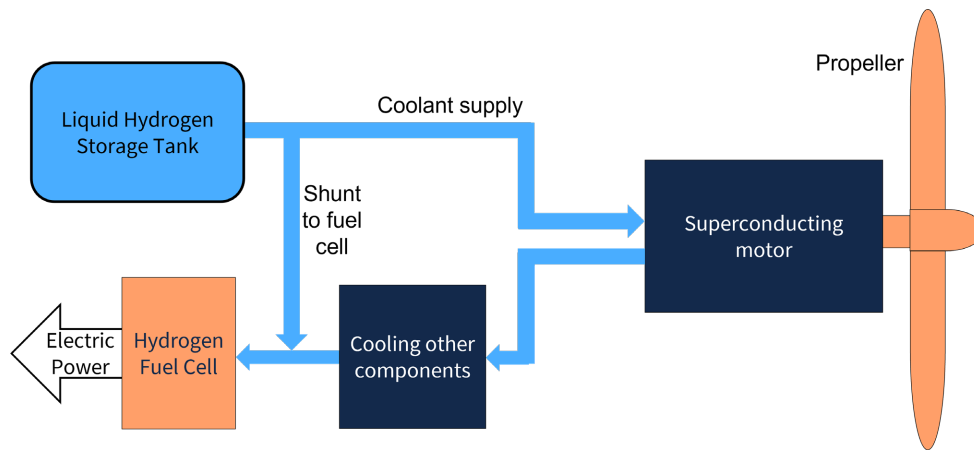


Fig. 84.—H₂ coolant and fuel circuit in the proposed CHEETA airplane.

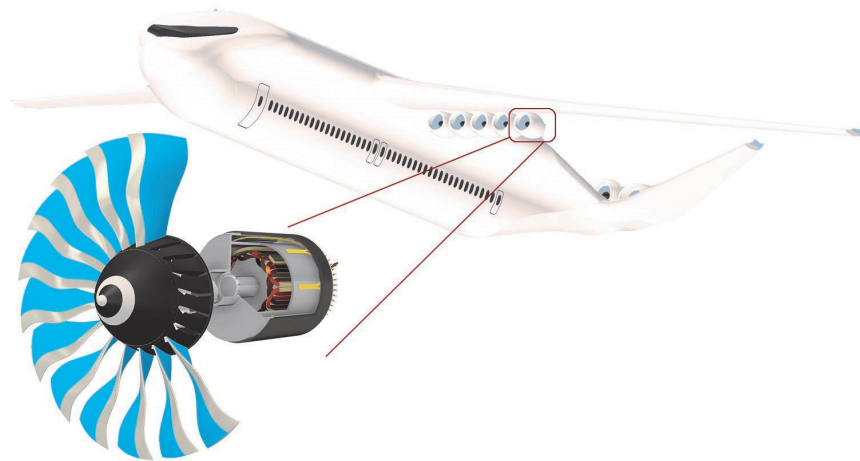


Fig. 85.—Electric-propulsion motor and conceptual DP aircraft.

CHEETA is a concept proposed to utilize the high specific-energy content of LH₂ through fuel-cell energy conversion and an ultra-efficient DEP system. The concept uses LH₂ as the energy source and a cryogen to enable a highly efficient electric system. This facilitates the usage of SC machines with SC power cables to efficiently convert electrical energy into mechanical energy. The system provides free cryogenic cooling for SC electrical components by letting the GH₂ used in the fuel cells evaporate by heat losses in the SC motor and cables. Fig. 87 shows the H₂ cooling circuit proposed for the CHEETA aircraft. If the required cryogenic power to remove generated heat losses in the electrical system is maintained below the available free cooling, an ultra-efficient electrical system could be obtained. This study explores an aircraft that uses 1068 GJ of total energy during a mission and has 20 MW of combined propulsion. The weight of the required LH₂ fuel is estimated to be 14,833 kg. Assuming latent heat of evaporation of 447 kJ/kg, 6.6 MJ of cooling energy is available in the aircraft. Fig. 88 shows the H₂ flow rate to the fuel cell during the entire mission profile. Designing the electrical machines at the cruise condition and budgeting the cooling to the available H₂ flow rate during the cruise will guarantee the optimal performance of the motor. The power of each motor and the available cooling power per motor can be estimated, assuming eight distributed underwing propellers. The required amount of H₂ flow rate during the cruise condition can be evaluated by taking a total cooling budget of 70 kW for electrical motors.

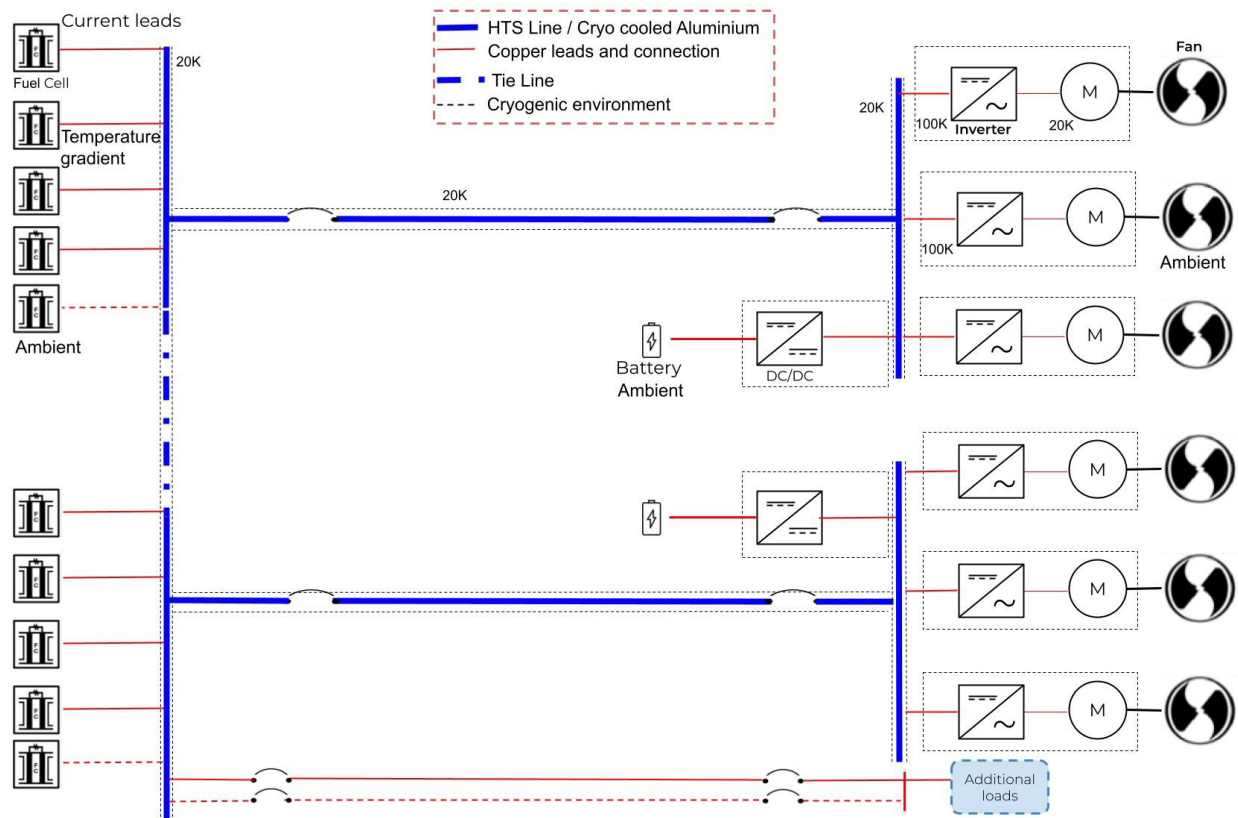


Fig. 86.—Electric propulsion power grid.

TABLE 14.—PROPULSION ELECTRICAL MACHINE SPECIFICATIONS

Reference machine	Ref. 103	Ref. 104	Ref. 105	Proposed
Core	Air	Iron	Iron	Air
Power, MW	0.85	10	5	2.5
Armature winding	MgB ₂	MgB ₂	MgB ₂	MgB ₂
Field winding	REBCO	YBCO	YBCO	MgB ₂
Speed, rpm	12,000	3,000	5,000	4,500
Power density, kW/kg	11.7	25.6	25.2	≥25
AC loss, kW	5.378	49.9	242	3
Air gap flux density, T	1	1		3
Air gap length, mm	20	20	20	5
OD, m	0.5	0.5	0.35	0.5
Operating temperature, K	20	20	20	20
Armature current density, A_{RMS}/mm^2	200	200	130	200
Field current density, A/mm^2	200	200	170	200
Shield current density, A/mm^2	-----	-----	-----	200

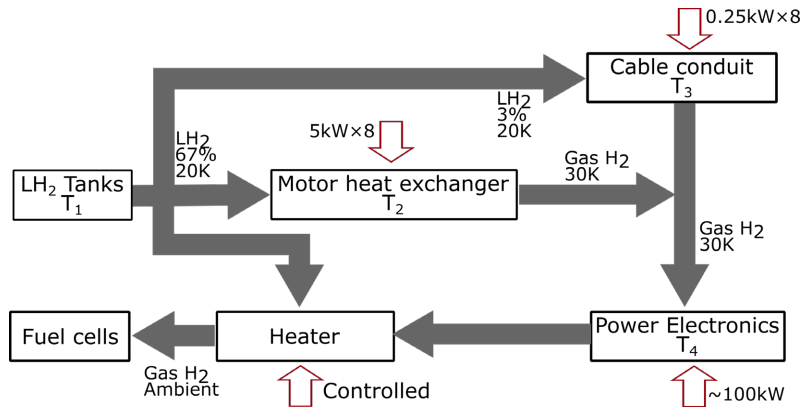


Fig. 87.—H₂ cooling circuit.

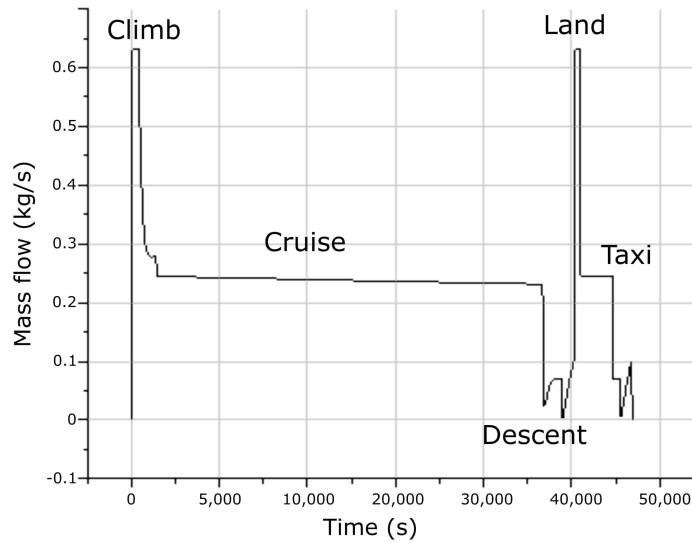


Fig. 88.—H₂ flow rate versus mission profile.

$$\text{Power of each motor} = 20/8 = 2.5 \text{ MW} \quad (41)$$

$$\text{Required flow rate} = 70 \text{ kW}/447 \text{ kJ/kg} = 0.156 \text{ kg/s} \quad (42)$$

This required flow rate is 68% of the cruise H₂ flow rate. For a safety margin, machines are designed with 60% of this allocated cooling.

$$\text{Available free cooling power per motor} = 40/8 = 5 \text{ kW} \quad (43)$$

$$\text{Targeted power density of each motor} \geq 25 \text{ kW/kg} \quad (44)$$

Therefore, if each 2.5-MW motor is designed to have AC losses below 3 kW, no additional cryo-cooling refrigeration equipment is necessary. A cooling method that utilizes the available LH₂ must be developed and may add considerable weight to the machine.

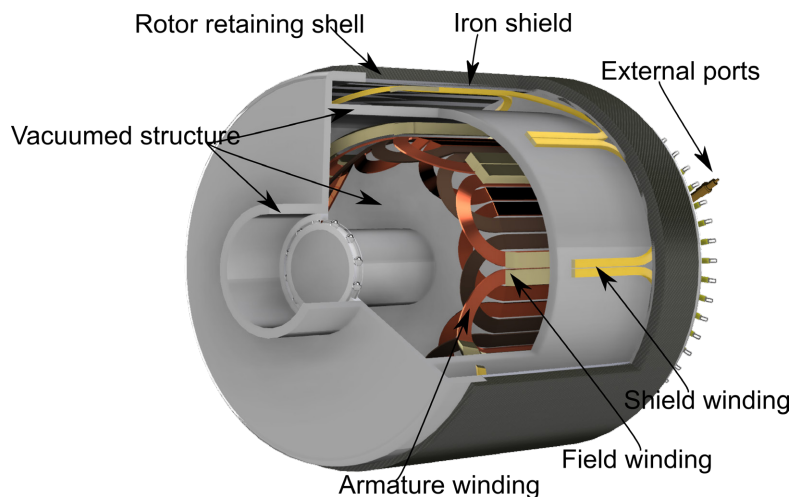


Fig. 89.—Model of an eight-pole machine.

3.1.1.1 Thermal and Mechanical Design

Electric propulsion machines operated at high electrical frequencies create significant losses in the armature coils. These losses are evaluated across a unit length of wire and integrated over the total volume to compute the total loss. An inside-out synchronous motor topology will be adopted for this machine. Field and shield coils are placed on the outer rotor, while armature coils are placed on the stator. This results in more compact volumetric designs. Fig. 89 shows the proposed mechanical design of an optimal motor. The overall machine design is a layered vacuum structure to maximize thermal paths to the windings, and LH₂ coolant will be dispersed to the stator and rotor through couplers connected to a fixed shaft. The LH₂ will be sent through metal pipes, and the heat will be extracted from the winding using conduction cooling. Field and shield excitation will be accomplished using a SC brushless exciter as proposed in Reference 106.

3.1.1.2 Fully Superconducting Machine Optimization Results

A 2.5-MW, 4,500-rpm, fully SC and actively shielded motor is optimized for electric propulsion. A hybrid topology consisting of both active- and passive-shielding methods is explored in this section to demonstrate the tradeoffs. The proposed machine specifications are tabulated in Table 15. Multifilament MgB₂ superconducting cables are chosen for the armature windings due to their low AC-loss characteristics. The same wires are also chosen for the field and shield coils. MgB₂ cable specifications are tabulated in Table 16. The design space considered is tabulated in Table 17. While setting the field and shield-current densities at 200 A/mm², the armature current density is allowed to vary during optimization. However, designs with more than 50% of the critical current density evaluated at the applied field are considered unfeasible, as they exceed the set risk margin.

TABLE 15.—PROPOSED MACHINE SPECIFICATIONS

Specification	Value
Power, MW	2.5
Speed, rpm	4,500
Power density, kW/kg.....	≥ 25
Air gap length, mm	5
OD, m	Max 0.5
Pole number	2, 4, 6, 8, 10, 12
Superconductor	MgB ₂
Operating temperature, K.....	20
Operating phase voltage, V_{RMS}	750
Operating current, A_{RMS}	1111
Armature slots per pole	6
Number of parallel connections	4
Armature current density, RMS, A_{RMS}/mm^2	Max 200
Field current density, A/mm ²	200
Shield current density, A/mm ²	200

TABLE 16.—MgB₂ CONDUCTOR DATA

Parameter design winding	Machine design	Prototype winding
Wire critical current density at 20 K and 2 T, J_c , A/mm ²	402	321.6
SC OD, mm	0.32	0.52
Filament diameter, d_f , mm	0.01	0.024
Number of filaments, n	114	54
SC fill factor, λ	0.15	0.12
Effective fill factor, λ_e	0.49	0.44
Transverse resistivity, ρ , Ω -mm	3.65×10^{-4}	3.65×10^{-4}
Effective transverse resistivity, ρ_e , Ω -mm	1.25×10^{-4}	1.25×10^{-4}
Twist pitch, L , mm	5	10

TABLE 17.—DESIGN SPACE

Parameter	Minimum	Maximum
Armature slot inner radius, X_1 , mm	70	-
Armature slot radial height, X_2 , mm	1	20
Radial distance between field coils and shield coils, X_3 , mm	3	50
Field slot radial height, X_4 , mm	1	20
Field slot circumferential width, X_5 , degree	1	$f(\text{no. of poles})$
Shield slot radial height, X_6 , mm	0.5	10
Shield slot circumferential width, X_7 , mm	0.5	$f(\text{no. of poles})$
Radial distance between shield coil and iron shield, X_8 , mm	0.1	2
Radial iron shield height, X_9 , mm	0.1	30
Armature current density, X_{10} , A/mm ²	50	200

3.1.1.2.1 Propulsor-Machine Co-Optimization

The tradeoff between motor weight and system efficiency was illustrated in this optimization for a constrained problem with fully SC motor topology, fixed motor speed and power, and a fixed propeller geometry. Optimization performed on the fully SC synchronous motor showed a significant tradeoff between motor weight, system efficiency, and motor OD. Fig. 90 demonstrates the degree of coupling between the motor weight and resulting system efficiency. As the machine weight is minimized, the motor diameter tends to increase, thus reducing propeller efficiency by obstructing the propulsive area. This offset gains in motor efficiency obtained by increasing diameter, which in turn decreases the overall system efficiency gains. This can be attributed to a much lower variation in motor efficiency over the design space when compared to the corresponding variation in propeller efficiency. The optimal machine OD for a 1.4-m open propeller is identified as 0.5 m. Two distinct regions of coupling magnitude are visible; the relative efficiency change with slight variations in weight for a 20 kg-class motor is much higher than one in the 50-kg class.

3.1.1.2.2 Open Outer-Diameter Optimization

To understand the impacts of OD on machine efficiency and weight, an open OD optimization is performed. For a 2.5-MW motor, optimization is targeted to minimize the weight and the losses. A detailed weight and thermal model is coupled with multiphysics optimization to capture the OD impacts. Fig. 91(a) shows the OD impacts in generator weight for all the feasible generators. It is evident that larger OD machines increase the weight but reduce the machine losses. Large OD increases the torque production and reduces the machine's electrical loading, reducing the AC losses. As the OD increases, the active-length weight reduces, but end-turn and end-cap weights increase significantly, which results in bulky motors. The results show that optimization converges toward a 0.5-m optimal OD machine for a 2.5-MW electric motor. Fig. 91(b) shows the total loss impacts in winding temperature for all the feasible generators. As the AC losses increase, the winding hotspot temperature also increases. The Pareto-optimal front of the optimization is shown in Fig. 92 with OD. It can be seen that more efficient machines tend to have larger OD and increased weight, as discussed earlier. A significant weight increase can be seen in machines OD above 0.5 m.

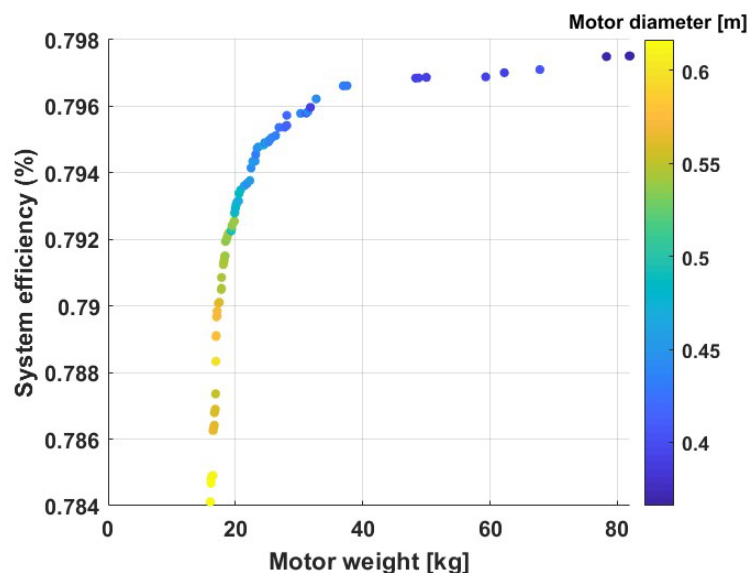


Fig. 90.—Propeller machine co-optimization.

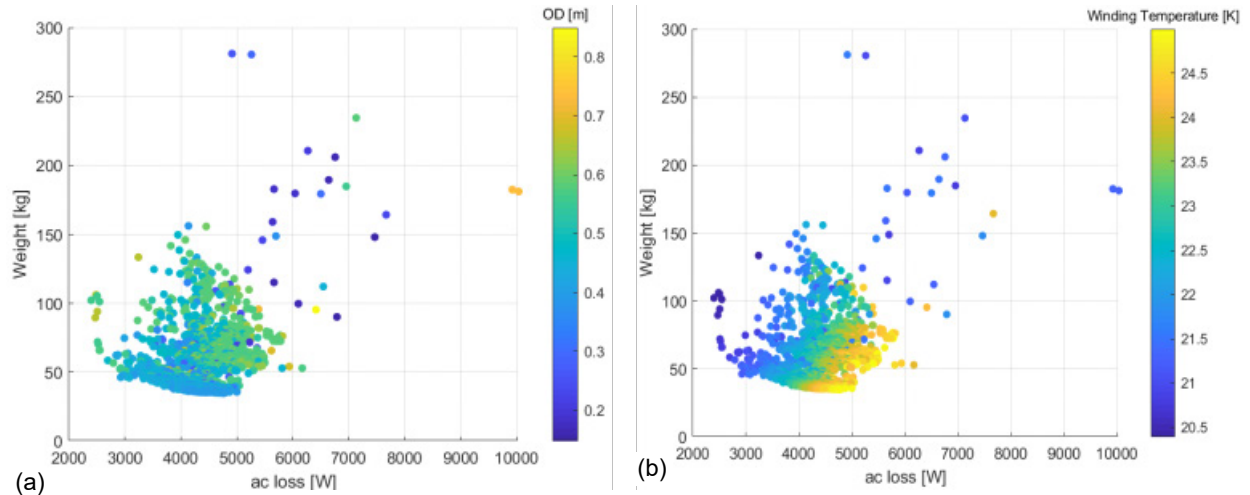


Fig. 91.—All generators weight versus AC losses. (a) OD variation. (b) Hot spot temperature variation.

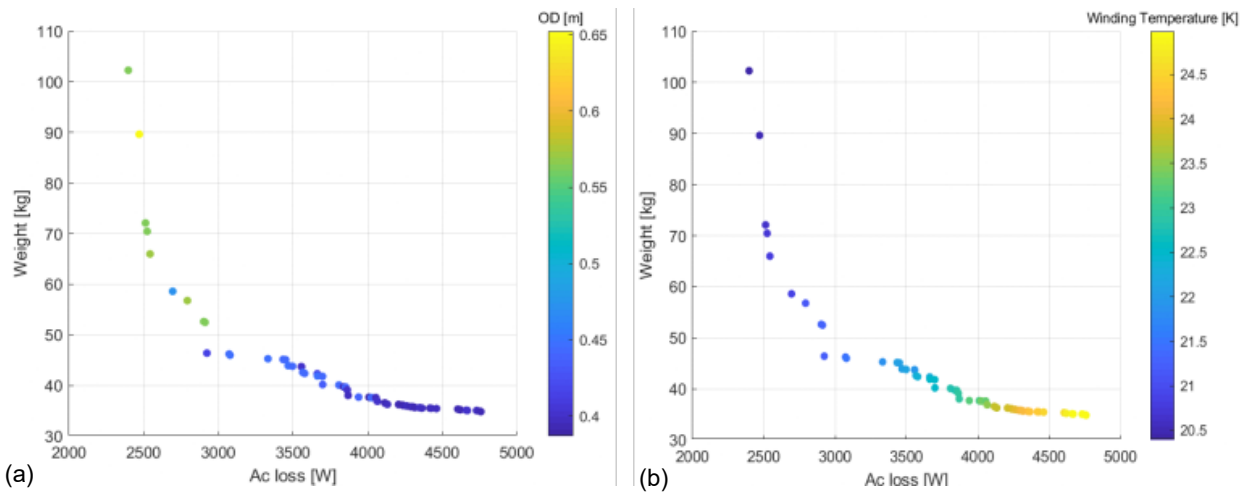


Fig. 92.—Pareto-optimal front weight versus AC losses. (a) OD variation. (b) Hot spot temperature variation.

3.1.1.2.3 Fully SC Machine Only Optimization

A Pareto-optimal front for the entire design space is shown in Fig. 93(a). Every design in the Pareto front is a 2.5-MW machine that satisfies the diameter, axial length, and external flux density constraints. Fig. 93(b) shows a Pareto front of practical designs for the CHEETA concept. All these designs have total losses below 3 kW. The results show a clear tradeoff between weight and AC losses. From the preliminary results, it is evident that pole count significantly impacts weight and AC losses. Higher pole counts result in lower minimum-weight designs; however, the minimum weight is achieved at the expense of higher losses. Eight-pole machines showed the best results within the feasible design space (Table 18). Therefore, eight-pole designs were further explored by opening up the boundaries and increasing the generation number. The resultant Pareto-optimal front of an eight-pole design is shown in Fig. 94(a) and Fig. 94(b). It can be seen that the winding temperature of the low-weight design with higher AC losses ends up above the critical temperature. A feasible Pareto-optimal front is shown in Fig. 95, where all the machines operate below 20 K. Low-AC-loss designs use more iron shielding and are heavier as a result, while minimum-weight designs eliminate the iron shield entirely, as can be seen in Fig. 96(a) and (b). The best designs, which have both optimal weight and losses, also converge toward active shielding alone, shown in Fig. 97.

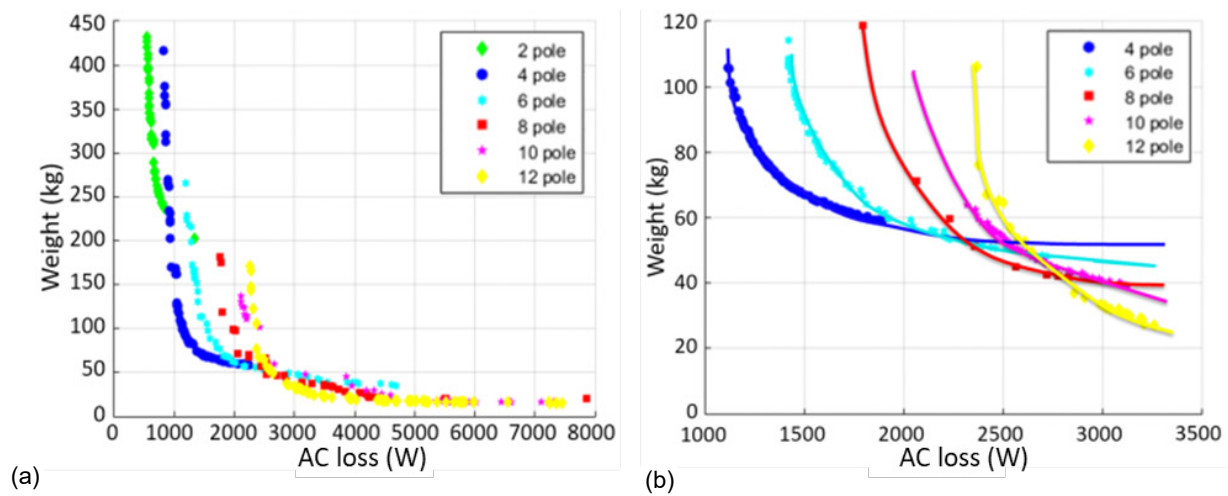


Fig. 93.—(a) Pareto-optimal front for entire design space. (b) Feasible design space.

TABLE 18.—COMPARISON OF MOTORS FOR VARYING DESIGN OBJECTIVE

Parameter	Best weight design	Best AC-loss design	Best overall design
Outer radius, m	0.249	0.249	0.249
Axial length, m	0.215	1.277	0.67
Average torque, Nm	24,708	4,154	7,917
Air gap flux density, T	0.814	0.33	0.464
Outside flux density, mT	0.953	0.001	0.1
Armature SC length, km	10.921	11.554	11.224
Field SC length, km	13.435	23.640	21.186
Shield SC length, km	2.425	3.094	4.337
Total SC length, km	26.782	38.288	36.74
Iron shield weight, kg	0.277	253	0.924
Total loss, W	4,158	1,502	2,255
Weight (iron and SC), kg	18.01	278	25.3
Cost (iron and SC), k\$	53.56	77.08	73.5

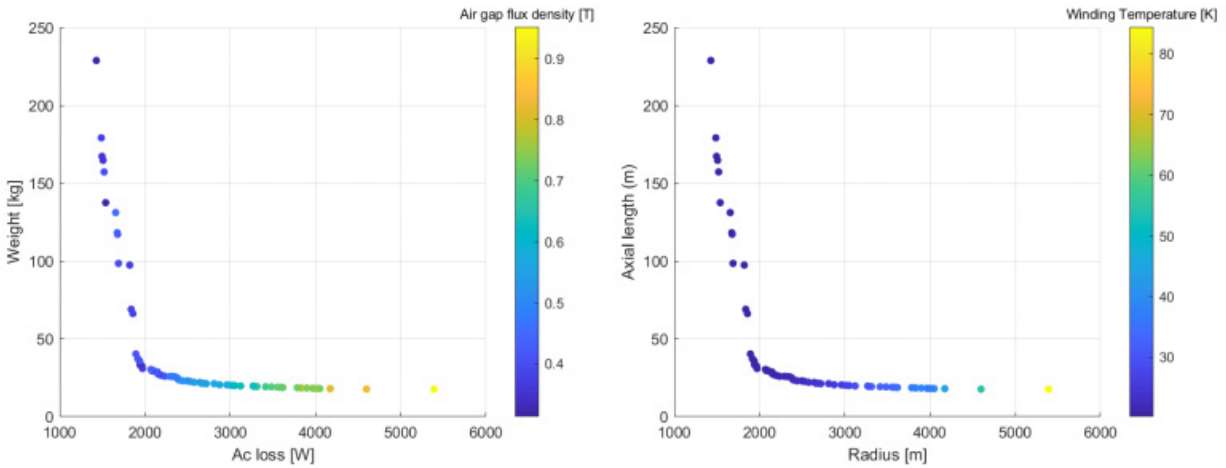


Fig. 94.—Optimal-Pareto front for eight-pole machine. (a) Weight and air gap flux density across range of AC loss. (b) Axial length and winding temperature across varying radius.

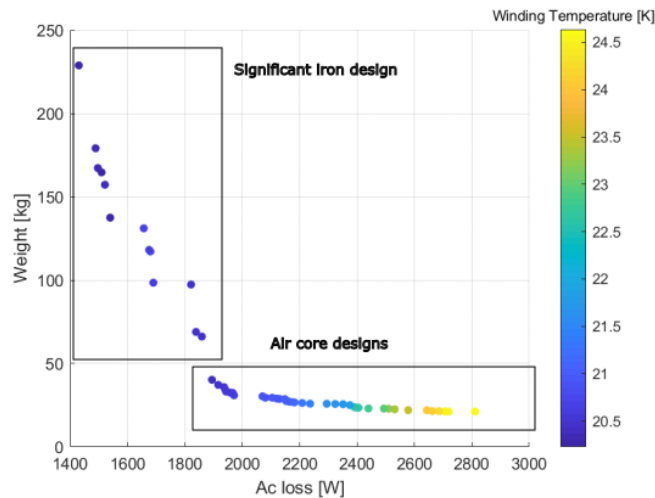


Fig. 95.—Feasible Optimal-Pareto front for eight-pole machine.

Results show that actively shielded designs significantly improve power density compared to their passive iron-shield counterparts. A comparison between different pole counts reveals that eight-pole machines best suit the design constraints. A 25-kg (SC and iron only) motor could be achieved with 2,255 W of AC losses. Assuming another 100% of additional weight for additional components, a 50-kw/kg specific power motor could be achieved. Utilizing the CHEETA concept, a 99.9% efficient electric propulsion motor could be obtained with a fully SC machine. Nearly all feasible designs have air gap flux densities between 0.3 and 0.8 T.

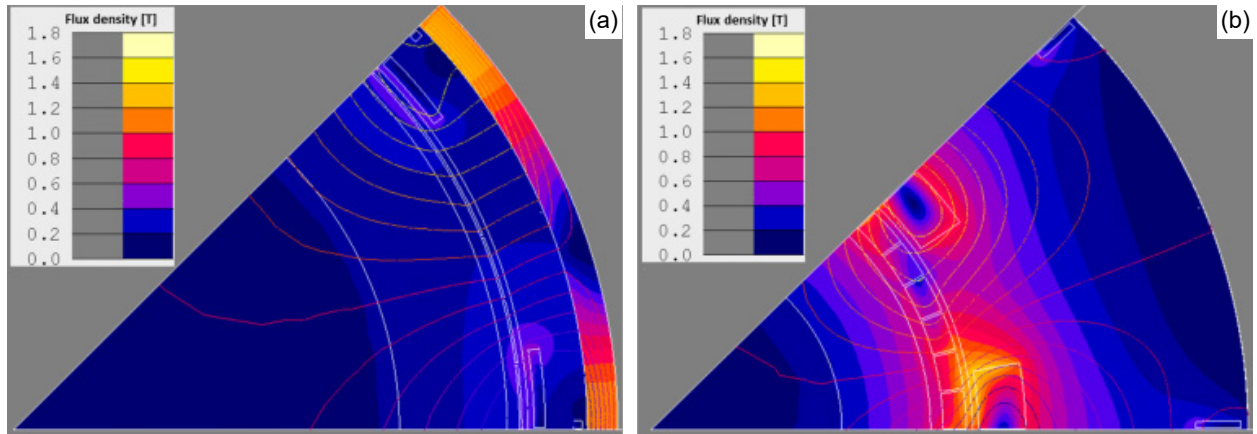


Fig. 96.—(a) Minimum AC-loss design for eight-pole machine. (b) Minimum-weight design for eight-pole machine.

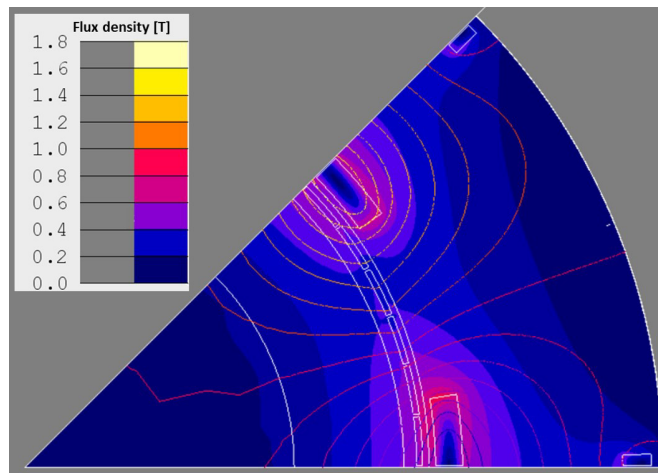


Fig. 97.—Optimized-weight and AC-loss design for eight-pole machine.

3.1.1.3 Alternative Machine Topologies

Alternative cryocooled machine topologies discussed previously are also explored for electric propulsion motors. The following sections summarize a few selected examples, where a fully SC machine is compared to partial SC permanent magnet (PM) and hyperconducting aluminum (hyper-Al) machine architectures. A summary of key outcomes when comparing these alternate machine topologies is provided in Table 19.

3.1.1.3.1 Inner-Rotor Machine-Only Optimization

An inner-rotor, fully superconducting machine was also analyzed and optimized for the same 2.5-MW electric propulsion motor. Results from the fully superconducting outer-rotor machine optimization indicated that reducing magnetic loading is an effective strategy for minimizing AC losses, thereby eliminating the need for shield coils. Based on these findings, an inner-rotor configuration incorporating an iron yoke was analyzed. Fig. 98(a) shows the Pareto-optimal front. Both inner-rotor and outer-rotor optimization converged toward a similar optimal generator. Fig. 98(b) shows the optimal inner-rotor machine air-gap flux density distribution.

TABLE 19.—COMPARISON OF MOTORS WITH
ALTERNATE MACHINE TOPOLOGIES

Parameter	Fully SC	PM SC	Hyper Al
Outer radius, m	0.5	0.5	0.5
Axial length, m	0.67	0.66	0.87
Average torque, Nm	7917	8000	7045
air gap flux density, T	0.46	0.38	0.63
Outside flux density, mT	0.1	0.1	0.1
Total armature SC length, km	11.2	17.8	12.1
Field SC length, km	21.2	0	7.3
Shield SC length, km	4.3	0	0.1
Total SC length, km	36.7	17.8	7.4
Iron shield weight, kg	0.9	1.3	1
Total loss, kW	2.3	3.2	2.7
Weight (iron and SC), kg	25.3	136	13

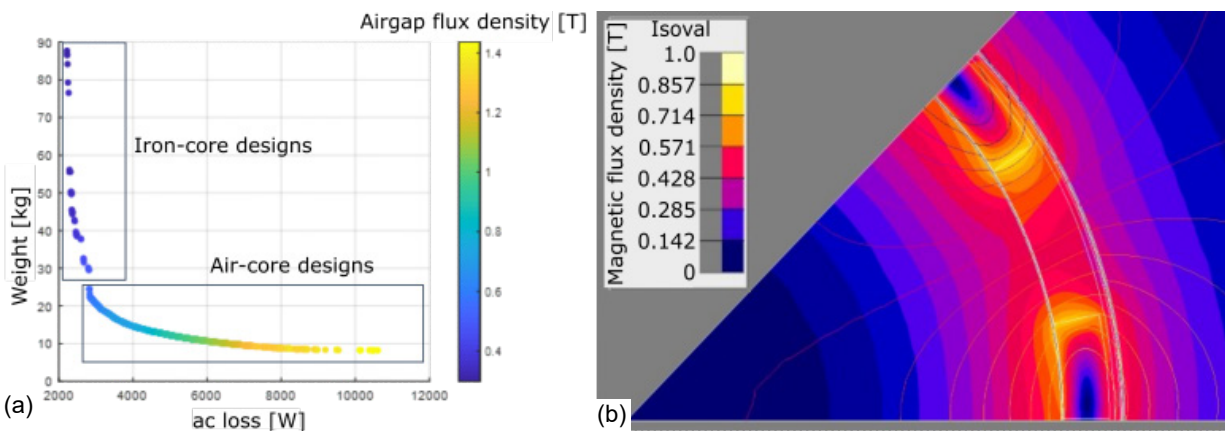


Fig. 98.—(a) Inner-rotor, fully SC Pareto-optimal front. (b) Optimal inner-rotor, fully SC machine.

3.1.1.3.2 Partial SC PM Machine-Only Optimization

Fully SC machine optimization showed that the optimal air-gap flux density is around 0.5 T, and a PM rotor was considered to replace the field winding. A warm PM rotor eliminates the need to cool the rotor, thus significantly improving the machine's reliability. Fig. 99 shows the Pareto-optimal front for weight and AC losses. The PM motors are significantly heavier compared to the fully SC motors. The majority of this weight comes from the bulky magnets in the rotor. Low-loss designs increase the air gap field at the expense of magnet weights. An SC armature winding with a PM rotor does not provide a significant advantage in specific weight compared to a conventional PM motor. This is due to the more substantial air gap adapted to a fully SC motor to employ a vacuum and radiation shield in the air gap. A fully SC motor design has a 3 times larger air gap than a conventional PM motor, significantly reducing the machine's magnetic loading.

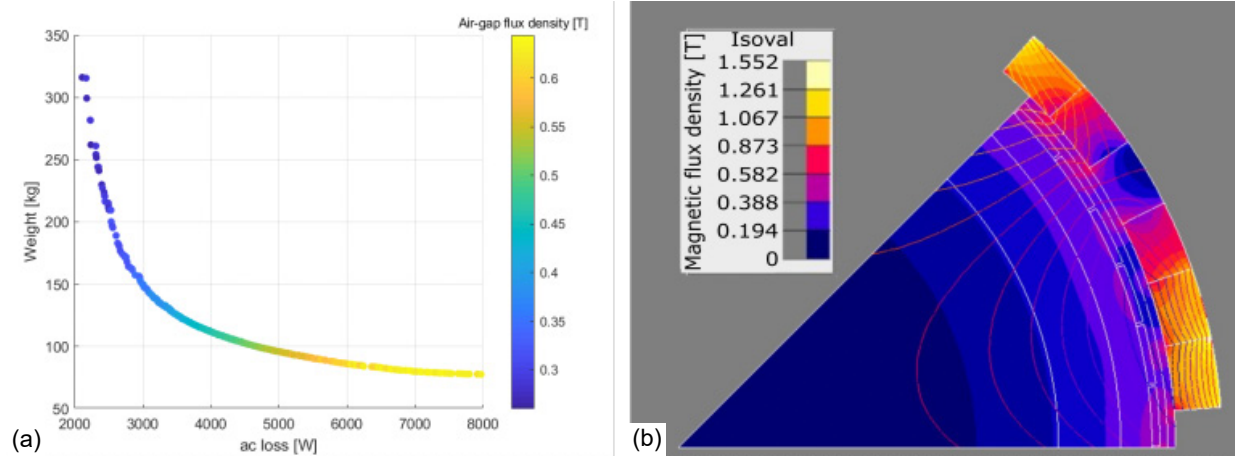


Fig. 99.—(a) PM machine Pareto-optimal front. (b) Optimal PM SC machine.

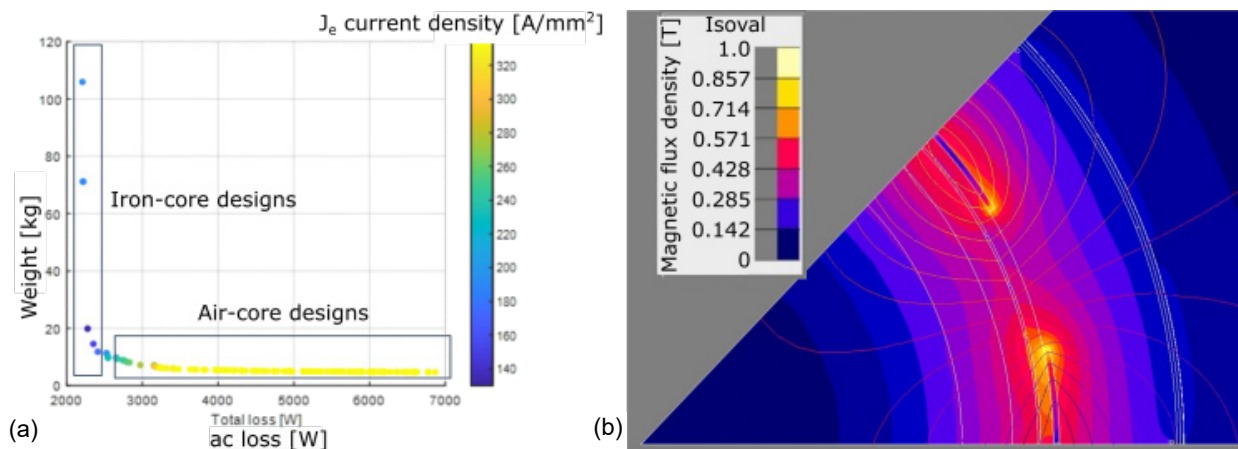


Fig. 100.—(a) Hyper-Al Pareto-optimal front. (b) Best Hyper-Al EM design.

3.1.1.3.3 Cryocooled Hyper-Al Machine-Only Optimization

In this machine topology, hyper-Al conductors are used as armature windings. Fig. 100 shows the Pareto-optimal front for weight and AC losses. Similar MgB_2 conductor construction with a residual-resistivity ratio (RRR) of 900 is assumed for multifilamentary hyper-Al conductor. Compared to MgB_2 SC, hyper-Al conductors do not have hysteresis losses and transport current losses. However, they generate ohmic losses quadratic to the operating current. Eddy current and coupling losses are generated similarly to SC wire. Since there are no quench risks for hyper Al conductors, the 50% critical current density margin limit is removed from the constraints. This allows the armature conductors to carry high currents at the expense of ohmic losses. Fully SC motor geometrical design space tabulated in Table 17 is used for optimization.

3.1.1.4 Thermal Design

The thermal design comprises separate cooling schemes for the stator and rotor. This section presents these cooling schemes and discusses justifications and tradeoffs.

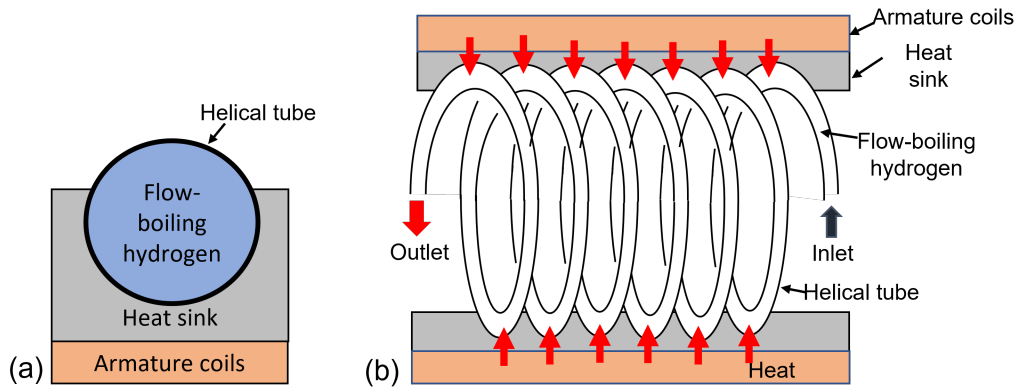


Fig. 101.—Helical tube HEX. (a) Embedded tube concept. (b) Topology concept (side view).

3.1.1.4.1 Stator Cooling

The heat produced by the stator is transferred to the H_2 through a HEX, illustrated in Fig. 101. This topology has the windings bonded to a thermally conductive heat sink. Embedded in the heat sink are helically coiled tubes through which the LH_2 flows. There is a heat conduction path from the windings to the tubes and a convection path from the tubes to the LH_2 . The helical tube curvature serves two purposes. First, the fluid's inertial resistance to changes in the flow path force the fluid to the curvature's outer edge, thereby keeping wet the edge closest to the conductors. This prevents the outer edge from drying out, which would worsen the heat transfer at this edge. Second, the helical curvature induces flow vorticity, again resulting from the fluid's inertial resistance to changes in flow path. Vorticity is known to enhance heat transfer in single-phase and two-phase fluids, which makes it desired in HEX applications (Refs. 107 to 110).

We undertook a design study to assess the feasibility of this cooling scheme and to determine appropriate design parameters. We constructed a 1D analytical thermal model of the HEX to evaluate the maximum coil temperature, HEX mass, and motor specific power for a given HEX design. The LH_2 coolant is taken to be at 20 K, and its boiling point at 1 atm. Between the coolant and the coil, a temperature difference of 3 K is assumed to account for coolant convection resistance, component conduction resistances, and contact resistances. The coldest side of the coil is then 23 K, meaning that the temperature rise to the warmest side of the coil, denoted as ΔT_{coil} , must be lower than 2 K to maintain the motor below its specified operating temperature of 25 K. To calculate ΔT_{coil} , the heat flux is multiplied by the overall coil thermal resistance in the transverse direction, denoted as r_{coil} . This impedance can be approximated as a series circuit using the equation of mixture:

$$r_{coil} = \frac{\eta_{wire}}{k_{wire}} + \frac{\eta_{ins}}{k_{ins}} + \frac{\eta_{imp}}{k_{imp}} \quad (45)$$

where k is the material thermal conductivity and η is the material volume fraction for the wire, insulation, and impregnant. In this analysis, the volume fraction of the impregnant and the wire is each 0.5, with the volume fraction of insulation taken to be negligible. The thermal conductivity may also depend on the arrangement of wire strands in the coil cross-section, though this effect has been neglected for the sake of simplicity.

The transverse thermal conductivity of the wire itself can be modeled by a series parallel network. The wire in this example is composed of approximately 15% MgB_2 , 33% Nb, and 51% Cu-30Ni. In the thermal circuit, the Cu-Ni impedance is configured in parallel with a series impedance of the Nb and MgB_2 . Because the thermal impedance of Cu-30 is so high compared to MgB_2 , this thermal network may

be approximated as that of pure Cu-30Ni. An empirical measurement of the Cu-30Ni thermal conductivity at 20 K could not be found, so it is approximated from measurements of other compounds. J.K. Hulm (Ref. 111) measured the thermal conductivity of Cu-20Ni to be 12 W/mK. Reducing that to account for the increased Ni content in Cu-30Ni, we get an estimate of 8 W/mK for Cu-20Ni. J.G. Hust (Ref. 112) measured the thermal conductivity of Monel, a Ni-heavy Cu-Ni alloy, to be 6 W/mK at 20 K. The average of these two values, 7 W/mK, is then selected for k_{wire} in Eq. (56).

The convective HTC was estimated using the Dittus-Boelter correlation:

$$h'_L = 0.023 \frac{k_L}{D} \left(\frac{DG}{\mu_L} \right)^{0.8} \left(\frac{C_{pL}\mu_L}{k_L} \right)^{0.4} \quad (46)$$

The Dittus-Boelter correlation was shown by Shirai, et. al (Ref. 68) to be somewhat accurate in predicting the HTC for flow-boiling H₂ in straight tubes.

Using the analytical thermal model, we then evaluated many designs with varied values for the number of parallel tubes, tube diameter, and the embed fraction. The number of parallel tubes is the number of parallel flow paths through which the H₂ flows splits. The tube diameter is the OD of the flow paths. The embed fraction is the fraction of the tube that is embedded in the heat sink. The results of the optimization are shown in Fig. 102. We selected the HEX design with the highest overall motor specific power. The parameters for this design are shown in Table 20.

The optimization results show that a feasible cooling system could be created to meet temperature and specific power targets for this motor. However, our estimate of temperature, and hence of machine specific power, are subject to great uncertainty. This uncertainty arises from the model used to estimate convective HTC in the coolant tubes. The Dittus Boelter correlation, which was used for estimating the HTC, does not account for vorticity resulting from the tube curvature. The effects of vorticity caused by curved tubes on flow-boiling cryogenic heat transfer have never been experimentally studied; however, we anticipate some enhancement effect similar to that of noncryogenic fluids. To reliably model our cooling system, an experimental study must be conducted to close this knowledge gap.

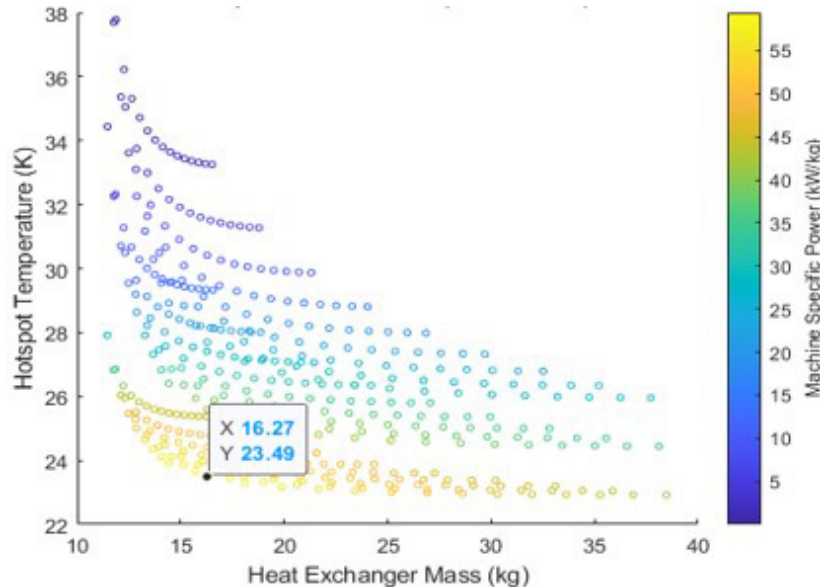


Fig. 102.—Results of optimization for three HEX parameters: number of parallel tubes, tube diameter, and embed fraction. The optimal design is indicated.

TABLE 20.—OPTIMAL STATOR
HEAT EXCHANGER DESIGN

Parameter	Value
Number of parallel tubes	1
Tube diameter, in.....	3/4
Embed fraction	2/5
HEX mass, kg.....	21.1 kg
Maximum coil temperature, K	24.2 K
Machine specific power, kW/kg.....	47.9
Fluid pressure drop, psi	0.88

We planned a series of experiments to establish a more accurate model for predicting HTC of flow-boiling cryogenic fluids in coiled tubes. These experiments boiled LN₂ in three different coiled tube configurations and measured the convective HTC. Three configurations were tested to evaluate the effect of tube curvature and coil curvature on the HTC. LN₂, rather than H₂, was used due to N₂'s greater ease of availability.

A diagram of the experimental setup is shown in Fig. 103(a). The LN₂ flow is driven by a pressure difference between two tanks: a pressurized supply tank and a nonpressurized return tank. A preheat section comprises heated tubes that partially vaporize the fluid to a desired vapor fraction before the fluid enters the test section. The test section comprises a heated coiled tube with resistance temperature detectors mounted to several points on the tube's outer surface. The heat inputs to both the preheat and test sections are controlled by separate variable power supplies, and the heat is delivered via a resistive nichrome wire wrapped around the tubes. The fluid flow rate is measured by a flow meter, and the fluid inlet temperature is measured by a temperature sensor immersed directly in the fluid flow. Pressure sensors measure the absolute pressure at the inlet of the test section and the differential pressure across the length of the test section. Combined, all these measurements provide the data necessary to calculate the HTC with any flow or heat input condition tested.

The experimental setup was constructed and is shown in Fig. 103(b). The convective HTC was measured at a range of mass flux, inlet vapor fraction, and heat flux. Additionally, the experiment tested three coiled tube configuration and two straight tube configurations. The data were then used to select an appropriate model for predicting HTC in flow-boiling cryogenic fluids. This model will account for the effect of tube curvature and tube diameter.

To validate our method of HTC measurement, we used the setup to measure the HTC of single-phase water convection heat transfer. The HTC was measured at two locations for a range of flow rates. The results are shown in Fig. 103(d). The measured HTC values agreed well with HTC values predicted by the well-established Gnielinski correlation for single-phase flow. This test validates the setup for use in measuring the HTC of flow-boiling N₂.

Another major source of modeling uncertainty is the compound used for potting the motor's magnetic coils. The thermal resistance of the potting compound typically dominates the overall coil thermal resistance, so the choice of potting compound greatly affects the coil temperature and motor specific power. Epoxy resins, while traditionally used for potting motor coils, have insufficient thermal conductivity for use in the CHEETA motor. A thorough search of epoxy resin thermal conductivity measurements at LH₂ temperatures found the most conductive epoxy to be a silver-filled epoxy with a conductivity value of 0.8 W/mK (Ref. 113). Our 1D analytical thermal analysis determined that to limit the temperature difference across the coil to 2 K, the potting compound must have a conductivity of at least 1.1 K. A maximum temperature difference of 2 K was selected to maintain the coil temperature

below 25 K, assuming a 3 K temperature difference from the coolant fuel to conduction and convection thermal resistances.

We conducted a literature review of several other candidate potting compound materials. A summary of our findings is found in Table 21. The most suitable replacement for epoxy resins is ionic solids. These materials have thermal conductivity values that are orders of magnitude higher than epoxy resins. Ionic solids also have low melt viscosity and low thermal contraction. These qualities are favorable for impregnating coils. Drawbacks include little to no industry experience and high melt temperature.

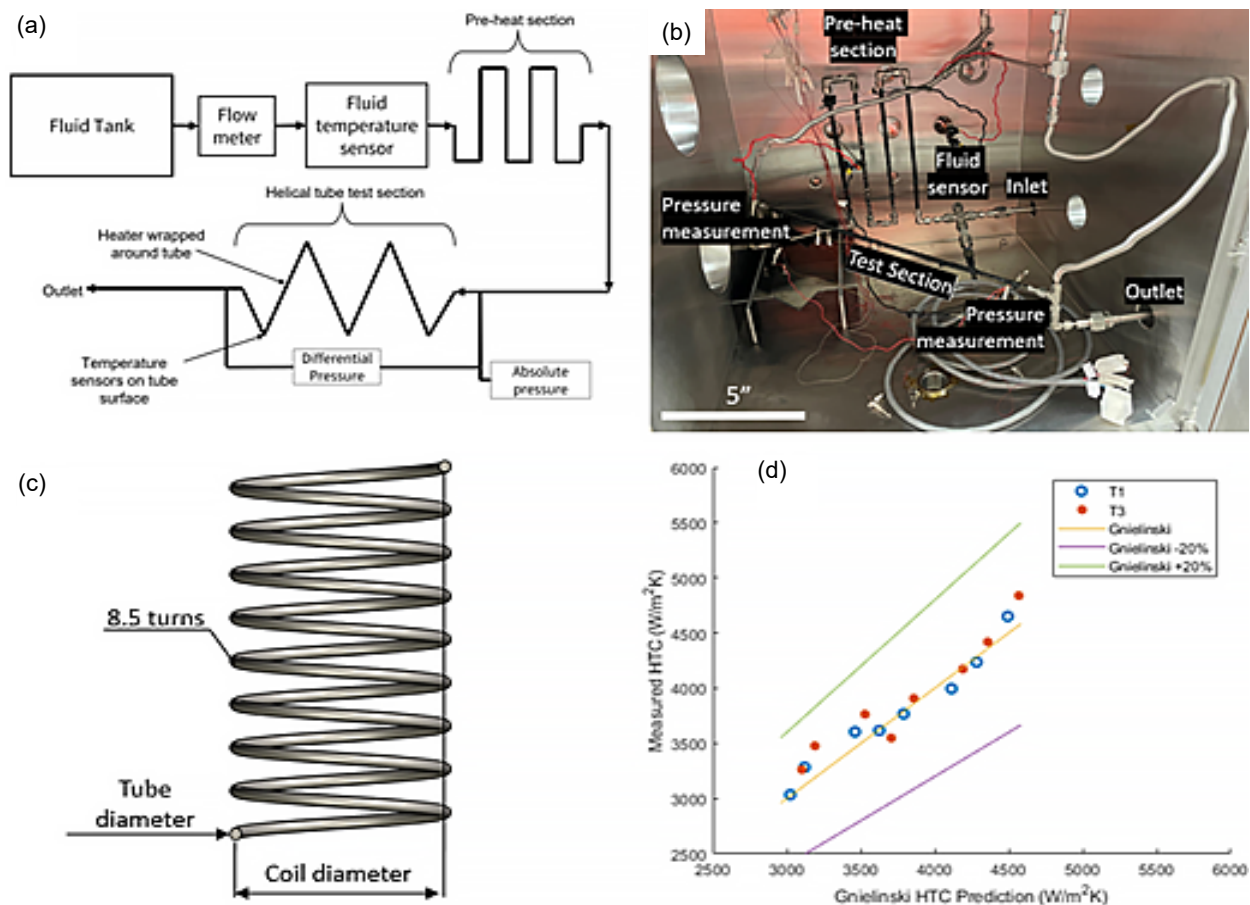


Fig. 103.—(a) Experimental setup used to measure convective HTC of flow-boiling LN₂ in coiled tubes. (b) Experimental setup used for measuring HTC in straight tubes. (c) Helical tube geometry. (d) Results of water validation test; colored symbols indicate HTC calculation for a given flow condition; colored lines indicate range where experimentally measure HTC agrees with Gnielinski correlation prediction within 20%.

TABLE 21.—SUMMARY OF ADVANTAGES AND DISADVANTAGES OF VARIOUS IMPREGNANTS

Impregnant	Advantages	Disadvantages
Epoxy resin	• Extensive industry experience • High mechanical strength	• Poor radiation resistance • High thermal contraction • Low thermal conductivity
Cyanate ester resin	• High radiation resistance • Low viscosity • High thermal diffusivity • Long usable life	• High cost • Low thermal conductivity • Little experience
Wax	• Extensive industry experience • Low bonding strength • Low resistance to cracking • May be able to unimpregnate and reuse the conductor	• Weak mechanical strength • Low thermal conductivity • High thermal contraction • Can melt and drain in warm environments • Large volume reduction upon solidification
Water ice	• Easy and economical impregnation • Decent thermal conductivity • No conductor degradation observed • Can unimpregnate and reimpregnate coils • Low viscosity	• Weaker mechanical properties than metal or epoxy • Requires proper containment of water • High tendency to degas • Little experience
Metal	• Very high thermal conductivity • Low melting point metals available • High mechanical strength	• Eddy-current generation • Some thermal degradation of coils observed • Difficulty of filling voids • Little experience
Ionic solids	• High thermal conductivity • Low thermal contraction • Low viscosity	• High impregnation temperature • Little experience
Polyimides/ bismaleimides	• Decent thermal conductivity • Low thermal contraction • High stiffness • High dielectric strength • High radiation resistance	• High viscosity • Short pot-life • Little to no experience
Cryogenic solids	• Excellent penetration properties • Can unimpregnate and reimpregnate coils • High thermal conductivity • Provide stability during loss of cooling	• Slow normal-zone propagation • Requires proper containment of liquid cryogens • Potential for thermal dry-out • Little experience

3.1.1.4.2 Rotor Cooling

We decided not to cool the rotor with LH₂ due to the safety hazards involved with the use of a rotary transfer coupling. Rotary transfer couplings are typically used to couple a rotating coolant loop with a stationary coolant supply. In SC machines, these couplings typically transfer liquid helium, such as in References 114 and 115. The couplings involve many complex and moving parts that present opportunities for leaks, especially for fluids like H₂ that have a known propensity for leaking. Since leaked H₂ could combine with air and combust, this cooling approach is unsuitable for an aircraft application.

We chose instead to conductively cool the rotor using a cryocooler. In this approach, a thermal conduction path exists between the rotor's field and shielding coils and the cold head of a cryocooler. At the same time, heat leak to the rotor is minimized by suspending the rotor via thermally insulating spokes. The cryocooler would spin synchronously with the rotor. While this approach has been successfully demonstrated for low-speed applications like wind turbines (Ref. 116), it has not been demonstrated for high-speed applications like this one.

We validated this cooling approach by testing a cryocooler that was rotating at high speeds. A photograph of our test setup is shown in Fig. 104. In this experiment, we rotated a Sunpower GT/GTLT cryocooler at speeds ranging from 0 to 1,000 rpm. The cryocooler was operated under a no-load condition as it was rotated. The temperature at the cold head of the cryocooler was measured at several speeds as the rotation was increased to 1,000 rpm and then decreased to 0 rpm.

The results are plotted in Fig. 105. The results show that rotational speeds up to 1,000 rpm do not adversely affect the cooling capacity of the cryocooler. Thus, our rotor cooling scheme is validated for speeds up to 1,000 rpm. Future tests will include higher speeds. Additional details of this test and of the rotor cooling scheme can be found in Reference 117.

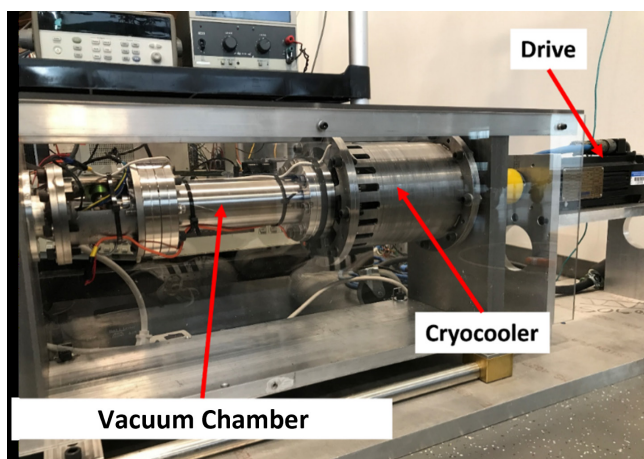


Fig. 104.—Rotating cryocooler test bench. Speeds ranged from 0 to 1,000 rpm.

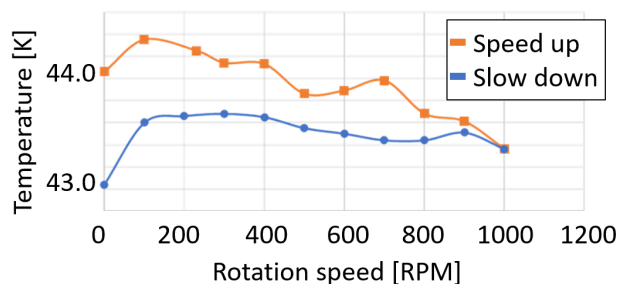


Fig. 105.—Results of rotating cryocooler test. Cooling performance is unaffected by rotational speed.

3.1.2 AC Loss Model Validation

Experimentally measuring AC losses is challenging due to the shallow losses in SC wires on the order of mW. Several successful AC-loss measurements are reported in SC wires and cables under a stationary alternating field environment (Refs. 118 to 120). These can be measured through the electromagnetic measurements (Refs. 121 and 122), the electric-heater-based calorimetric method (Ref. 123), or the cryogen boiloff-rate-based calorimetric method (Ref. 124). Losses must be measured in the transverse rotating field to validate the loss estimation in electrical machines. Adding a rotating field to the loss measurements increases the complexity of the measurement setup and subsequent loss measurements. A rotating electrical field inside a cryogenic environment can be created by a PM rotor (Refs. 125 and 126), specially arranged alternating field coils (Ref. 127), or by rotating a sample in a uniform field (Ref. 128). Each method has limitations in the applied field level, frequency, and achievable sample temperature. Measurement capabilities of most of the available test benches and the UIUC AC-loss measurement setup are tabulated in Table 22.

Fig. 106 compares the peak measurement capability of available AC loss measurement test benches to UIUC. The experimental setup discussed in this section is used to experimentally validate the AC losses in MgB_2 windings for electric propulsion and wind turbine application. Fully SC electric-propulsion machines operate at the 0.4- to 0.7-T air-gap field range, and higher electrical frequencies measure 0 to 300 Hz. Wind turbines operate at the 0.5- to 1-T air-gap field range and at low frequencies of less than 10 Hz. No measured loss data is available for MgB_2 conductors in these electrical machine application spaces.

A two-pole rare-earth PM rotor is used to create a high magnetic field in the air gap. Fig. 107 illustrates the flux density variation inside the PM rotor air gap. Fig. 108 shows the flux density variation along the air-gap radius. It can be seen that the air gap has a uniform magnetic field of up to 90% of the air gap. Samples are placed in the air gap. A rotating field is applied by mechanically rotating the rotor. The applied electrical frequency is changed by varying the rotating speed. The heater-based calorimetric method is used to measure the losses in the MgB_2 sample conductors. Fig. 109 shows the cross-section of the measurement setup. The PM rotor and the sample are placed inside a stationary ultralow vacuum chamber to eliminate the convection losses. The radiation shield, magnetic shield, and windage loss shield are placed inside the AC loss setup to minimize the error. The radiation shield is thermally grounded to the second stage of the cryocooler, and 15 layers of multilayer insulation (MLI) are added outside the shield to reduce radiation losses further. A rotary coupling is used to rotate the magnets inside the chamber. Samples are attached to a cryocooler to maintain them at the required temperature. A picture of the measurement test bench in the laboratory is shown in Fig. 110.

TABLE 22.—ROTATING AC-LOSS MEASUREMENT CAPABILITIES

Reference	Field, T	Frequency, Hz	Temperature, K	Cooling method
UIUC test bench	1	160	20	Cryocooler
PM rotor (Ref. 125)	0.5	40	77	LN_2
Alternating field (Ref. 127)	0.02	90	77	LN_2
PM rotor (Ref. 128)	0.35	60	77	LN_2

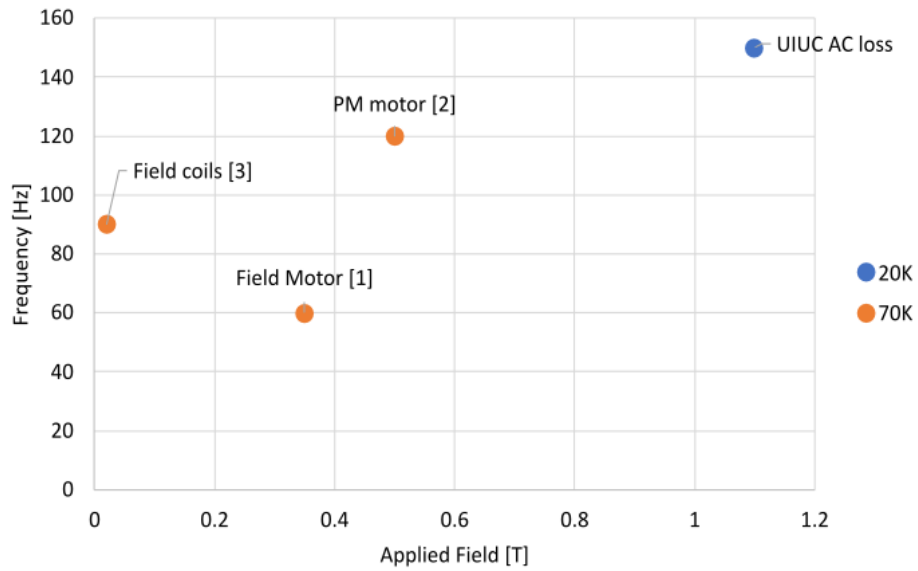


Fig. 106.—AC-loss measurement test setup.

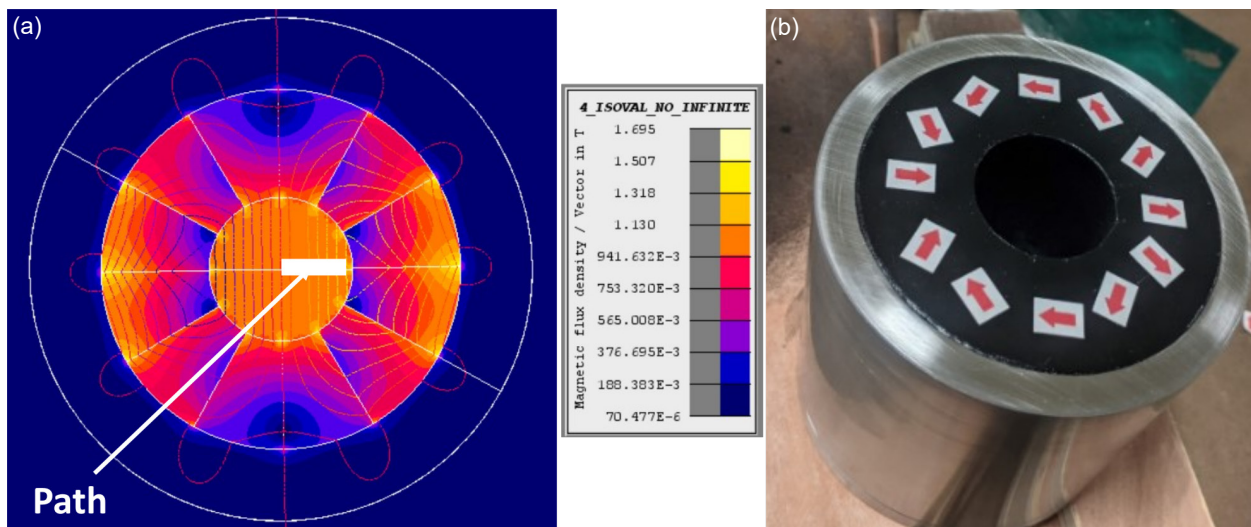


Fig. 107.—Flux density variation in the air gap. (b) Fabricated magnet rotor.

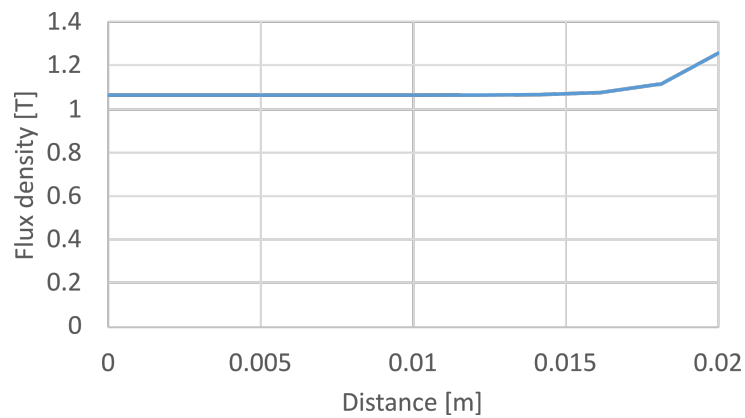


Fig. 108.—Flux-density variation along the radius.

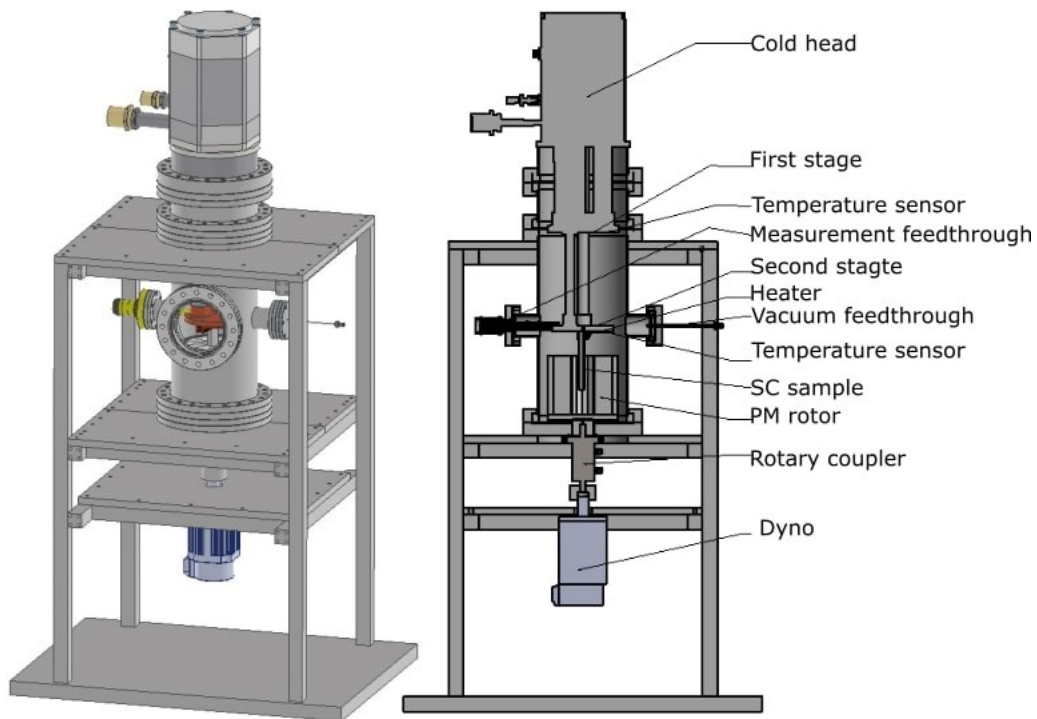


Fig. 109.—AC-loss measurement test setup.

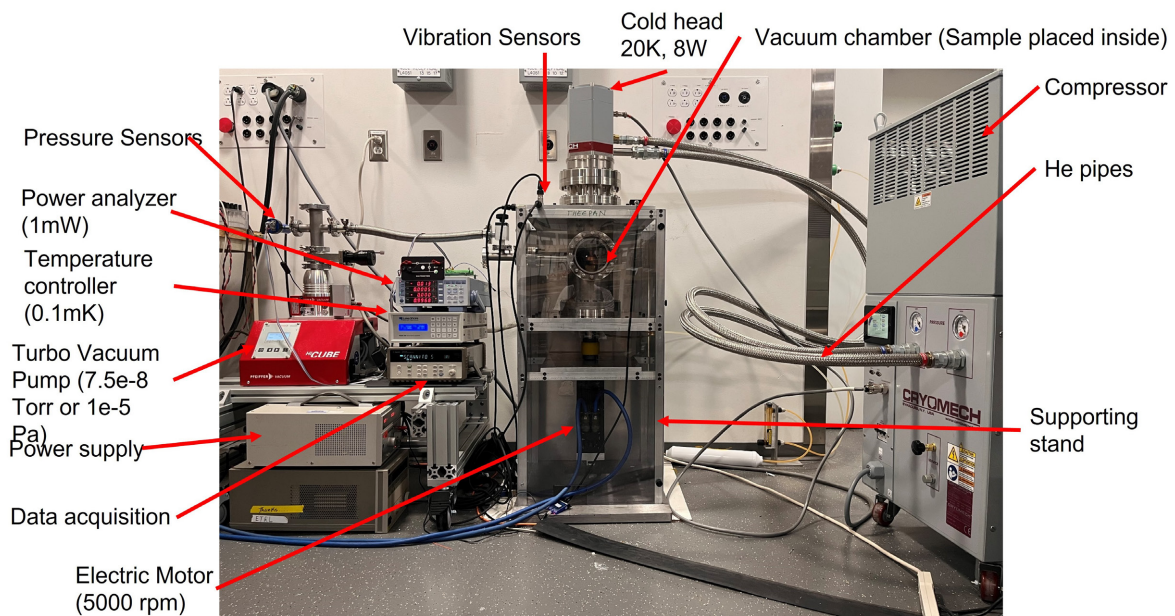


Fig. 110.—AC-loss measurement test setup.

3.1.2.1.1 Superconducting Samples

The react and wind method is utilized to make SC MgB_2 samples. Sample wires are first cut into lengths of 45 cm and attached to an SS316 steel plate as shown in Fig. 111(a). The wires are heat treated and cut into 6-cm pieces to make a sample. The cut SC samples are then attached to a ceramic sheet, as illustrated in Fig. 111(b). A ceramic holder is used instead of a metal holder to eliminate eddy current losses in the holder while providing excellent heat conduction to the SC samples. Aluminum nitride (AlN) and boron nitride (BN) ceramic sheets are used as sample holders.

Finally, samples are impregnated using a vacuum resin-pull method to get a good resin fill and adequate thermal conductivity (Fig. 111(c)). Fig. 112 explains the vacuum resin-pull method, where the SC sample is placed inside the mold and resin is filled inside the mold against gravity using a vacuum pump. This reduces the entrapped air bubbles inside the sample and provides better thermal conduction between the SC wires and the ceramic holder. Fig. 113 gives a final MgB_2 sample used to measure AC losses.

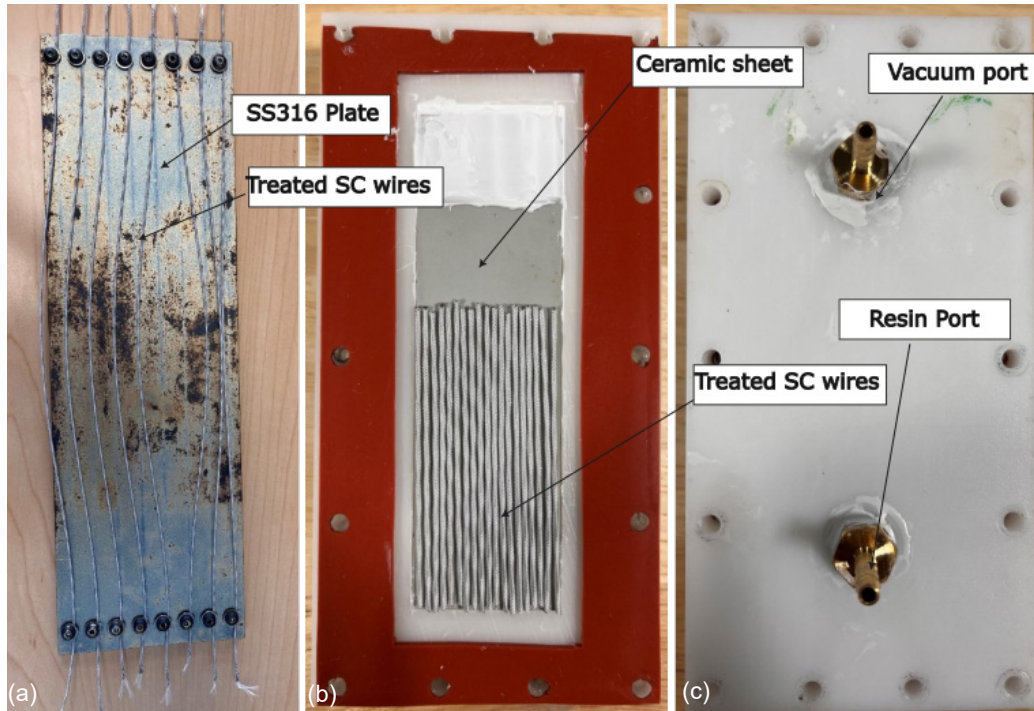


Fig. 111.—SC MgB_2 samples preparation. (a) Wires attached to SS316 steel plate for heat treatment. (b) Heat treated 6-cm pieces attached to ceramic sheet. (c) Sample impregnated via vacuum resin-pull method.

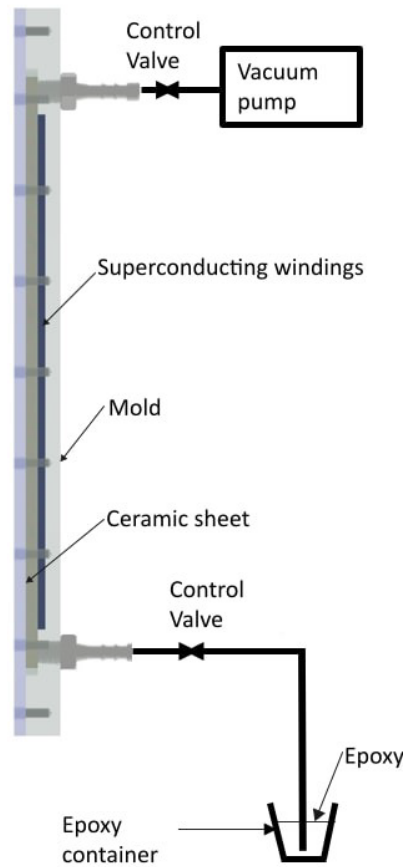


Fig. 112.—Vacuum resin-pull method for making samples.



Fig. 113.—SC MgB_2 samples.

3.1.2.1.2 Calorimetric Method

The heater-based calorimetric method is employed to measure the generated AC losses. Fig. 114 presents the cold head arrangement inside the vacuum chamber. A cryocooler is operated in a constant power mode to cool the sample. A heater is attached to the sample to control the heat load on the cold head and the cold head temperature. A temperature controller is used to control the heater. Fig. 115 shows the proportional–integral–derivative (PID) controller implemented to maintain a constant sample temperature. No losses are generated in the wire at stable conditions and idle operation. The heat load on the cryocooler is equal to the heat load put into the heater and the system thermal losses. When the rotating field is administered, the external disturbance is applied to the system in the form of AC losses. The heat load will increase due to the additional AC losses, and, because of that, the cold head temperature will increase. The PID controller maintains a constant temperature by reducing the heat dissipated in the heater. This reduction is equivalent to the AC losses generated in the sample.

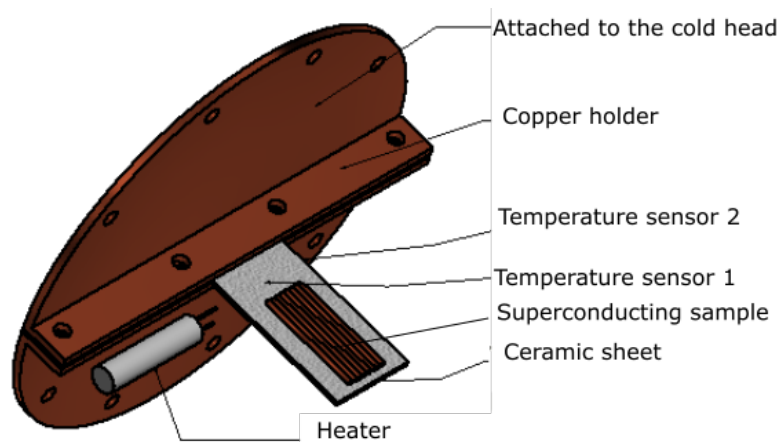


Fig. 114.—Calorimetric arrangement inside the vacuum chamber.

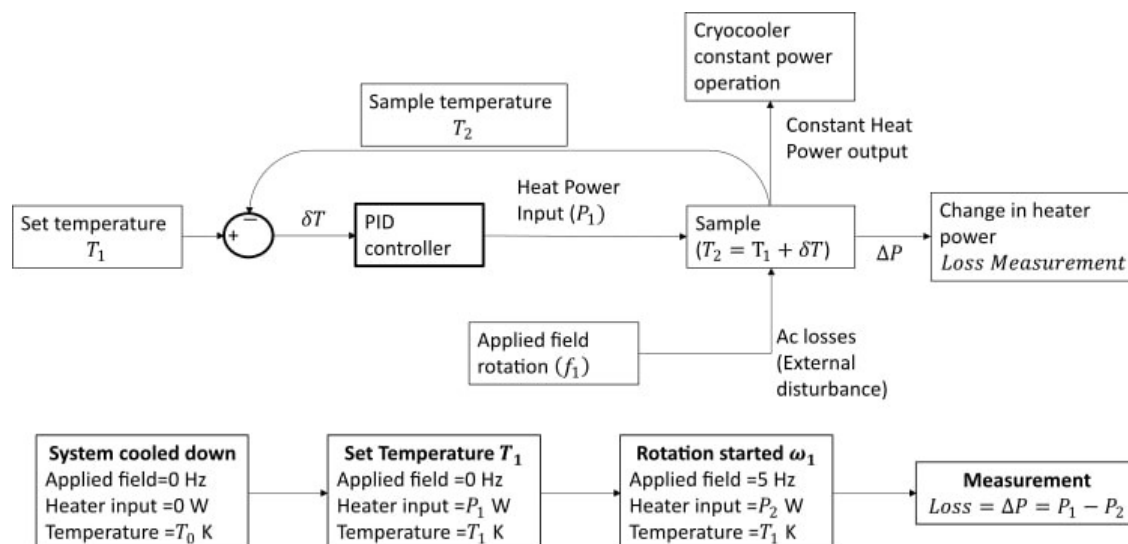


Fig. 115.—Control flow chart.

3.1.2.1.3 Calibration and Measurements

First, rotating fields are generated inside the vacuum chamber without attaching samples to the cold head, and the temperature is monitored. This guarantees that no additional losses are generated inside the chamber due to the mechanical movements. The following tests are performed to calibrate the setup before measuring AC losses in SC samples.

- Epoxy sample: Ceramic holder with an epoxy layer is used to estimate the windage losses in the system
- Copper wires: Copper wire with the same OD is used to validate the measurement technique
- Untreated SC wires: Estimate the eddy current losses and validate the heat treatment process
- Treated SC wires: Measuring AC losses

The measurement setup is calibrated to measure up to 10 mW of losses at 1-mK accuracy. Epoxy samples, copper wires, and untreated SC wires did not provide significant losses at low frequencies. Therefore, a special sample shown in Fig. 116 is made for calibrating the setup under low-frequency AC loss measurements. This sample is made with the same ceramic sheet used to make the SC samples. As shown in Fig. 116, nichrome wire is wrapped around the ceramic sheet, and an additional temperature sensor is attached to the tip of the sample. Fig. 117 shows the sample temperature and cold head temperature during the initial cooling down. This shows that the sample temperature and the cold head temperature closely match. The nichrome heater is powered by another power supply to heat the sample. Power input to the sample and the sample temperature are measured. Heat input is increased in 10-mW steps and then measured using the AC loss measurement setup. Fig. 118 shows the calibration procedure. It can be seen that, as the heater input is increased in 10-mW steps, the sample temperature also increases, but the cold head temperature is maintained at 19-K set temperature. It can be also seen the heater power input is increased while the control heater power input is decreased. This can be further demonstrated in Fig. 119. Losses are measured from when step heat is applied to when the first steady-state temperature is achieved. Fig. 120 shows the input heat versus measured heat with zoomed-in cold head temperature measurements. Cold head temperature is maintained within 2-mK accuracy while heat input is increased from 0 W to 100 mW in 10-mW steps. A box plot of the measurement is shown in Fig. 121. Calibration showed that the losses could be measured with 5% accuracy. Error percentage based on the mean measured loss is shown in Fig. 122. The mean error was 1.39%, and the median error was 0.52% with one outlier of 10%. This shows that the measurement setup has good accuracy to measure AC losses at low frequencies where total losses are under 100 mW.

Fig. 123 presents the calorimetric measurement conducted at 0.5-Hz applied frequency. Initially, the sample is maintained at 20 K and the rotating field is applied at 19 Ks. Increasing sample temperature indicates additional AC losses generated in the sample. The PID controller decreases the heater input power to maintain the sample temperature at 20 K. This heat reduction is used to estimate the AC losses in the sample. Wind turbines operate at a low-frequency range. For example, a 10-rpm wind turbine generator with a 10 to 60 pole count will have electrical frequencies varying from 0.8 to 5 Hz. Experimental AC-loss measurements are conducted in this range and compared against analytical and FEA loss estimation. Fig. 124 gives the measured AC losses on an SC sample at wind turbine frequencies. Using the total wire length used in the sample, loss in conductors can be evaluated in W/cm^3 . Fig. 125 shows the comparison of the results in W/cm^3 between measurement and prediction. The analytical and FEA prediction closely matches the experimental measurements. This validates the AC-loss models employed in multiphysics optimization and the feasibility of the machine design proposed for the wind turbine.

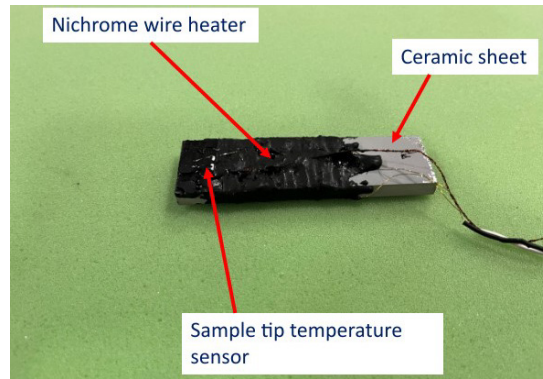


Fig. 116.—Calibration sample.

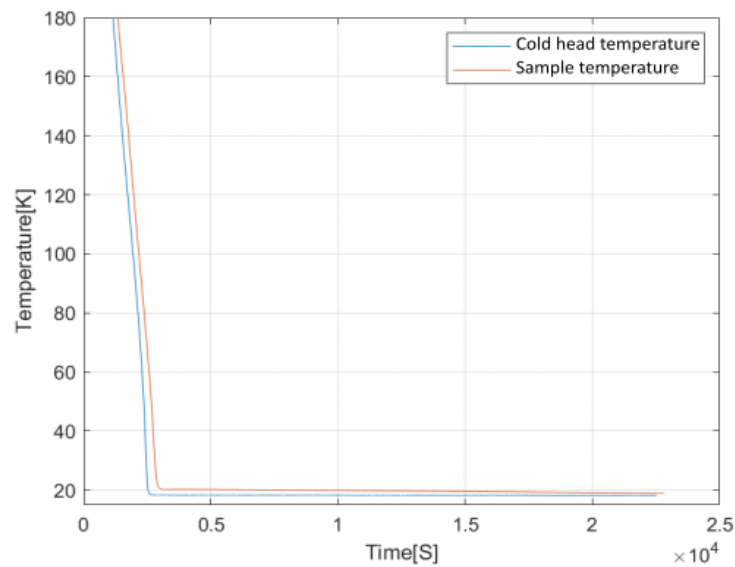


Fig. 117.—Cold head temperature and sample temperature versus time.

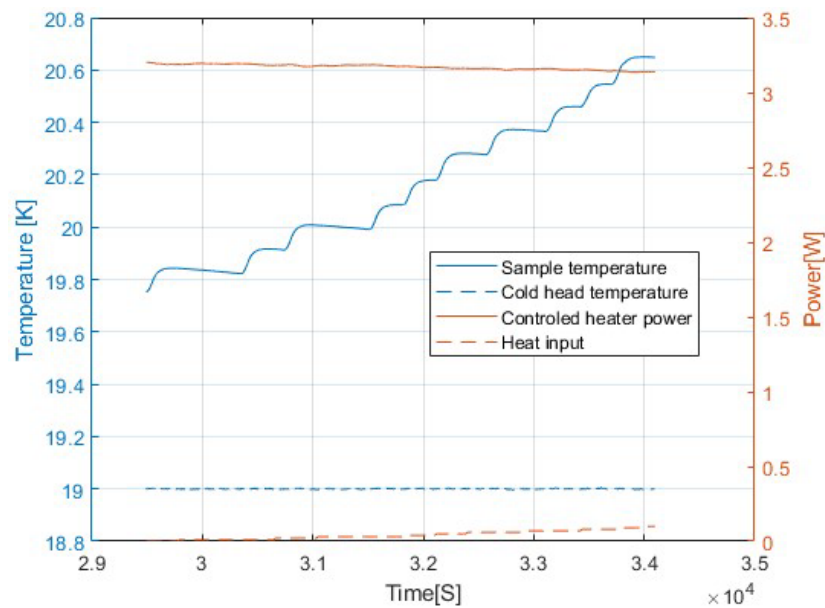


Fig. 118.—Calibration procedure.

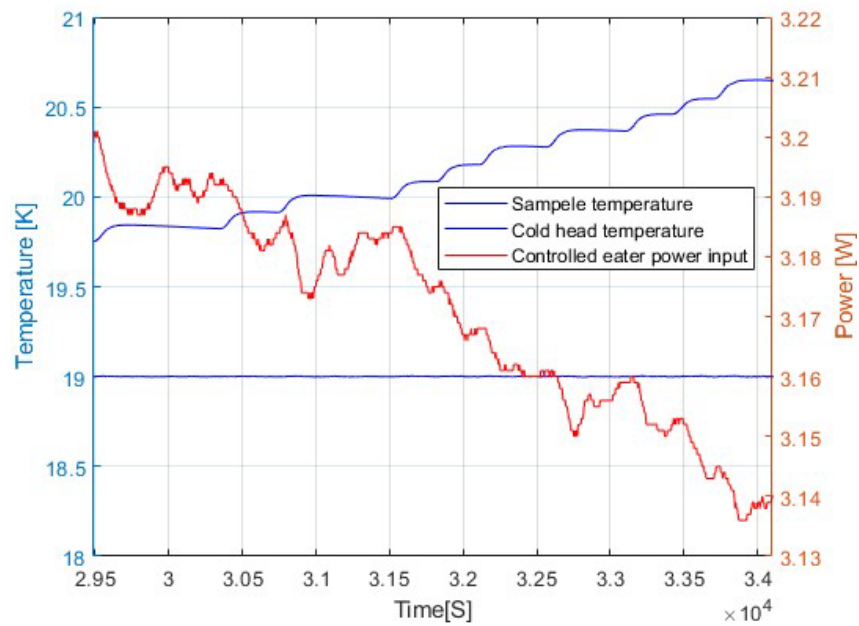


Fig. 119.—Calibration procedure: Sample temperature and control heater input.

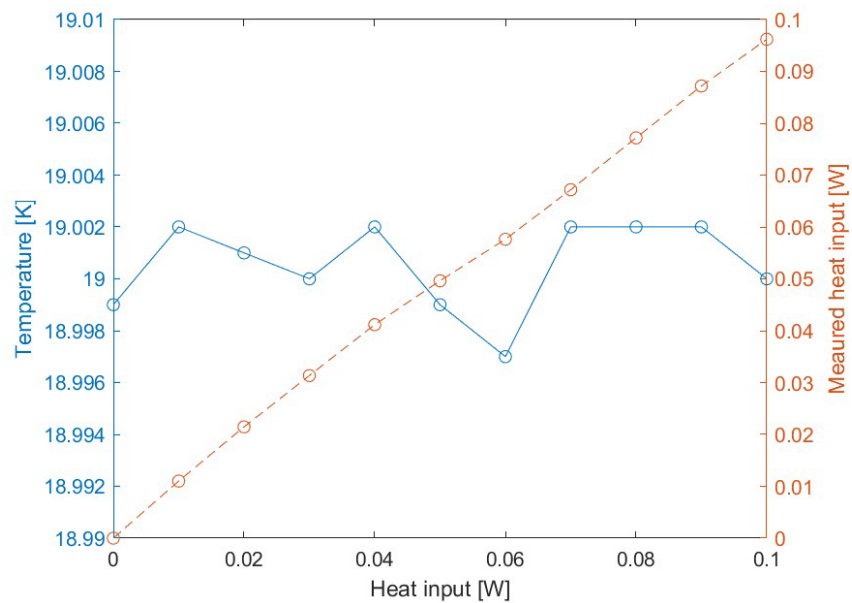


Fig. 120.—Input heat versus measured heat.

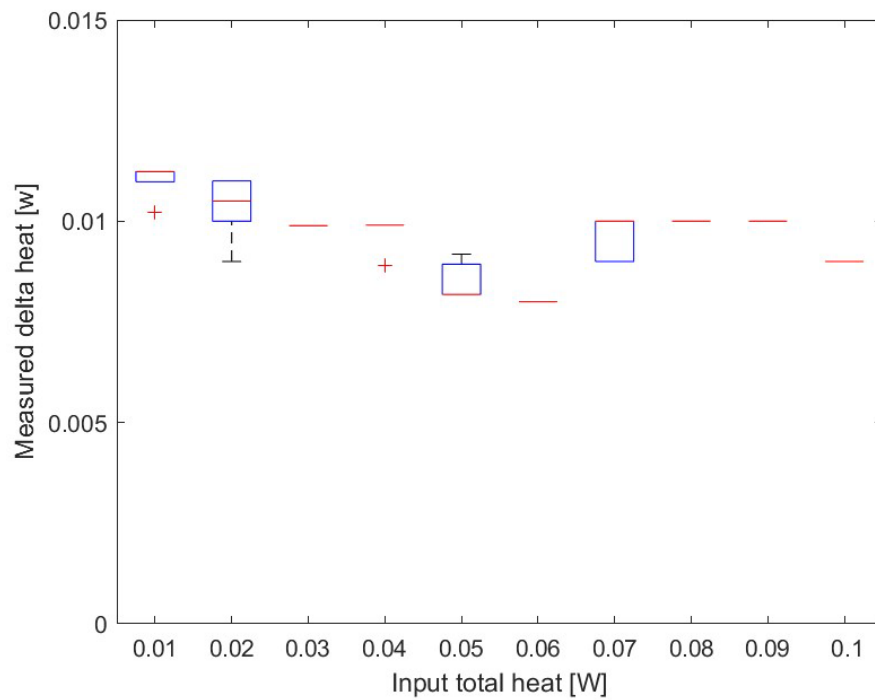


Fig. 121.—Box plot of measured heat input.

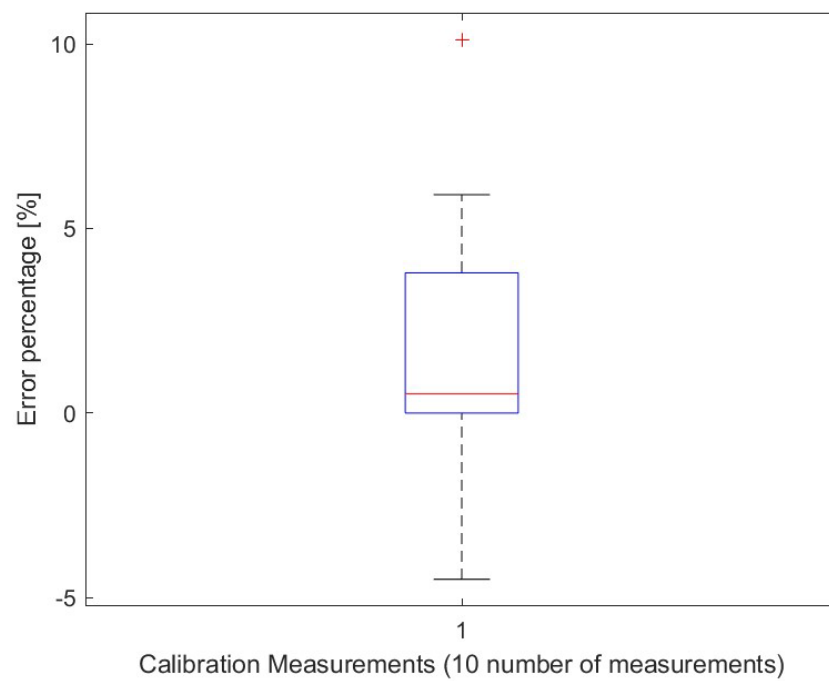


Fig. 122.—Box plot of measurement error.

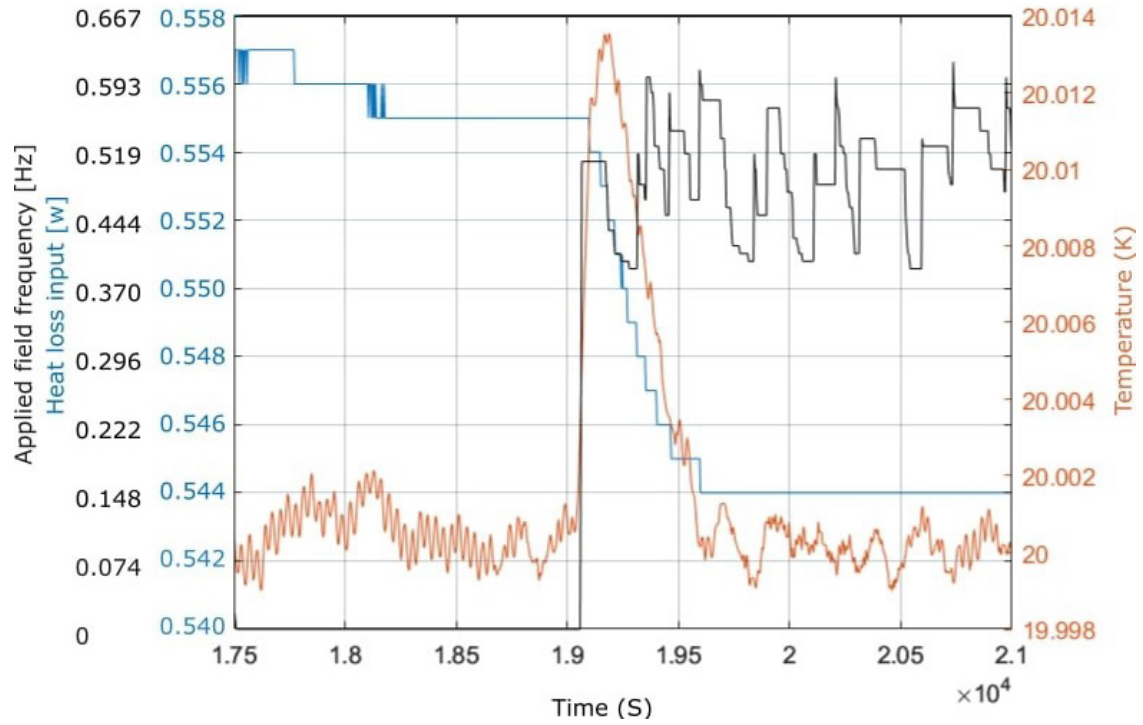


Fig. 123.—Calorimetric method measurement.

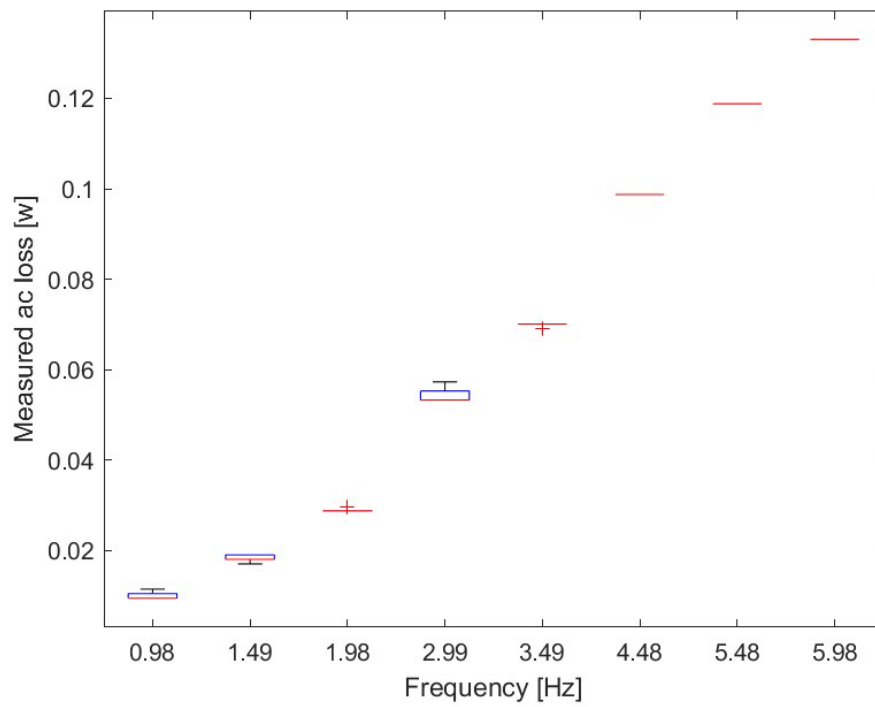


Fig. 124.—Frequency versus loss measurement.

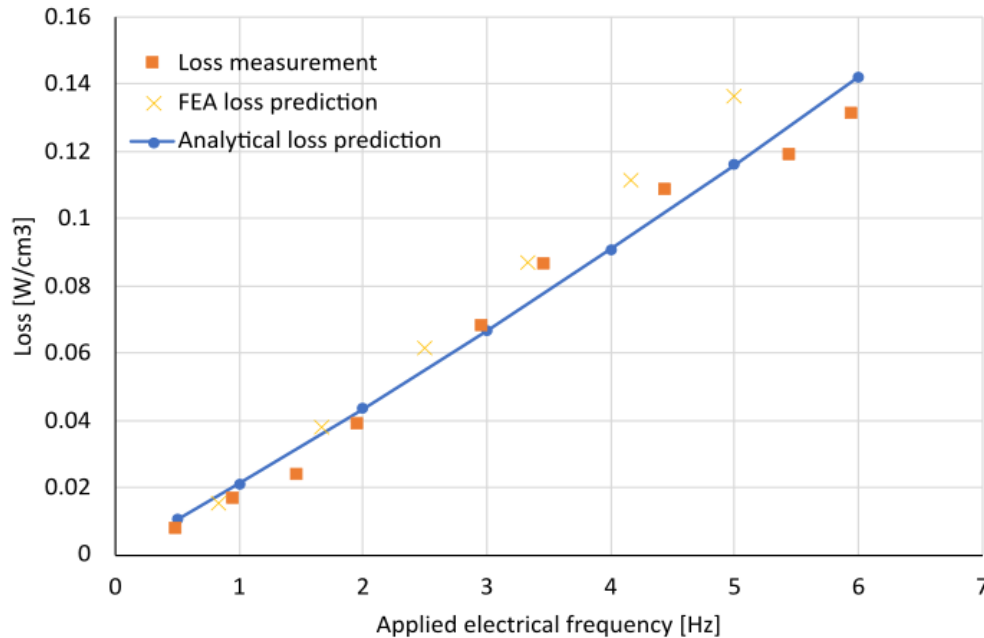


Fig. 125.—Frequency versus loss validation.

3.1.3 SC-Armature-Winding Design Validation

Winding dimensions were obtained from the machine optimization study. A scale model of the motor armature winding was manufactured to demonstrate the winding performance. Fig. 126 illustrates the 2D and 3D FEA solution of flux density variation on the winding cross-section at rated peak current. The winding has a 0.2-T peak flux density at the active region and 0.225 T at the end-winding region. Self-field is 50% of the machine air-gap flux density. Low-AC-loss MgB₂ SC cables were chosen for the optimization study, and commercially available MgB₂ cables were used in the prototyping. Table 16 compares these conductor specifications.

Because high-frequency applications such as the electric-propulsion metallic coil former will create significant eddy current losses, the armature winding must be designed with a nonmetallic coil former. Fig. 127 shows the two types of windings manufactured with the steel and the boron nitride coil former. This boron nitride coil former is designed to replace the metallic coil former in high-frequency applications.

All of these windings have the same cross section with different axial lengths. The following section describes the method used to fabricate the MgB₂ windings in this order: insulation and sensor placement, winding process, heat treatment, soldering, and vacuum impregnation. Windings are fabricated using the wind-and-react method. Fig. 128 shows the winding components. Mica insulation is added between the metal coil former and the windings to provide additional insulation. Cernox[®] (Lake Shore Cryotronics, Inc.) sensors are attached in multiple winding locations to measure the temperature. After adding insulation between the winding and the coil former, the unreacted MgB₂ wires are wound in the coil former as shown in Fig. 129. During the winding process, special attention is given to the strain limits. For an unreacted wire, the strain limit is 4%; for a reacted wire, it is 0.4%. Since this limit is where wire performance begins to degrade, respective strain limits of 2% and 0.2% are used in the winding process. These strain limits can be converted to the minimum bending radius. The bending radius for the 1.28-mm OD prototype cable is around 64 mm and 640 mm, respectively. MgB₂ cables are wound with approximately 22 N tension with two turns per layer.

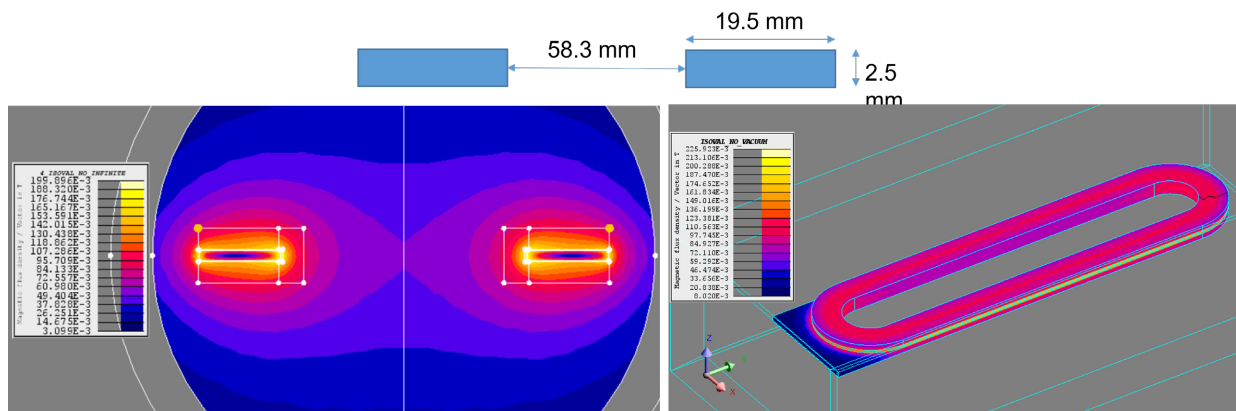


Fig. 126.—Self-field flux-density variation in winding.

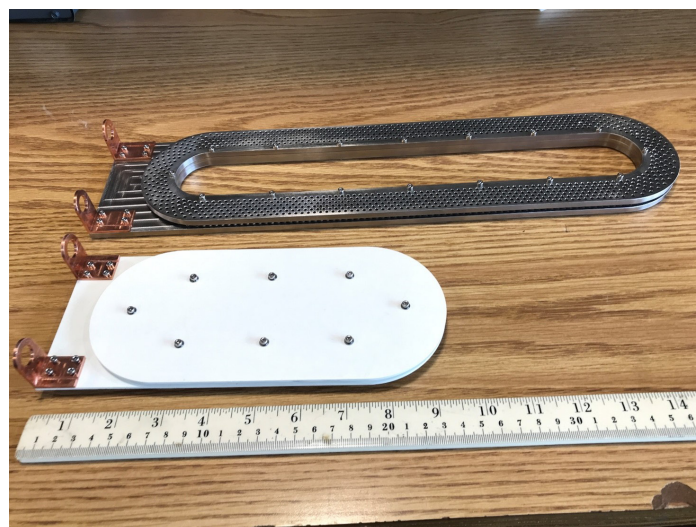


Fig. 127.—Coil former. (a) Steel. (b) Boron nitride.

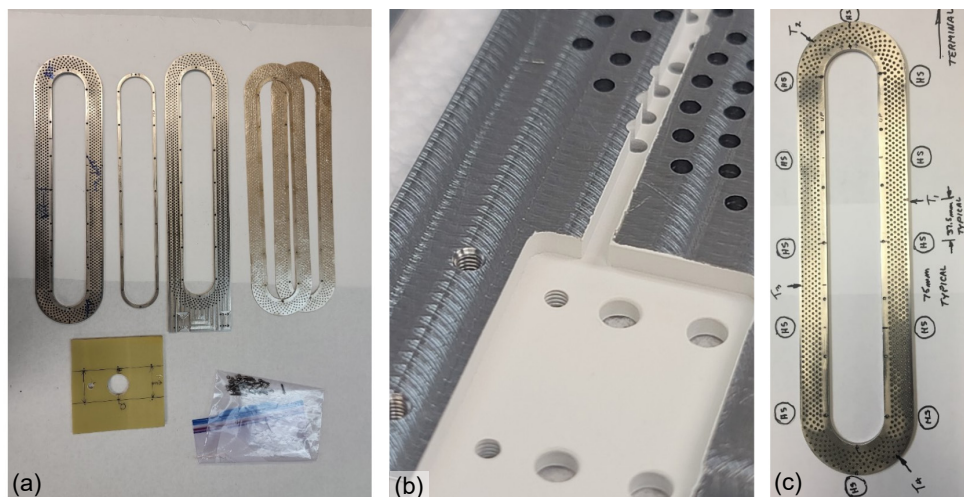


Fig. 128.—(a) Winding components. (b) Mica insulation. (c) Sensor placements.

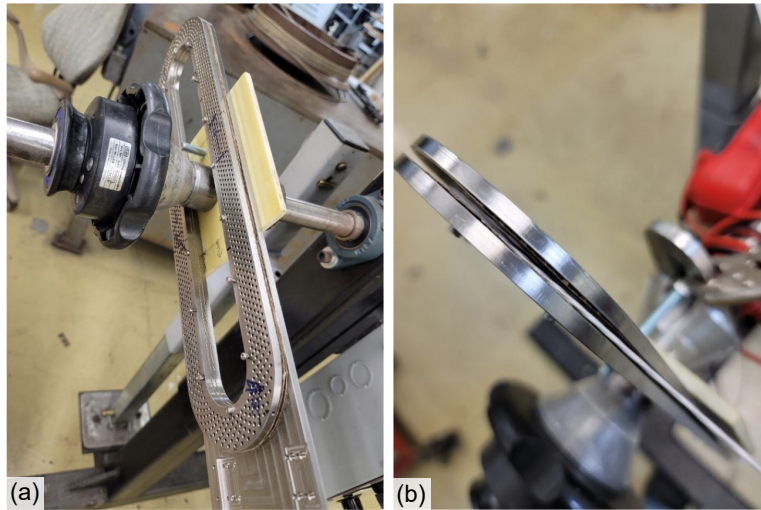


Fig. 129.—(a) Coil former in winding machine. (b) Winding process.

After the cable is wound on the coil former, the windings are heat treated at 650 °C for approximately 90 min. The heat treatment process is performed in an argon gas atmosphere. End wires are rolled as shown in Fig. 130 to minimize the Mg leakage from the exposed cut ends. Due to the exterior Monel sheath (which is about 70% Ni), making the electrical connection to Mg₂ wires is challenging as compared to copper wires. Flux core tin-lead (60-40) solder with a rosin flux is used to remove oxidation and solder MgB₂ to the copper terminals. Reacted wire samples are fragile, and the interface between the wire sample and test rig or coil former should be designed to minimize stresses/strains on the wire that could destroy the sample. The axial strain limit of these multifilament wires is around 0.3%.

A coil wound with an SC wire braided with insulation, S-glass for small wind-and-react, and a polymer fiber for react-and-wind wire has entrapped air and moisture that must be replaced by epoxy. This is done by the vacuum pressure impregnation (VPI) process. The first vacuum outgasses the epoxy, and the operation is repeated when the epoxy is added to the coil. As shown in Fig. 131, the potting pan is a way to simulate the VPI process. The entrapped air is removed through the low-viscosity epoxy and the plethora of holes in the former by leaving the top of the coil open while drawing a vacuum on the enclosure. This VPI process is stopped when there are no more visible bubbles above the epoxy, as shown in Fig. 131(b). The exterior is cleaned of excess layers of epoxy to have good thermal conduction after the coil has been potted and cured. The potted and partially cleaned MgB₂ winding is shown in Fig. 132. The measured room temperature (RT) winding resistance is the same as before potting: 3.7 Ω. The MgB₂ coil designed by UIUC was characterized by the Center for Superconducting and Magnetic Materials (CSMM) at The Ohio State University. The testing chamber is a vacuum-sealed chamber cooled by two Sumitomo SRDK-415D cold heads with a Cernox[®] sensor on each cold head to monitor temperature. The coil is conduction cooled by copper straps providing a thermal link to the cold heads. Three Cernox[®] sensors are placed directly on the steel frame of the coil to measure temperature. The cryostat and the cooling arrangement of the armature windings are shown in Fig. 133. As seen in Fig. 133, the armature winding is placed on the center of the cryostat. The cooling is divided into two stages. The first stage offers radiation protection using a painted Al (black finish) cover and is maintained at 70 K.

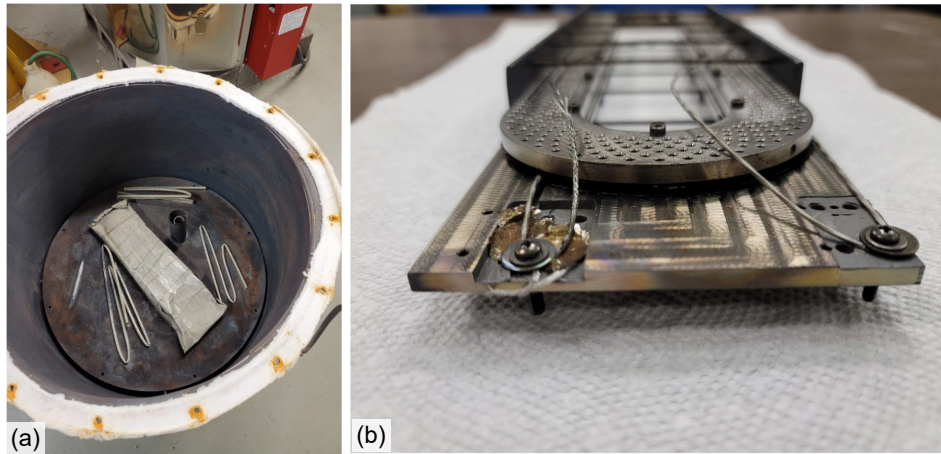


Fig. 130.—(a) Heat treatment process. (b) Winding after heat treatment.

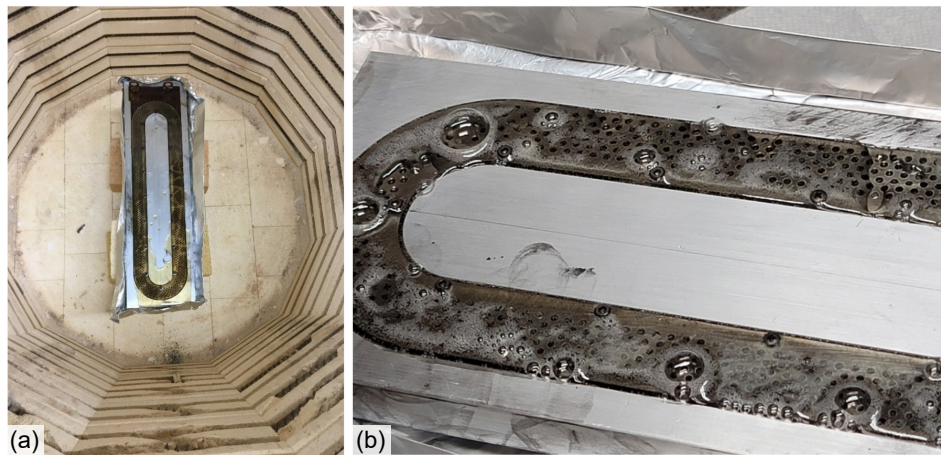


Fig. 131.—(a) Vacuum impregnation process. (b) Air bubbles coming out.

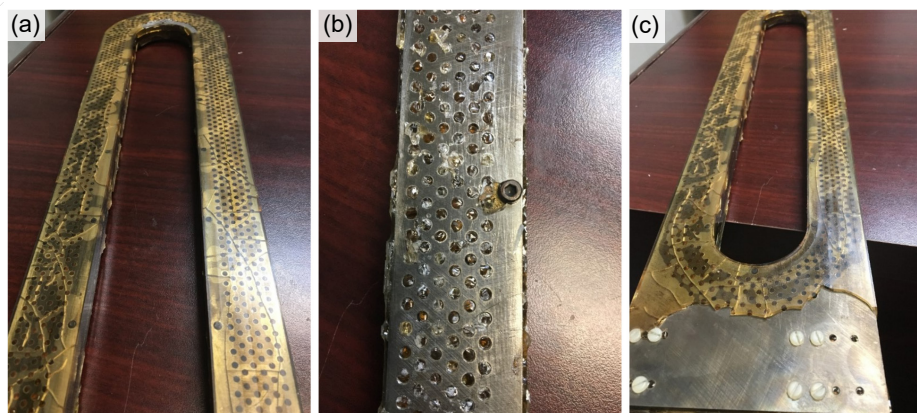


Fig. 132.—Completed winding.

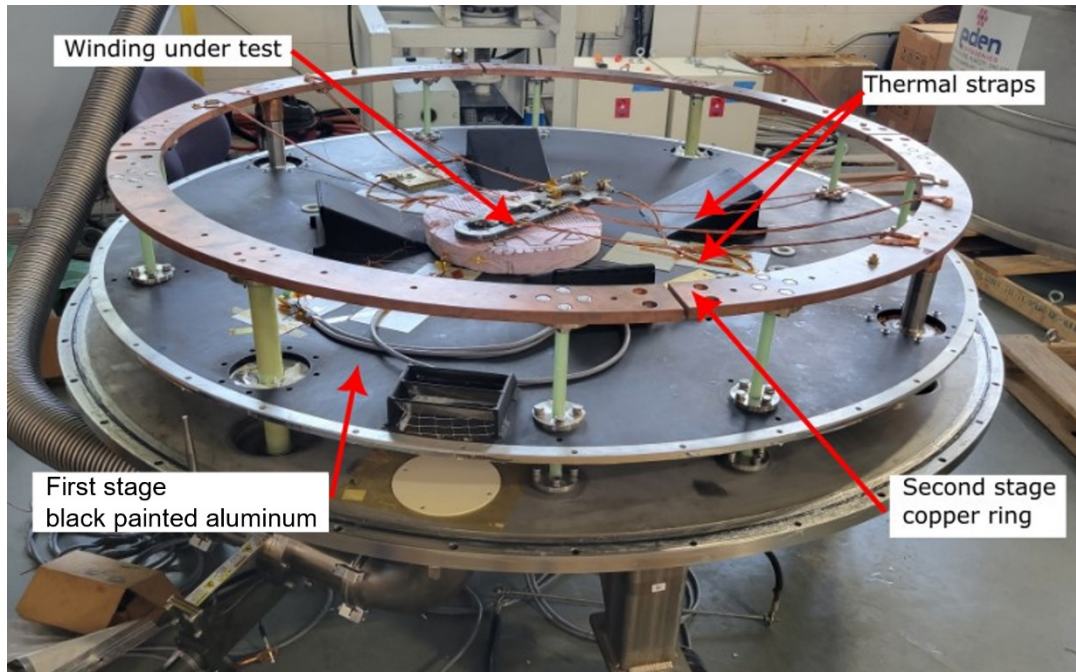


Fig. 133.—Armature winding under test: cooling arrangements.

The second stage is a copper ring (four pieces) cooled by cryocoolers to 4 K. The cryocooler cold head is directly connected to two copper rings that cool the winding to the desired temperature. Thermal straps have attached the armature winding to the copper ring. These thermal straps are soldered to stainless-steel washers on the armature winding to minimize contact thermal resistance. Current leads and thermal straps are made from Litz copper cables to reduce AC losses due to the armature winding self-field. Current leads are optimized for current and heat leak-in, and ohmic losses are thermally grounded at first stage of cryostat (70 K). A layer of MLI is added for radiation protection, as shown in Fig. 134. Finally, the whole setup is enclosed using a steel lid and is vacuumed. Fig. 135 shows the thermal straps and voltage taps connected to the armature winding. Two voltage taps are placed between the interface of the copper connection and the current lead for accurate measurement.

The cooling progression is shown in Fig. 136. The cold head temperature quickly reached 3.8 K, while winding had a nonuniform temperature profile. A continuous 50 mA was applied during the cooldown procedure to visualize the T_c transition, and the resulting decrease in resistance of the coil is shown in Fig. 137, with the voltage drop versus temperature shown in Fig. 138. This indicates that the winding transformed into the SC state. Although the coil reached a low temperature, the T_c transition could not be measured with use of the copper current leads available for this experiment.



Fig. 134.—Armature winding under test: MLI for radiation shield.

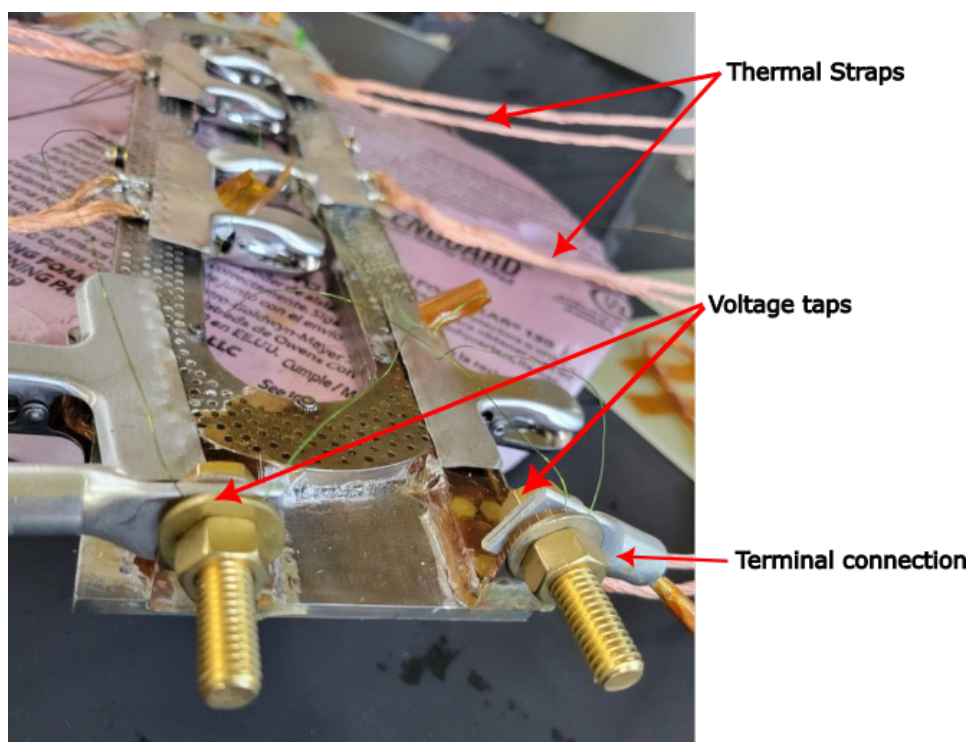


Fig. 135.—Armature winding under test: terminal connection and sensor placements.

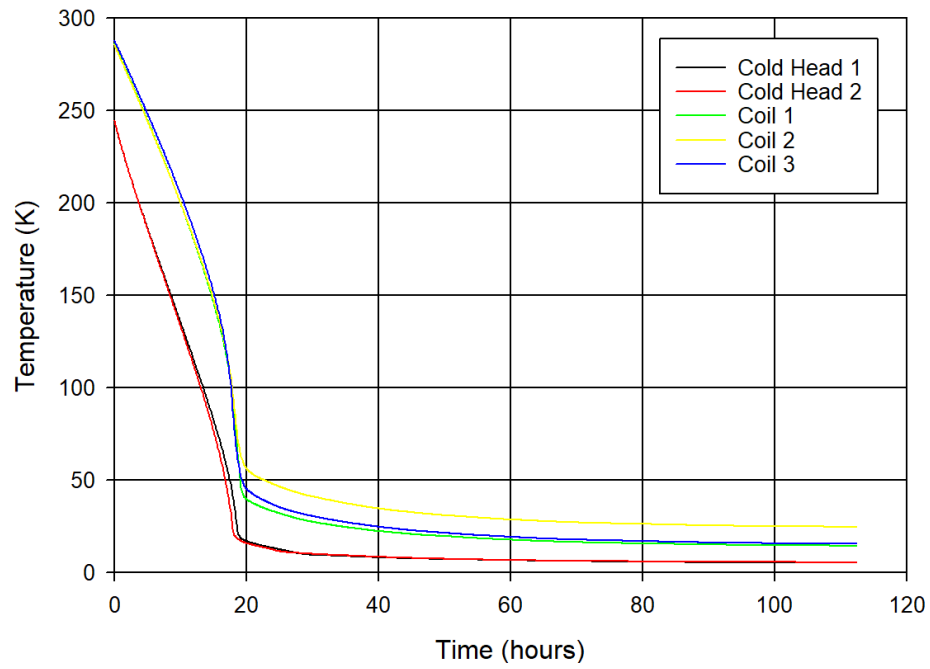


Fig. 136.—Winding temperature measurements.

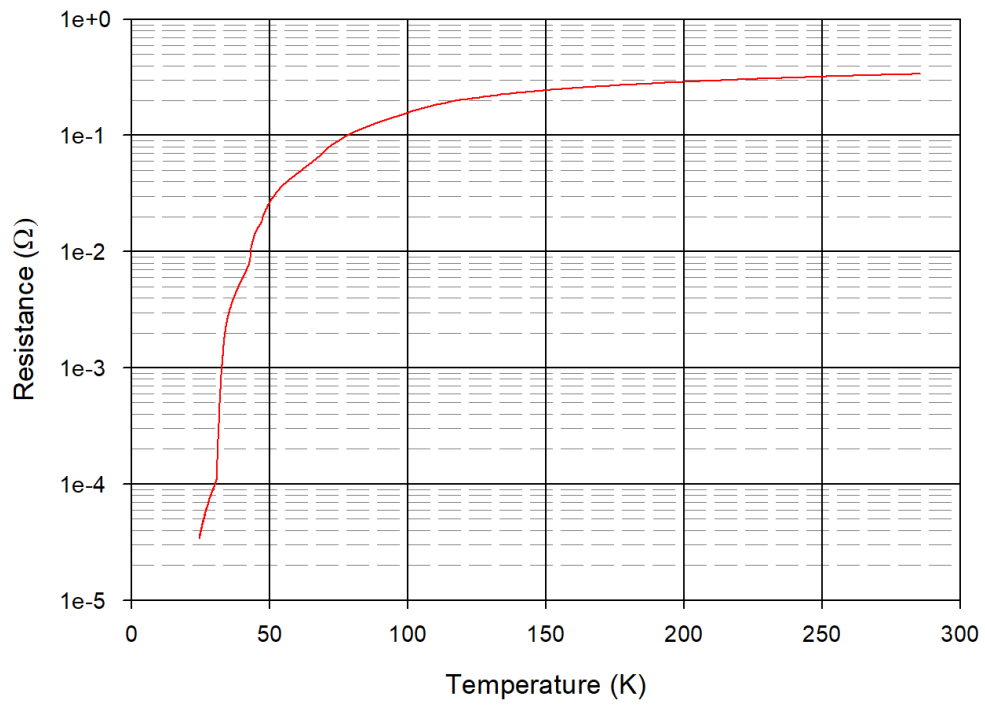


Fig. 137.—Resistance of coil with 50 mA.

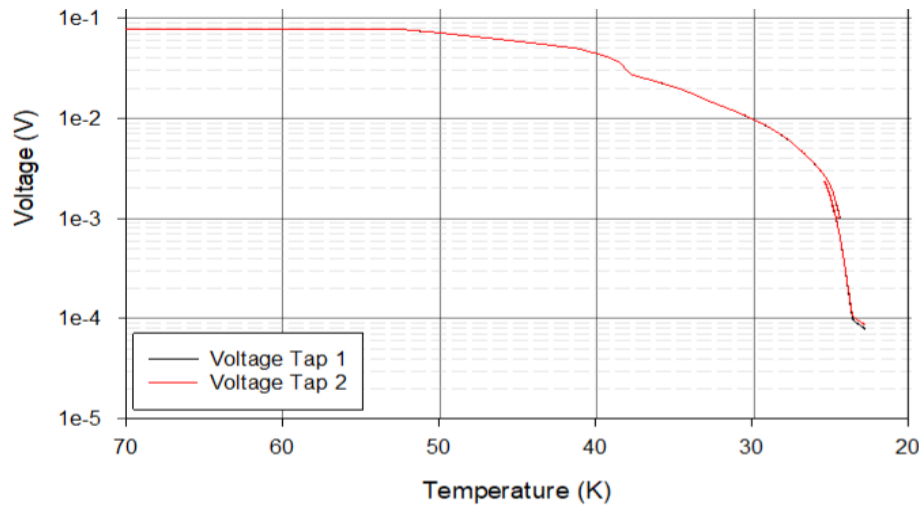


Fig. 138.—SC transition of winding.

A direct-current (DC) test was conducted to verify a SC state has been achieved. Fig. 139, shows the I - V curve of the coil. A high baseline slope of $1.08 \text{ m}\Omega$ was measured, as expected considering copper current leads, and the critical current (I_c) was measured as 2 A using the $1\text{-}\mu\text{V}/\text{cm}$ criterion. But after integrating the offset from the copper leads, the corrected I_c is reported as 5.2 A with an n -value of 5.2, as shown in Fig. 139.

A series of AC loss measurements were conducted at frequency ranging from 50 to 250 Hz and current range from 3 to 5 A_{RMS} . Fig. 140 shows the measured AC loss for various conditions. AC loss was limited to 2 W, as any value greater than 2W forces the system into a thermal runaway. This is because the cooling provided by the copper straps could not allow the coil to reach equilibrium while staying below T_c . This creates normal zones in the SC regions that contribute more AC loss than SC counterparts and forces a linear increase in temperature. The AC loss was determined by subtracting the heat capacity of the cold heads at equilibrium without induced AC from the equilibrium-induced AC state.

The AC loss trendline follows a power of index of 1.54 and 1.73 with increasing frequency for $3A_{RMS}$ and $4A_{RMS}$, respectively. The Eddy current in the steel casing and coupling losses in the MgB_2 wires dominate the AC loss contributions.

3.1.4 Rotor Cooling Validation

A rotating cryocooler test-bench setup was developed to validate the feasibility of the alternative cooling scheme. Sunpower's CryoTel GT Cryocooler (Fig. 141) (Ref. 129) was chosen for field-winding cooling due to its light weight, high efficiency, and low noise performance. This is a free piston, Stirling-cycle-based cryocooler. The cryocooler specification is tabulated in Table 23. Performance characteristics of this cryocooler under rotating conditions are benchmarked using this test stand. Fig. 142 shows the cooling performance of the cryocooler at the stationary operating conditions with various heat-rejection options. The cryocooler has a minimum cold head temperature of 33 K at 0 W lift. A 23°C forced air-cooling rejection on the cryocooler can lift around 5 W at 50 K. Fig. 143 shows the 3D computer-aided design (CAD) modeling of the rotating cryocooler test bench. The test bench consists of a vacuum chamber connected to the cryocooler. A heat load is attached to the cold head, and the cold head temperature and cryocooler performances are monitored under rotating conditions. The cryocooler uses a slip ring for power and measurement data transfer. Fig. 144 shows the test bench setup where the cryocooler and vacuum chamber are rotated using an external drive.

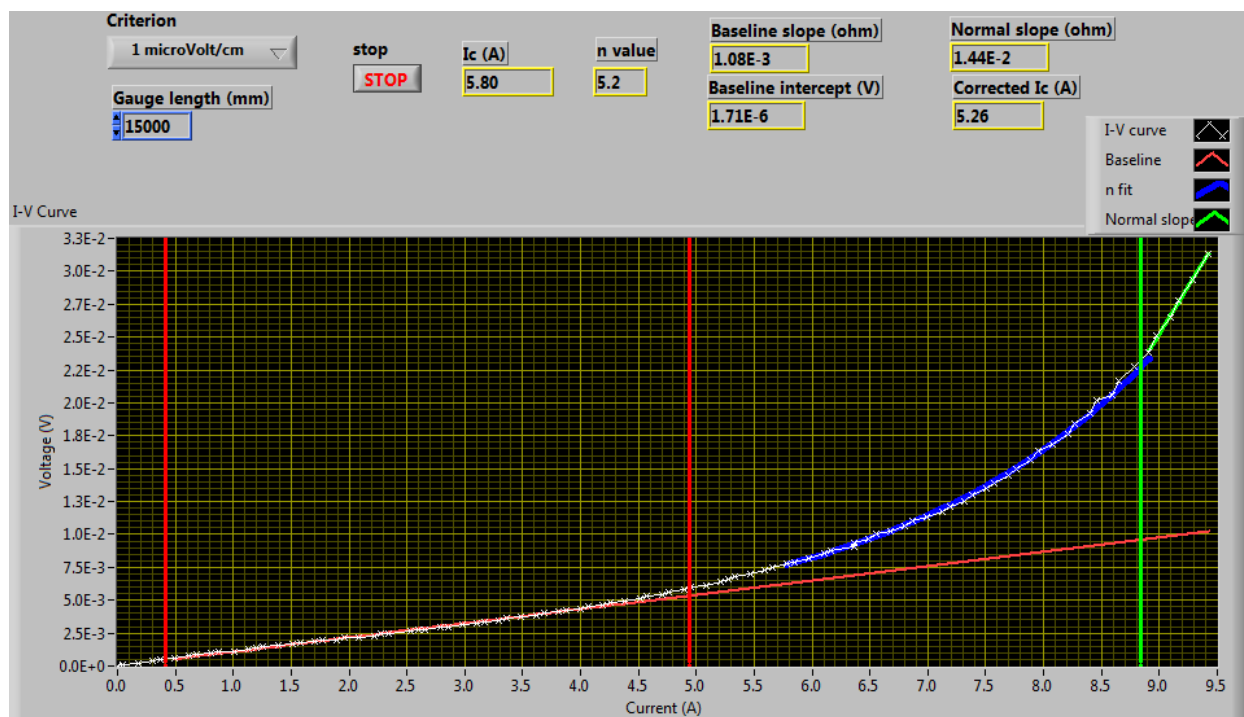


Fig. 139.—Superconducting to normal transition.

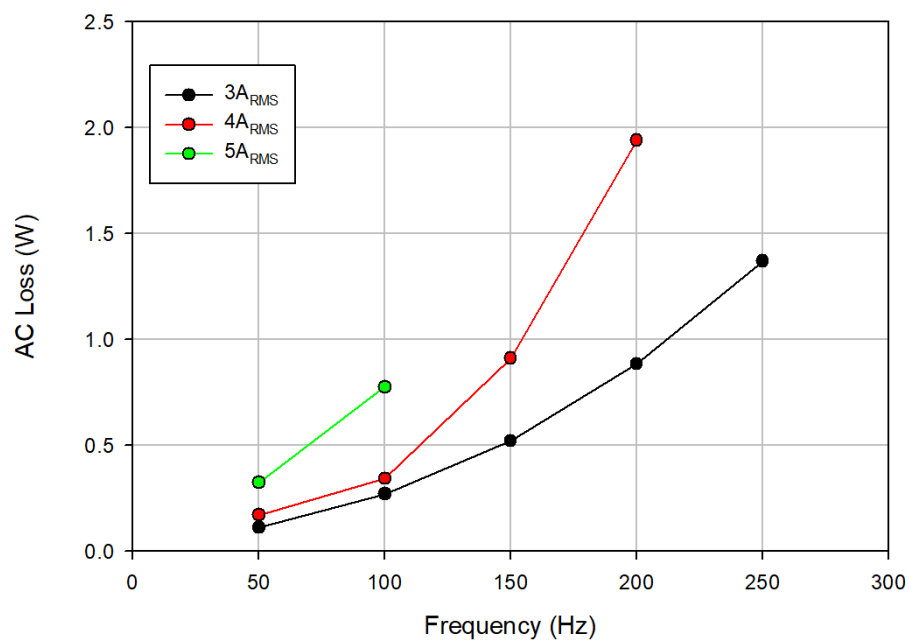


Fig. 140.—Heat generation from AC loss for various conditions.

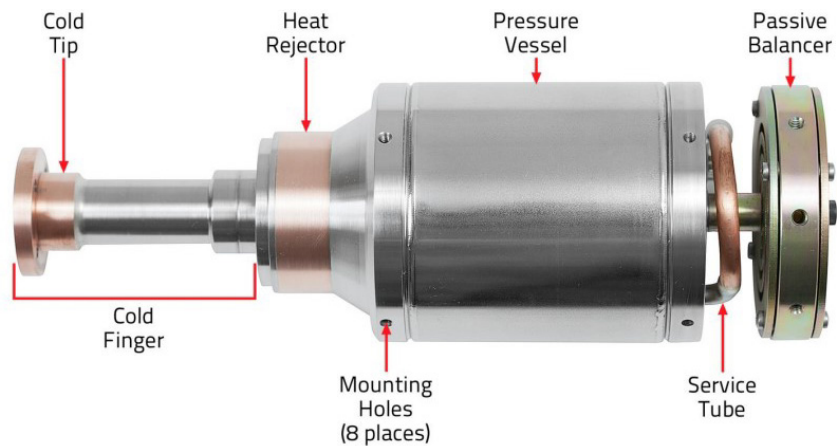


Fig. 141.—Sunpower's CryoTel GT cryocooler used in the test bench (Ref. 129).

TABLE 23.—CRYOCOOLER SPECIFICATIONS

Cryocooler axial length with passive balancer, mm.....	275
Cryocooler pressure vessel OD, mm.....	83
Cold tip diameter, mm	25.4
Cryocooler weight with passive balancer, kg	3.1
Cryocooler input power, W.....	240
Heat rejection at 50 K with 23 °C cooling, W	5

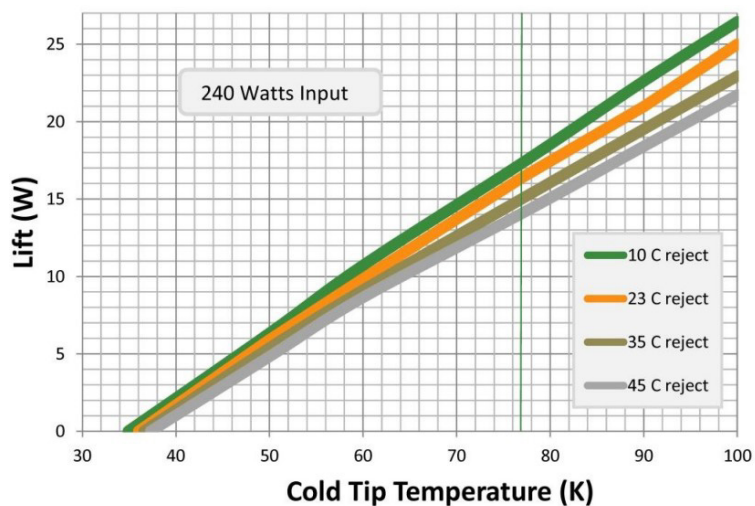


Fig. 142.—Nominal performance of cryocooler at various heat rejections (Ref. 129).

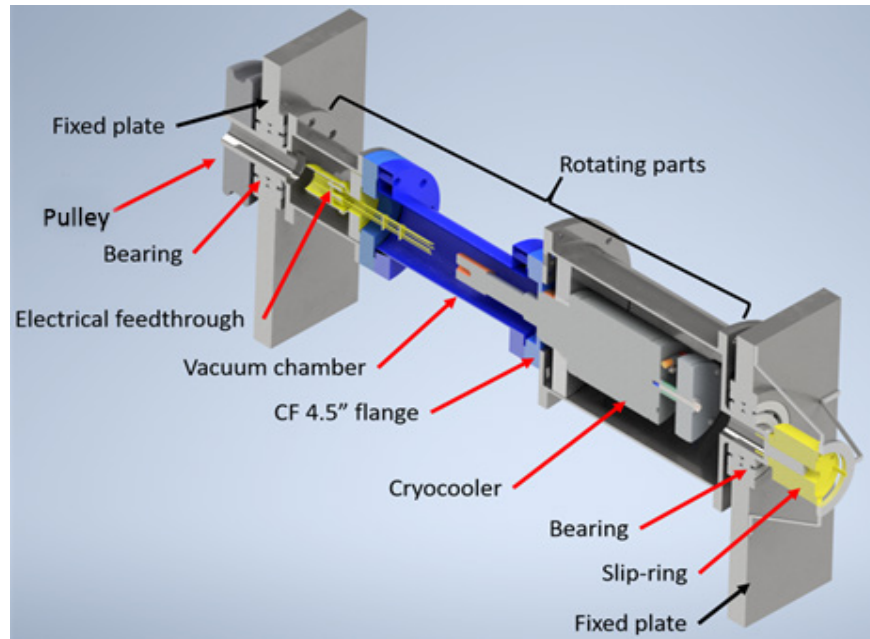


Fig. 143.—3D CAD of the rotating cryocooler test bench.

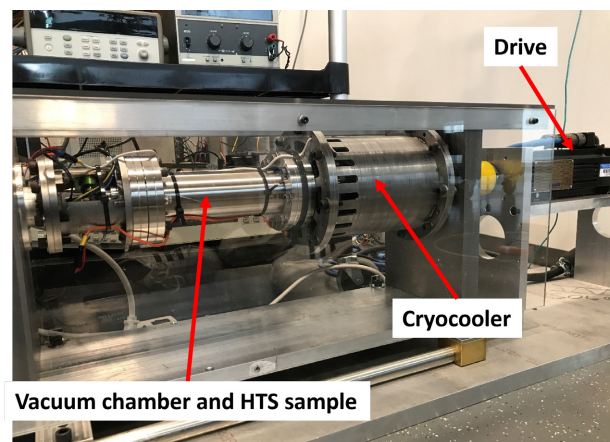


Fig. 144.—Rotating cryocooler test bench.

For initial testing, the cryocooler is rotated up to 1,000 rpm without powering the device and then tested up to 350 rpm while applying a heat load. Vibration and cold head temperatures are recorded before and during the rotation. Fig. 145 shows the cold head temperature before and during the rotation at 350 rpm. These preliminary results show that the cryocooler can maintain the targeted cold head temperature while rotating. Fig. 146 shows the vibration data on the test stand during the rotation before and after the cryocooler is turned on. The cryocooler is turned on while it is rotating. A default bias of 1g is added to the axial direction (z) to distinguish the data from the radial direction (x and y) vibration measurements. Fig. 147 shows vibration of the cryocooler during stationary and rotating operation. The results showed that the cryocooler has significant vibration in the axial direction regardless of the rotation, and the impact of rotation on the vibration is insignificant. In future results, high-frequency tests will be performed in armature-winding, rotating cryocooler, and AC loss measurement.

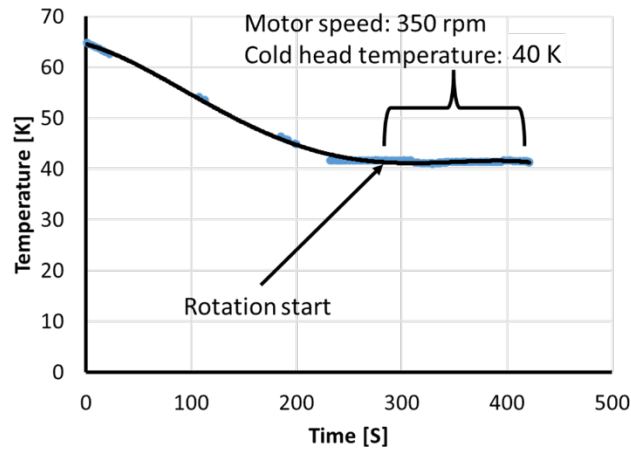


Fig. 145.—Cold head temperature during rotation.

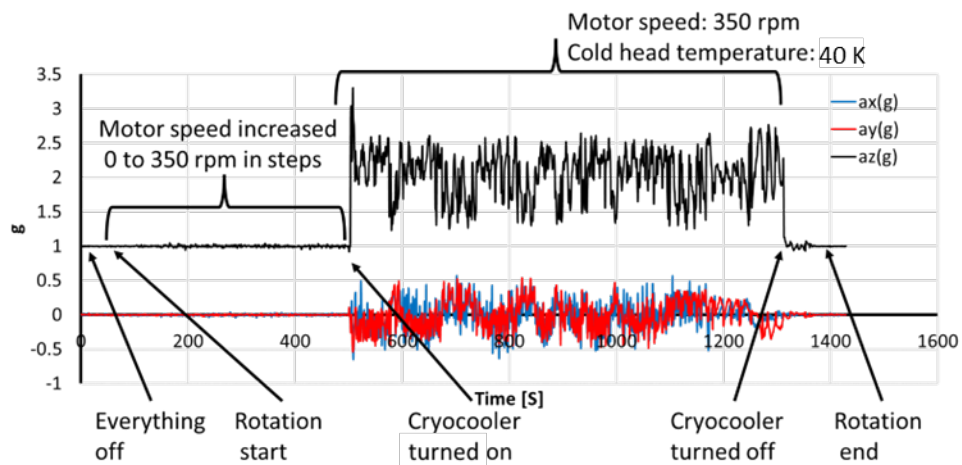


Fig. 146.—Vibration comparison during rotation.

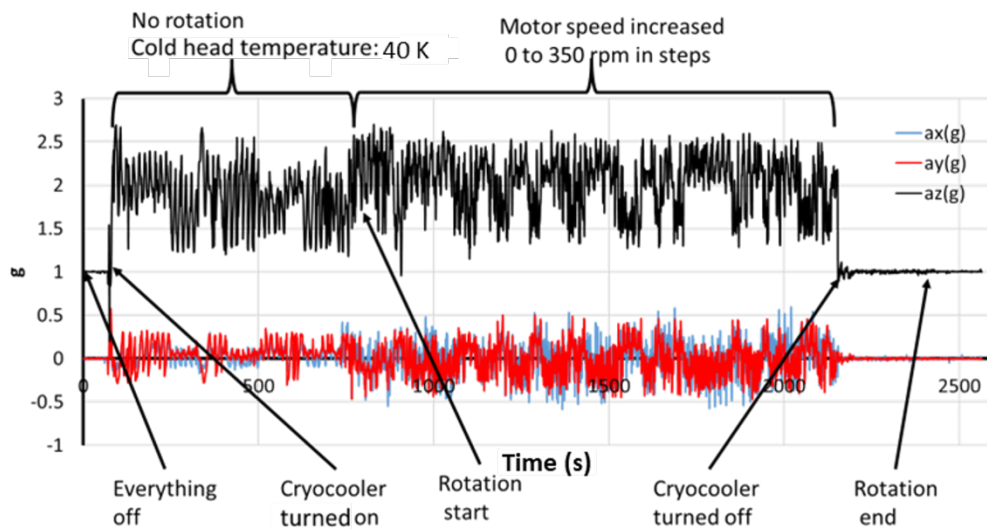


Fig. 147.—Vibration comparison during rotation and when stationary.

3.1.5 Quench Modeling, Detection, Protection, and System-Level Validation

One of the most important considerations in the implementation of the integrated cryo-motor system of the CHEETA aircraft concept is the SC motor quench detection and protection. The project entails understanding the stability and quench dynamics because these are the key to designing a true detection system. The SC motor being designed and developed will use MgB₂ coils for carrying both AC and DC currents and faces challenges in the true quench detection. Extensive literature review was performed to evaluate the best possible quench detection techniques and quench protection schemes.

3.1.5.1 Quench Dynamics and Stability

A quench refers to the sudden loss of superconductivity when the temperature of the material is raised. In the SC state, the resistance of the magnet coil windings is zero and hence no energy is required to maintain current flow. If the coil temperature rises above the superconductivity threshold T_c , the windings suddenly develop a finite resistance. The several-dozen amperes of circulating current passing through this elevated coil resistance create heat. This heat causes a sudden, explosive boiloff of LH₂. The aim of any quench protection system is to prevent permanent conductor degradation in the event of a fault condition that induces a quench. Quench protection involves three key steps, all of which must be accomplished within a time budget determined by the rate of growth of the disturbance and the resilience of the conductor. The three key steps are as follows:

- Detection: There are several methods to detect a quench, including measuring coil voltage drop, coil strain, inductive coupling, temperature, and pressure.
- Assessment: The disturbance needs to be assessed to determine if it is going to induce a quench and to prevent false positives.
- Protection: Action required to prevent degradation if the winding or magnet is quenching. For windings with a large stored energy, this is typically accomplished through heaters embedded in the windings and a dump-resistive circuit into which the stored energy is dissipated.

Developing an effective quench protection system requires understanding the stability and quench dynamics of the magnet in order to design a detection system. The safe operational limits of the conductor must also be known. These two factors ideally determine the time budget for protection. High T_c superconductors have quench characteristics different from those of low T_c superconductors. In general, quench propagation velocities in high T_c superconductors are slower than those in low T_c superconductors. This indicates that the coils wound with the high T_c superconductors are unlikely to be self-protecting when the quench occurs. In this study, we will focus on MgB₂ and YBCO magnets, both of which are high-temperature superconductors (HTSs).

3.1.5.2 Quench Detection

Various quench detection methods have been proposed based on temperature, pressure, ultrasonic waves, flow velocity, and voltage. Voltage and acoustic techniques, however, are the most commonly used approaches. The voltage-based techniques for detection of quenching in both AC and DC circuits will be discussed here. The partial-power active power detection method can be implemented both for AC and DC currents (Ref. 130). The principle is shown in Fig. 148(a), where L_1 and L_2 are self-inductances of the coils, and r_1 and r_2 are the resistances of the coils emerged with the quench. The power P is given by

$$P = (v_1 - v_2)i = (r_1 - r_2)i^2 \quad (47)$$

In the SC state, r_1 and r_2 are zeros and P is zero. When the quench occurs, the resistances are not zero.

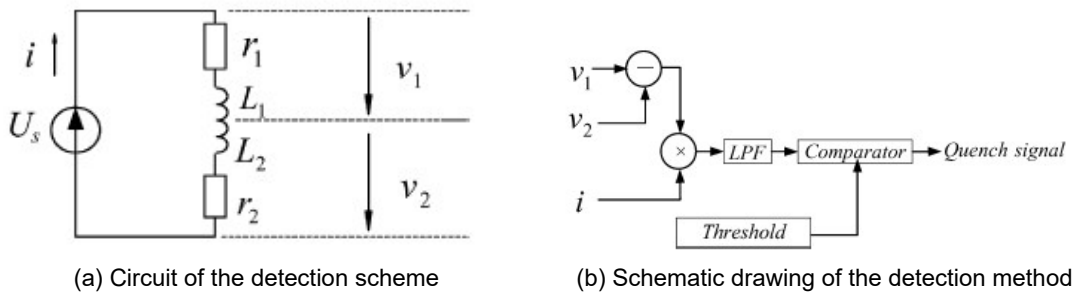


Fig. 148.—Partial active power detection method as proposed in Reference 130.

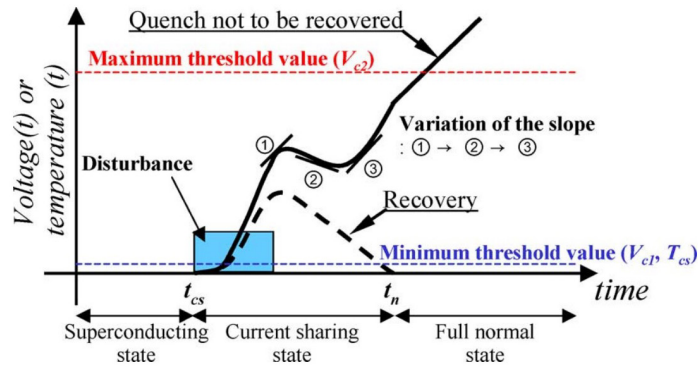


Fig. 149—Quench detection method proposed in Reference 131.

So, the quench can be detected except when $r_1 = r_2$, which is considered to be a rare case. When quench occurs with $i = I \sin(\omega t)$, then

$$P = \frac{(r_1 - r_2)I^2}{2} - \frac{(r_1 - r_2)I^2 \cos(2\omega t)}{2} + \text{noise} \quad (48)$$

As shown in Fig. 148(b), a low-pass filter (LPF) is added after the original partial power. After being filtered by the LPF to reduce the noise, the signal is used for the quench recognition. When its value becomes larger than a specified threshold, it is recognized as the quench. This threshold would, of course, be dependent on the superconductor's properties being used. Reference 131 describes a similar voltage-based quench detection by observing the normal transition behavior of HTS coils. This method could detect the quench more rapidly. The idea behind this method is shown in Fig. 149. The initial quench behaviors of HTSs can be classified into three classifications:

- Typical runaway
- Recovery (if the cryo-coolant is enough for this quench)
- Runaway after insufficient recovery

This system is able to detect quench by means of three reference signals—the minimum threshold, the maximum threshold, and the time rates of changing temperatures and voltages. The minimum threshold is used for triggering the actual process of the quench detection. Even if only one value of all acquired data exceeds minimum thresholds after generating the disturbance, the system will start carefully to assess the transition to quench in the sample.

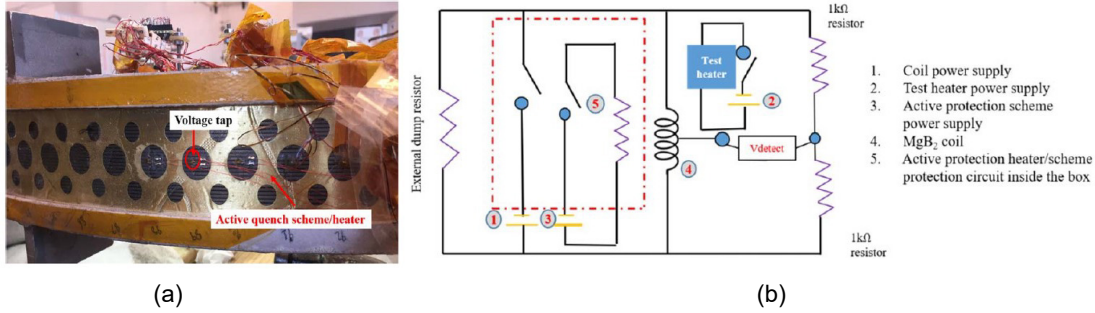


Fig. 150—Active quench protection method as proposed in Reference 132. (a) Active quench protection heater used for coil. (b) Schematic of electrical circuit used for quench detection and protection of SC coil.

3.1.5.3 Quench Protection

Due to the relatively large minimum quench energy (MQE) values, MgB_2 coils are less likely to quench than low-temperature SC coils (such as NbTi coils). However, when a quench does occur, the normal zone in a MgB_2 coil propagates more slowly (normal zone propagation velocity (NZPV) is 5 to 150 cm/s) compared to normal zone in a NbTi coil (10 to 50 m/s), which can lead to higher temperatures at the local spots due to the longer accumulation of heat locally (in the normal zone). At a lower operating temperature and a fixed magnetic field, the coil I_c can get significantly higher, which will inevitably introduce more joule heating. When large coil currents are being employed during normal zone propagation, the magnitude of NZPV is further augmented as a result. This makes both active and passive protection schemes more difficult to implement.

The normal zone in the test coil is created by means of a microheater placed on one of the turns inside the solenoid winding (Ref. 132). The microheater (its initial power being selected empirically) is switched on for a short time, just as long as was necessary to create a normal zone in a particular part of the winding (about 0.1 s). Thus, the original cause of the normal zone's appearance is eliminated, and the zone is left on its own. This mode of operating the microheater simulates the formation of a normal zone as the result of a catastrophic flux jump or as a result of the displacement of the coil windings. After the disconnection of the microheater, the subsequent behavior of the normal zone is duly recorded. The active quench protection scheme is shown in Fig. 150.

3.1.5.4 Quench Propagation Numerical Model

Assuming uniform temperature over any cross section of the conductors, the time evolution of the temperature along the wire has been obtained by solving the 1D heat flow equation

$$\frac{d}{dx} \left(K(T) \frac{dT}{dx} \right) + \frac{IE(T)}{A} + q_{in} - C(T) \frac{dT}{dt} = 0 \quad (49)$$

where x is the distance along the wire; t is time; T is temperature; A is area of the cross section; $K(T)$ and $C(T)$ are the thermal conductivity and the specific heat per unit volume of the conductor, respectively; $q_{in} = Q_{in}/(LA)$ is the initiating energy pulse; I is the transport current flowing through the wire; and E is the electric field along the wire, which depends on the temperature. The resulting joule heating per unit volume, $IE(T)/A$, has been calculated by assuming the metal matrix and the superconductor as two impedances connected in parallel. For $T < T_c$: $E_m(T) = \rho_m(T)J_m(T)$ in the matrix and $E_{sc}(T) = E_0 \left(\frac{J_{sc}(T)}{J(T)} \right)^{n(T)}$ in the superconductor. For $T > T_c$: $E(T) = \rho_n(T)I/A$. The current I is given by

$$I = I_m + I_{sc} = (1 - f_{sc})AJ_m(T) + f_{sc}AJ_{sc}(T) \quad (50)$$

where f_{sc} is the superconductor fractional area, ρ_n is the resistivity of the composite in the normal state, ρ_m is the resistivity of the metal matrix and $E_0 = 0.0001$ V/m. As a first approximation, we have considered here that the normal resistance of the composite is dominated by the resistance of the sheath. J_c is the critical current density of the superconductor ($J_c = I_c/f_{sc}A$), which depends on the temperature. The current share of the superconductor and the metal matrix, I_{sc} and I_m , respectively, should satisfy the current equation (Eq. (50)) for a given transport current I . The heat transfer equation can be solved numerically with the contour conditions $T(x=0) = T(x=L) = T_0$, using the thermal and electrical properties measured on conductors. Using the material properties of MgB_2 , (Ref. 108) can be solved to determine the quench propagation velocities, MQEs, and minimum propagating zones (MPZs) (Ref. 133).

3.1.6 Superconducting Motor Design: Conclusions

This section detailed the design of a fully SC aircraft propulsion motor. The motor is cooled using LH_2 , which also provides power to the aircraft via a fuel cell. Constraints on the motor design are delineated by CHEETA. An outer-rotor topology is employed, with the armature coils on the stator and the field and shielding coils on the rotor. An electromagnetic design was selected based on the results of an optimization study. Separate cooling schemes are employed for the stator and rotor. The stator cooling scheme is a HEX with helical tubes embedded in a thermally conductive heat sink. LH_2 flows through these tubes, absorbing the heat generated by AC losses in the armature coils. The helical tube curvature benefits the cooling in two ways: maintaining fluid contact with the tube wall closest to the conductor and inducing vorticity in the flow. We presented a stator HEX design that was optimized for maximum motor specific power. LH_2 is not used to cool the rotor due to safety risks associated with a rotary transfer coupling. Rather, a cryocooler is used to cool the rotor. An experiment was conducted to validate this approach for speeds up to 1,000 rpm.

The work presented here paves the way for practical aircraft propulsion motors with high specific power. These motors will be a key enabler of emission-free commercial aircraft. Next steps include further testing of our risk-reduction experiments. Following completion of the current program we intend to perform a complete set of LN_2 flow-boiling tests in straight and coiled tubes and may test with LH_2 as well. We also intend to test our rotating cryocooler experiment up to 4,000 rpm and include a heat load on the cryocooler. Additionally, we plan to test our rotating AC-loss experiment at higher speeds. Following completion of these risk-reduction experiments, next steps include the build and test of the stator, rotor, and eventual machine.

3.2 Cryogenic Power Electronics

The use of power electronics at cryogenic temperatures (CTs) offers several significant opportunities but also brings additional challenges not seen in RT systems. Throughout this study, selected semiconductor and power converter components were studied at CTs to understand the performance implications of these devices under low-temperature conditions. An additional design study and experimental assessment was performed on an inverter configuration developed through the current study.

3.2.1 Component Characterization

For this project, we have investigated the cryogenic (LN_2 temperature) characterizations of different semiconductors and power converters. The following conclusions can be drawn from our investigations:

1. Conventional Si metal-oxide-semiconductor field-effect transistor (MOSFET) and Si insulated-gate bipolar transistor (IGBT) are advantageous at LN_2 temperature with significant reductions of conduction loss and switching loss (turn-off switching loss).

2. Silicon carbide (SiC) MOSFET has degraded performances at CT. The on-state resistance is increased up to 2 to 3 times when compared with RT. The switching performance for SiC MOSFET is determined by the structure and voltage levels. The evaluated SiC MOSFETs with voltage rating less than 900 V have relatively stable performance under low temperatures. In contrast, the 1.2-kV SiC MOSFETs all show degraded turn-on switching performances with a long-tail turn-on voltage.
3. GaN high-electron-mobility transistor (HEMT) has improved performance under CT with both conduction loss and switching loss reductions. The turn-on voltage time rate of change, dV/dt , is increased from 40 to 80 V/ns.
4. Conventional ferrite core has degraded performance at low temperature with high core loss. The Al electrolytic capacitor does not work properly and the capacitance drops significantly.
5. Due to the superior performance of GaN HEMT, more investigations were made.

The overcurrent test for GaN HEMT was performed, and results demonstrated that the evaluated GaN HEMT can withstand 250 A overcurrent at CT, whereas this value is only 160 A at RT. A GaN HEMT-based DC-DC converter was also investigated under different conditions. The results demonstrated that system efficiency can be improved under CT.

3.2.1.1 Experimental Methods

3.2.1.1.1 Static Characterization

Fig. 151 shows the static characterization setup. LN₂ is used to achieve CT. The designed printed circuit board (PCB) is placed inside of the cryogenic chamber and connected with the test fixture through the front panel. To mitigate the measurement errors caused by the long connection cables from the test fixture to the cryogenic chamber, the Kelvin connection (as illustrated in Fig. 151) is adopted. Another advantage of the designed PCB is that it can enable multiple device characterizations by simply modifying the connections on the front panel, which saves a great amount of time and energy.

3.2.1.1.2 Dynamic Characterization

To achieve the dynamic characterizations (switching loss evaluation and dynamic $R_{ds(on)}$), a double pulse test (DPT) setup as shown in Fig. 152 is used. Fig. 153 shows the system setup for the dynamic characterization. The load inductor is placed outside of the chamber while the gate driver (GD) and DPT board are placed inside to evaluate the performance of power devices under CTs. Fig. 154 shows the details of the designed DPT board and measurement methods. The gate-to-source voltage is measured by using a 200-MHz passive voltage probe (Tektronix P2220), the drain-to-source voltage is also measured by using a 200-MHz passive voltage probe (Tektronix P5120), and the device current is monitored by using a 1200-MHz current shunt resistor (T&M Research Products, SSDN-015). BNC tip adaptors are adopted for the gate-to-source and drain-to-source voltage measurements to reduce the parasitics introduced by the measurement. The differential voltage probes and current probes cannot work properly under low temperature due to the additional circuits, which are not cryogenic friendly, whereas the passive voltage probe and current shunt resistor are appropriate for low-temperature measurements with high bandwidth. To achieve the on-state voltage measurement, a clamping circuit as shown in Fig. 155 is used. In the off-state, the voltage across the Zener diode is clamped at 3.3 V; in the on-state, the voltage across the Zener diode equals to the sum of the device voltage and SiC Schottky diode forward voltage drop.

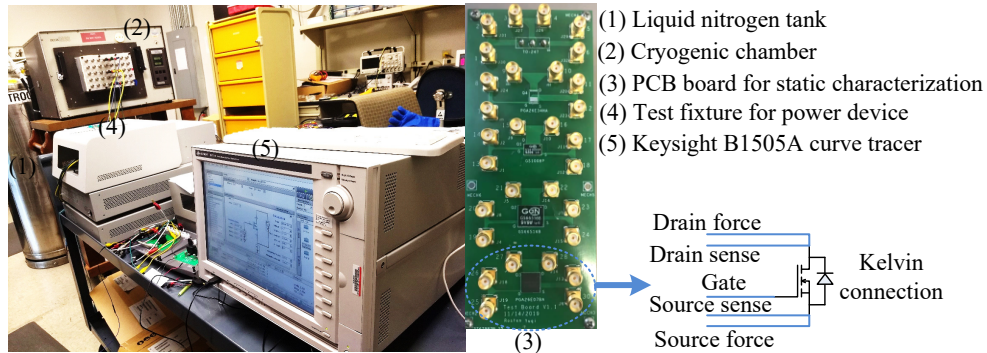


Fig. 151.—Static characterization setup.

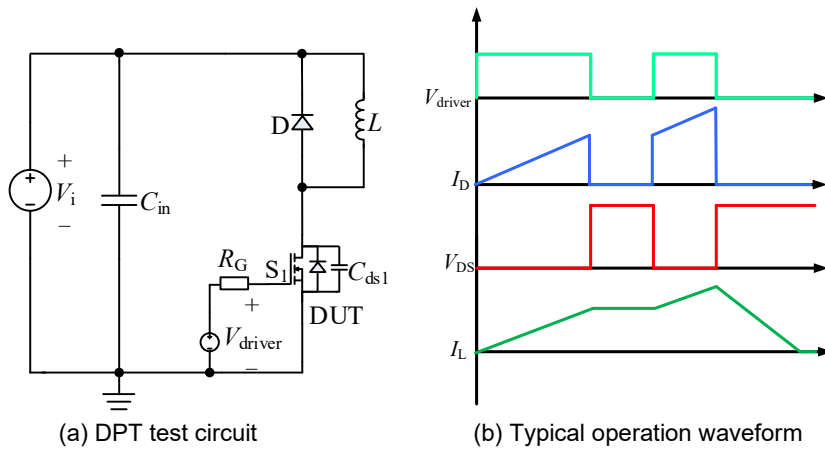


Fig. 152.—DPT test circuit and its typical operation waveform.

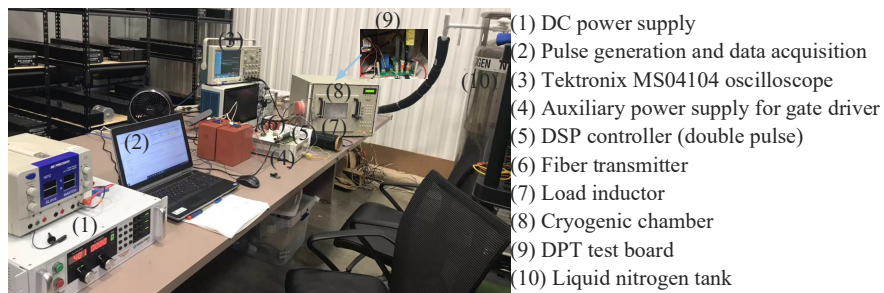


Fig. 153.—Dynamic characterization setup.

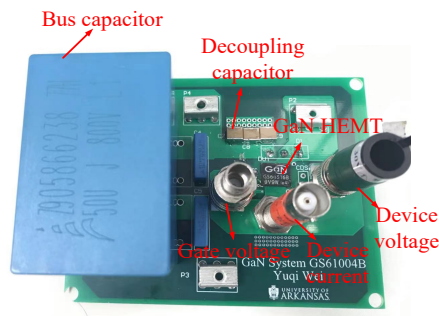


Fig. 154.—Designed DPT test board.

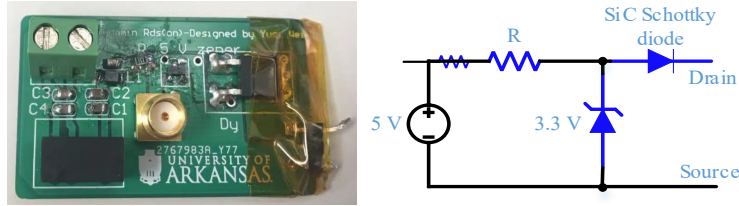


Fig. 155.—Clamping circuit for cryogenic dynamic $R_{ds(on)}$ evaluation.

TABLE 24.—SELECTED COMMERCIAL SEMICONDUCTORS

Part number	Manufacturer	Material and structure	Rated voltage/current	Note	Symbol
IXFH26N50P	IXYS	Si MOSFET	500 V/26 A	Static and dynamic	Si MOSFET 1
IPW65R041CFD	Infineon	Si CoolMos	650 V/68 A	Static	Si MOSFET 2
FGA5065ADF	ON Semiconductor	Si IGBT	650 V/50 A	Static and dynamic	Si IGBT 1
IGW40T120	Infineon	Si IGBT	1200 V/40 A	Dynamic	Si IGBT 2
SCT3120AL	Rohm	SiC trench MOSFET	650 V/21 A	Static and dynamic	SiC MOSFET 1
SCT3022AL	Rohm	SiC trench MOSFET	650 V/93 A	Static	SiC MOSFET 2
C3M0120090D	Wolfspeed	SiC planar MOSFET	900 V/23 A	Dynamic	SiC MOSFET 3
MSA12N080A	Monolith	SiC planar MOSFET	1200 V/20 A	Dynamic	SiC MOSFET 4
SCH2080KE	Rohm	SiC planar MOSFET	1200 V/40 A	Dynamic	SiC MOSFET 5
SCT50N120	STMicroelectronics	SiC planar MOSFET	1200 V/65 A	Dynamic	SiC MOSFET 6
C2M0025120	Wolfspeed	SiC planar MOSFET	1200 V/63 A	Dynamic	SiC MOSFET 7
TP65H035WSQA	Transphorm	Cascode GaN	650 V/47.2 A	Static and dynamic	Cascode GaN 1
TP90H050WS	Transphorm	Cascode GaN	900 V/34 A	Static	Cascode GaN 2
GS66504B	GaN System	GaN HEMT	650 V/15 A	Static and dynamic	GaN HEMT 1
GS66516B	GaN System	GaN HEMT	650 V/60 A	Static and dynamic	GaN HEMT 2
GPIHV30SB5L	GaN Power	GaN HEMT	1200 V/30 A	Static	GaN HEMT 3

3.2.1.2 Evaluation of Selected Semiconductors: Results and Discussion

Semiconductor devices of different materials from different manufacturers were selected to evaluate their performances under low temperatures. A series of semiconductors were selected and characterized, including the state-of-the-art 900-V Cascode GaN field-effect transistor (FET) and 1200-V GaN HEMT devices.

Table 24 provides a summary of the semiconductors selected for evaluation. Results and discussion follow here.

3.2.1.2.1 On-State Resistance Comparisons

The normalized on-state resistance is defined as the ratio between the on-state resistance value at a given temperature and the RT on-state resistance value. Fig. 156 shows the normalized on-state resistance comparison results of different semiconductors; the base value at RT is labeled in the legend. The SiC MOSFETs' on-state resistances increase dramatically with the decrease of temperature; the on-state resistance value at 93 K is around 2 to 3 times larger than the RT values, while all other devices show a decreasing trend with the decrease of temperature. Among them, the Si MOSFET IXFH26N50P demonstrates the largest on-state resistance reduction, which is around 83% reduction at CT. Meanwhile, the GaN HEMT GS66504B also shows a large on-state resistance reduction under CT; the on-state resistance value at CT is around one-third of the RT value.

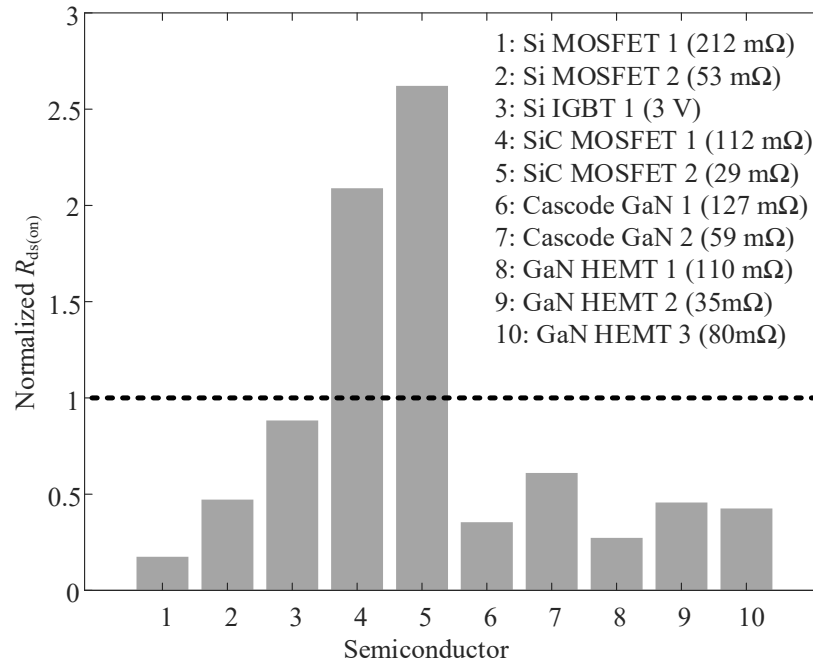


Fig. 156.—Normalized on-state resistance comparisons for different semiconductors.

The on-state resistance reductions for the Si MOSFET and Si IGBT are caused by the increased accumulation, inversion, and bulk carrier mobility. The bulk mobility of the junction-gate field-effect transistor (JFET) region and the accumulation layer mobility just below the gate is responsible for the overall on-state resistance. Since the temperature is well above the carrier freezeout, the dominant temperature-dependent parameter is the mobility. The increased electron mobility under low temperatures leads to the on-state resistance reduction for GaN HEMT, given that the channel resistance is inversely proportional to the electron mobility. The mobility is governed by ionized impurity due to interface charge and a crystal defect scattering, which is the dominant scattering mechanism at lower temperatures. The Cascode GaN FET is composed of a low-voltage Si power MOSFET and depletion mode GaN HEMT, which also sees on-state resistance drop under low temperature. In contrast, the SiC MOSFET shows an increased on-state resistance at CT because large amounts of electrons are trapped and few free electrons are available for conduction of the inversion layer of the interface.

3.2.1.2.2 Threshold Voltage Comparisons

Fig. 157 shows the comparison results of the normalized threshold voltage. Aside from the GaN system devices, all the other power devices witness an increase of the threshold voltage under CT. The threshold voltage increases for the Si MOSFET, SiC MOSFET, and Si IGBT can be explained by the reduction of intrinsic carrier concentration. The temperature-dependent parameters in threshold voltage are intrinsic carrier concentration and the bandgap. At lower intrinsic carrier concentration, more gate voltage is required to compensate for the increased band bending, which in turn increases the threshold voltage. For the Cascode GaN FET, the threshold voltage is determined by the low-voltage Si power MOSFET, which also increases with the decrease of temperature. However, of the evaluated three GaN HEMT devices, one demonstrates a negative temperature coefficient at low temperature, while two of them demonstrate positive temperature coefficient. The threshold voltage in single-chip GaN devices depends on the substrate used, the gating mechanism, the effect of deep-level defects, and traps. For

example, the positive temperature coefficient of threshold can be explained by the positive temperature coefficient of Schottky barrier height. Given that conduction band discontinuity and 2D electron gas carrier density are weak functions of temperature, the dominant contribution of the threshold variation possibly comes from the electron capture by the interface states or deep-level bulk traps. However, the causes for the different threshold voltage performances need further investigation.

3.2.1.2.3 Turn-On and Turn-Off Switching Loss Comparisons

Fig. 158 shows the normalized turn-on switching loss comparisons among different devices. The turn-on switching losses for the 1.2-kV SiC MOSFETs increase with the decrease of temperature due to the increased overlapping time between device voltage and current. For all other devices, the turn-on switching losses decrease with the decrease of temperature or remain almost unchanged. Among these other devices, the turn-on switching loss reductions for GaN HEMTs are larger than for other types of semiconductors due to the lower threshold voltage of these GaN HEMT devices.

Fig. 159 shows the normalized turn-off switching loss comparisons among different devices. Similarly, the turn-off switching losses for the 1.2-kV SiC MOSFETs increase with the decrease of temperature, while others either remain almost unchanged or decrease with the decrease of temperature. Si MOSFET and Si IGBT have larger turn-off switching loss reductions than the other types of semiconductors.

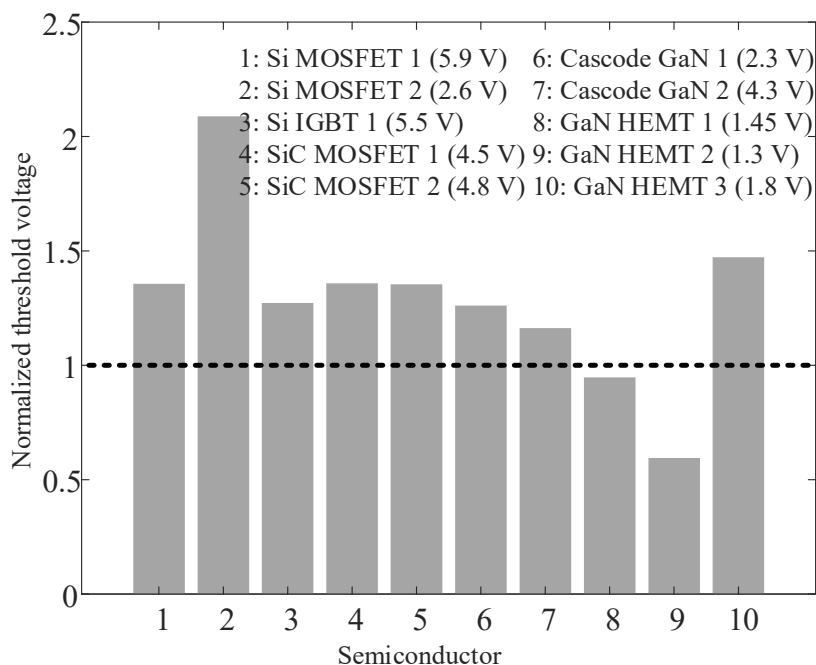


Fig. 157.—Normalized threshold voltage comparisons for different semiconductors.

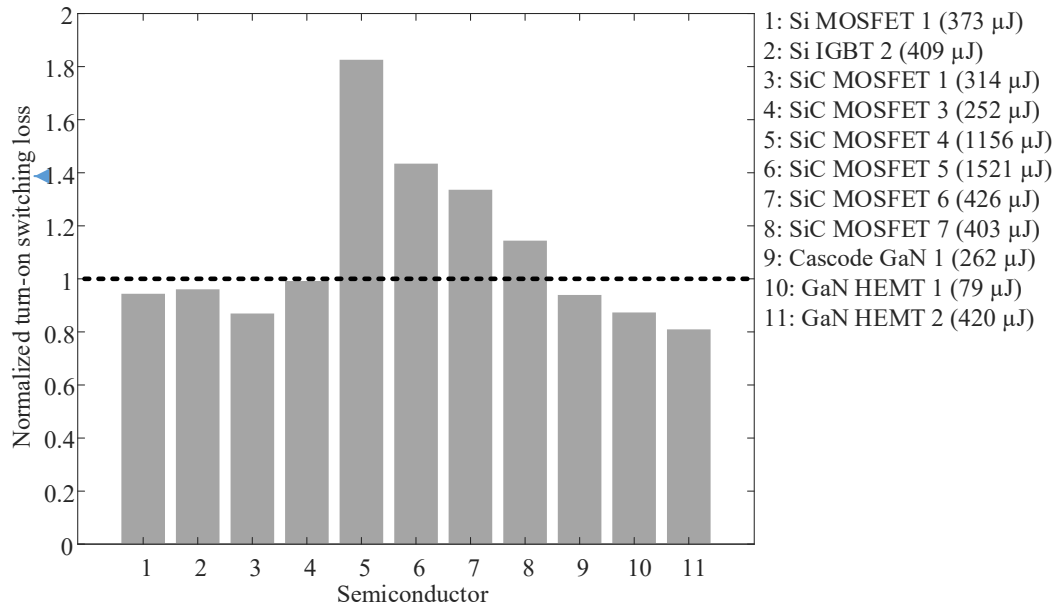


Fig. 158.—Normalized turn-on switching loss comparisons for different semiconductors.

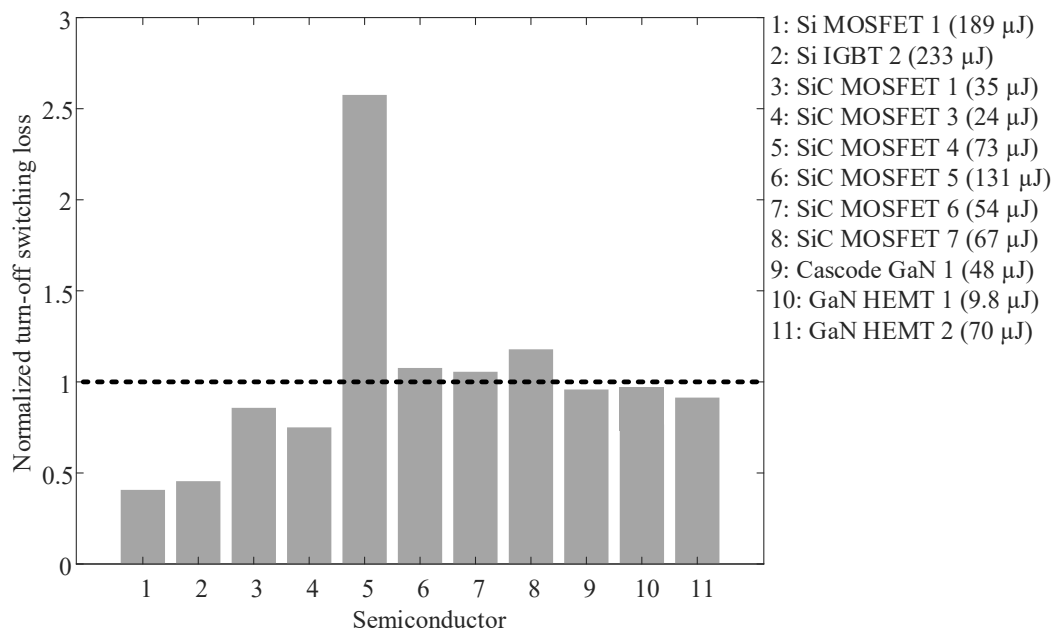


Fig. 159.—Normalized turn-off switching loss comparisons for different semiconductors

The following conclusions can be drawn based on the characterization results shown in Fig. 157 to Fig. 159:

- The traditional Si MOSFET and Si IGBT are advantageous at cryogenic applications due to the reductions of both conduction loss and switching loss.
- SiC MOSFET is not favored for cryogenic applications due to the increase of on-state resistance. In addition, the switching processes for the 1.2-kV SiC MOSFETs are significantly degraded, which leads to the increase of switching loss.
- Cascode GaN and GaN HEMT are advantageous for cryogenic applications.

3.2.1.3 Key Performances for Various Semiconductors

In this section, some key performances for different semiconductors are presented and discussed, which demonstrates the features of different semiconductors.

3.2.1.3.1 Si MOSFET (IXFH26N50P)

Fig. 160 compares turn-on and turn-off switching processes at 143 K and RT for Si MOSFET. The switching process is determined by three factors: (1) threshold voltage, (2) inversion carrier mobility, and (3) gate drain capacitance. For the turn-on process, the threshold voltage of the device is increased at low temperature, which slows down the turn-on process. The inversion carrier mobility is increased significantly at low temperature, but this speeds up the turn-on process. Thus, the overall turn-on performance is almost unchanged. In contrast, the turn-off switching process is improved significantly due to the increase of threshold voltage and inversion carrier mobility and the slight reduction of gate drain capacitance. Approximately 6% turn-on switching loss reduction and 60% turn-off switching loss reduction is achieved at 143 K when compared to RT.

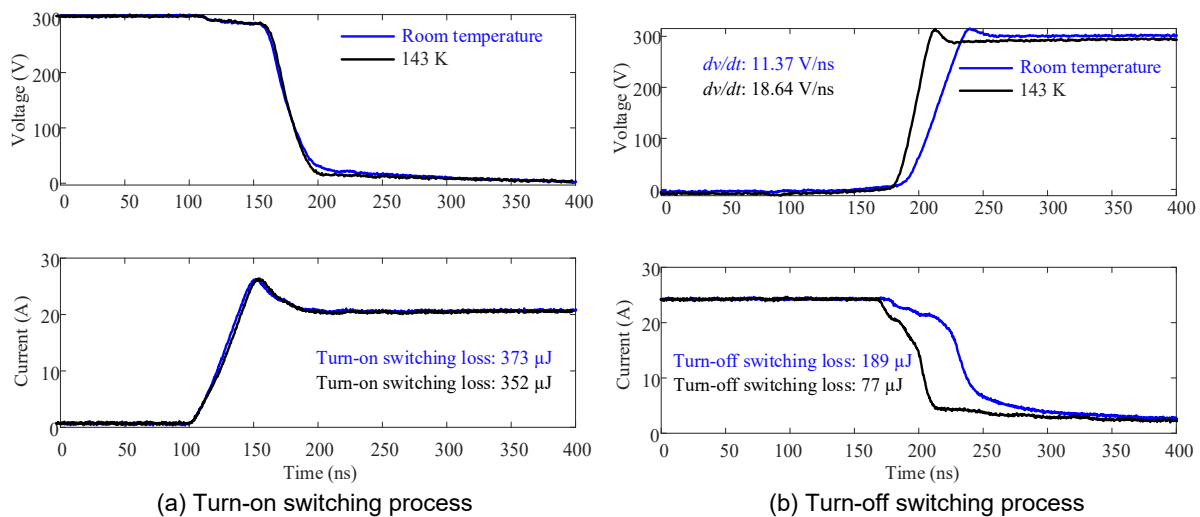


Fig. 160.—Switching process comparisons for Si MOSFET.

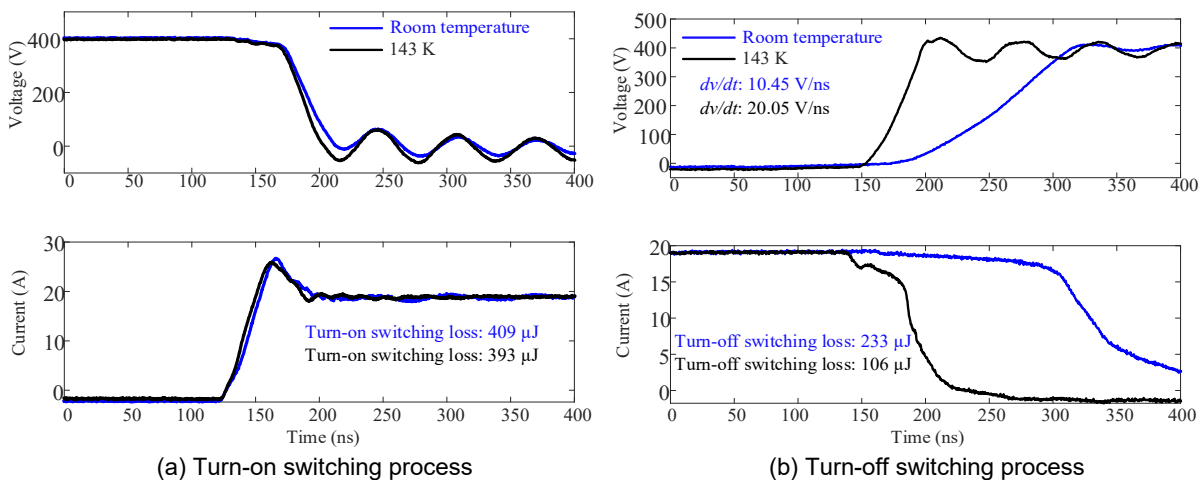


Fig. 161.—Switching process comparisons for Si IGBT.

3.2.1.3.2 Si IGBT (IGW40T120)

Fig. 161 compares switching processes for Si IGBT at RT and 143 K. As with Si MOSFET, the turn-on switching process remains almost unchanged. The turn-off switching process is improved significantly due to the increase of threshold voltage and inversion carrier mobility. The gain of the internal P-N-P transistor decreases with lower temperatures, which in turn helps obtain a larger initial drop in collector current. This is mainly due to the reduction of minority carrier lifetime in N+ and P+ silicon. The turn-off dV/dt is almost doubled at 143 K when compared with RT. Approximately 55% turn-off switching loss reduction is achieved for the IGBT.

3.2.1.3.3 650-V SiC MOSFET (SCT3120AL)

Fig. 162 compares turn-on and turn-off switching processes at 143 K and RT for 650-V SiC MOSFET. The inversion carrier mobility for SiC MOSFET is reduced at low temperature and the threshold voltage is increased, which slows down the switching process. The gate drain capacitance is slightly reduced at low temperature to speed up the switching process. However, as can be seen from Fig. 162(a), the turn-on process can be divided into two stages. For the first stage, the voltage drops faster at CT when compared with RT. For the second stage, the device voltage drops slowly to fully turn-on voltage in this stage. Overall, both the turn-on and turn-off switching processes remain almost unchanged.

Fig. 163 compares turn-on and turn-off switching processes at 143 K and RT for 900-V SiC MOSFET. As with 650-V SiC MOSFET, both the turn-on and turn-off switching processes remain almost unchanged.

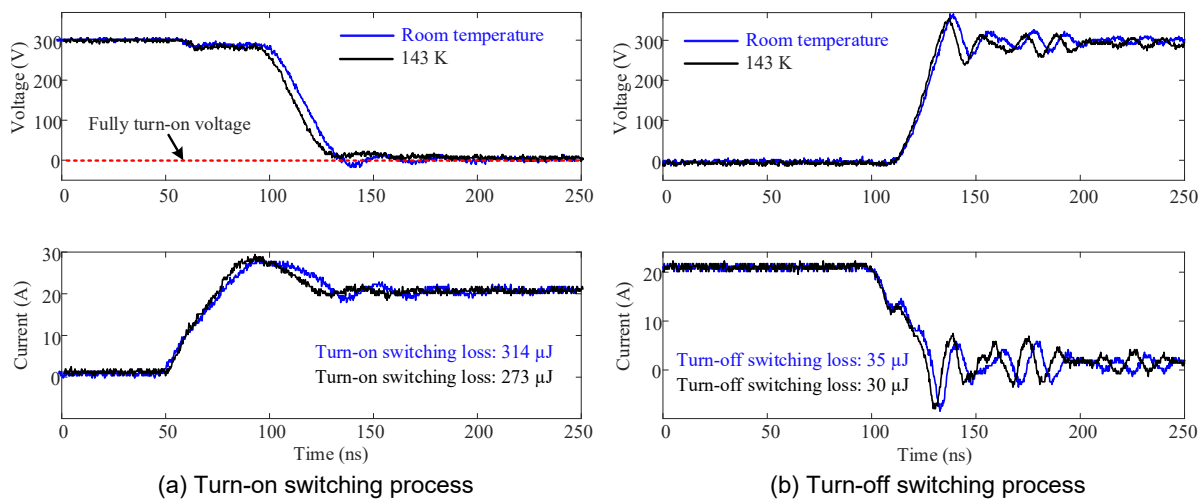


Fig. 162.—Switching process comparisons for 650-V SiC MOSFET.

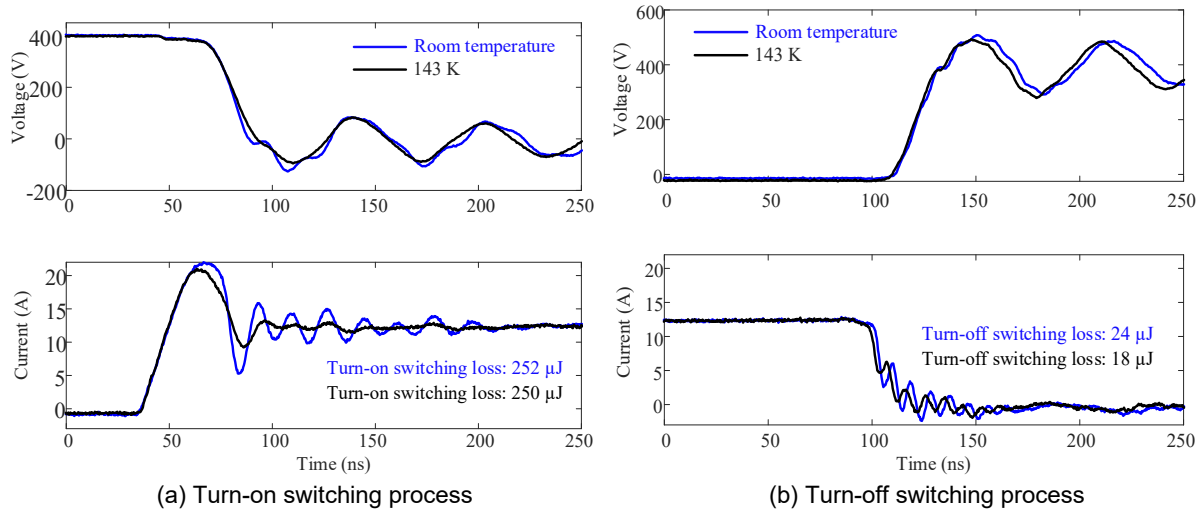


Fig. 163.—Switching process comparisons for 900-V SiC MOSFET.

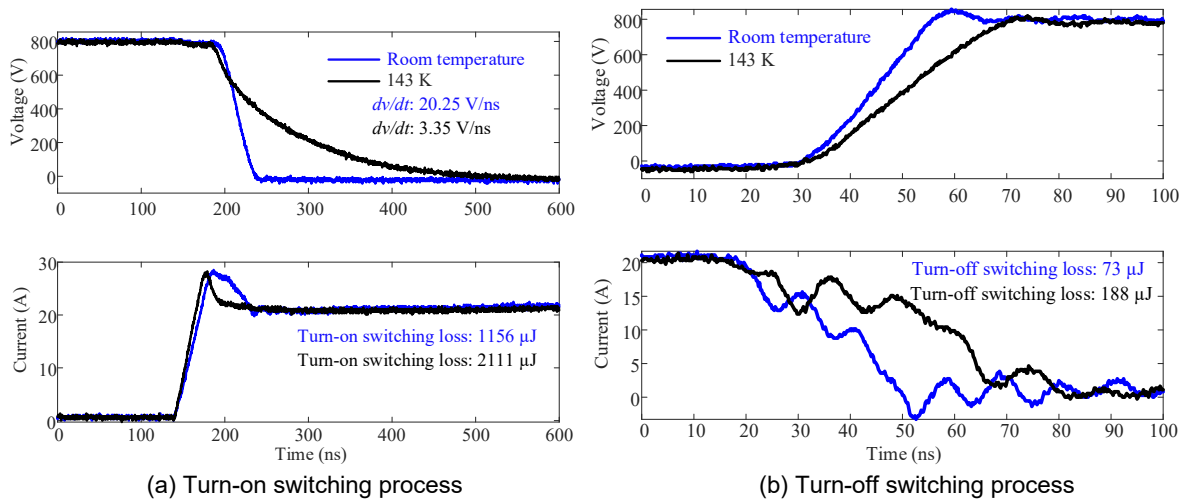


Fig. 164.—Switching process comparisons for 1.2-kV SiC MOSFET.

3.2.1.3.4 1.2-kV SiC MOSFET (MSA12N080A)

Fig. 164 compares turn-on and turn-off switching processes at 143 K and RT for 1.2-kV SiC MOSFET. Unlike 650-V and 900-V SiC MOSFETs, the 1.2-kV SiC MOSFET demonstrates a long-tail turn-on voltage, which increases the turn-on switching loss dramatically (83%). Meanwhile, the turn-on dV/dt also drops from 20.25 V/ns at RT to 3.35 V/ns at 143 K. Moreover, the turn-off switching process is also degraded at low temperature, with a switching loss is increased by 158% at low temperature. Fig. 165 shows the dynamic on-state resistance performance comparison at RT and 143 K by using the clamping circuit as shown in Fig. 155. The dynamic on-state resistance at 143 K takes around 1.5 μ s to reach the static on-state resistance. The degraded switching performance is mainly attributable to the reduction of inversion carrier mobility. Variations in performance between 650-, 900-, and 1200-V SiC MOSFETs are caused by differences in the device structures.

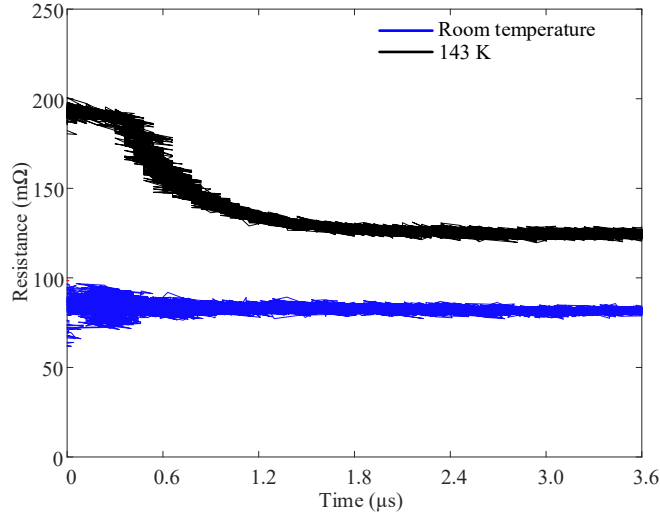


Fig. 165.—Dynamic on-state resistance comparison between RT and 143 K.

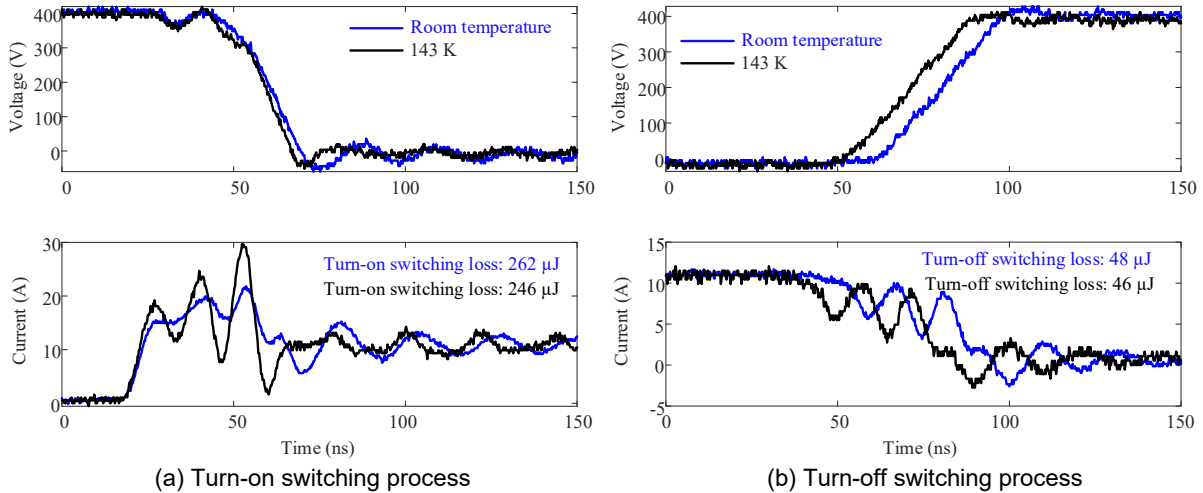


Fig. 166.—Switching process comparisons for 650-V Cascode GaN FET.

3.2.1.3.5 Cascode GaN FET (TP65H035WSQA)

Fig. 166 compares turn-on and turn-off switching processes at 143 K and RT for Cascode GaN FET. For the Cascode GaN device, the switching performance is determined by the low-voltage Si MOSFET. Thus, its performance is similar to that of Si MOSFET, where turn-on switching performance remains almost unchanged while turn-off switching performance is improved.

3.2.1.3.6 GaN HEMT (GS66504B)

Fig. 167 compares turn-on and turn-off switching processes at 143 K and RT for GaN HEMT 1. The turn-on switching process is improved at 143 K. The improved switching performance for GaN HEMT is mainly attributable to the increase of electron mobility. However, the turn-off switching process remains almost unchanged, which can be explained by the reduction of threshold voltage at low temperature (it slows down the turn-off process). Fig. 168 shows the dynamic $R_{ds(on)}$ performance over the temperature. Significant reduction of dynamic $R_{ds(on)}$ is also observed at low temperatures.

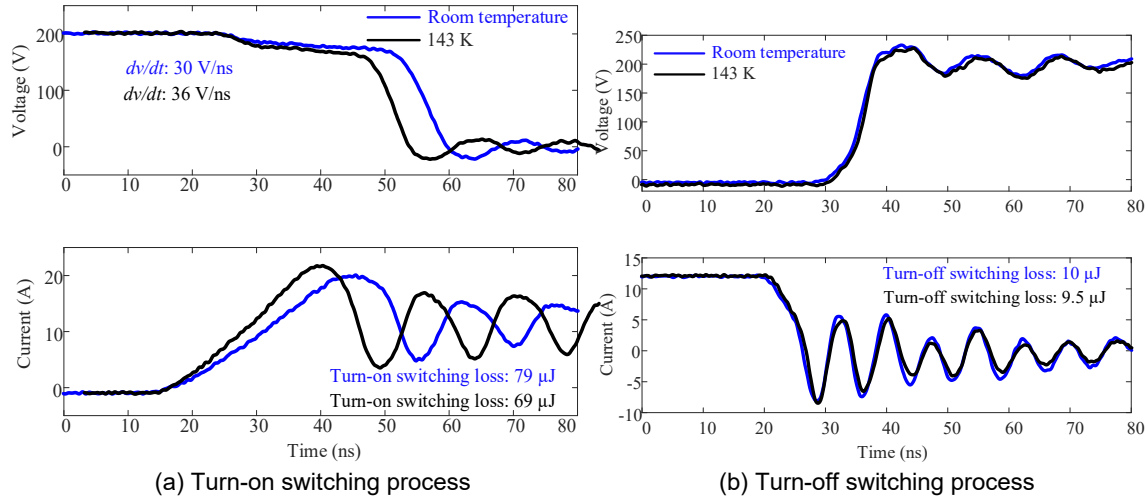


Fig. 167.—Switching process comparisons for 650-V GaN HEMT1.

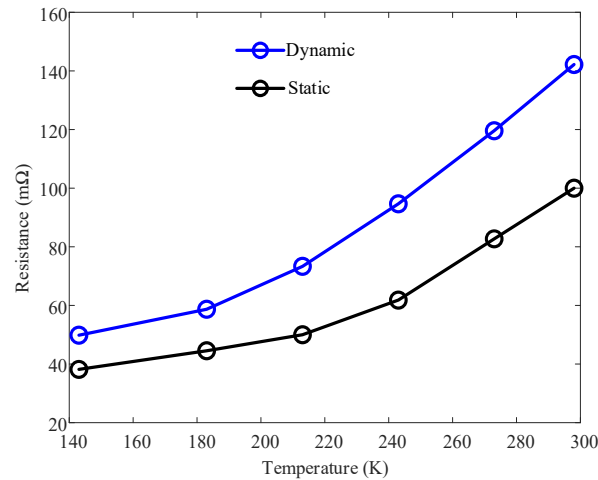


Fig. 168.—Dynamic $R_{ds(on)}$ performance at low temperatures for GaN HEMT1.

3.2.1.3.7 GaN HEMT (GS66516B)

Fig. 169 compares turn-on and turn-off switching processes at 143 K and RT for GaN HEMT 2. The turn-on time is almost halved, as shown in Fig. 169(a). The turn-on dV/dt is increased from 47 V/ns at RT to 82 V/ns at 143 K. Meanwhile, the turn-on dI/dt also increases from 2.69 A/ns at RT to 4.72 A/ns at 143 K. Due to the reduction in turn-on time, the turn-on switching loss for the GaN HEMT is also reduced under low temperature operations (19% reduction). The peak current also increases from 42.14 A at RT to 51.47 A at 143 K due to the fast switching speed at low temperature. Fig. 170 demonstrates the dynamic $R_{ds(on)}$ performance over the temperature. Significant reduction of the dynamic $R_{ds(on)}$ is also observed at low temperatures.

Nevertheless, the high dV/dt and dI/dt should be carefully considered when designing the power converter for low-temperature operations, which poses challenges for the GD and PCB design.

In summary, all the semiconductors except the 1.2-kV SiC MOSFETs demonstrate superior dynamic performances at low temperature when compared with RT. The turn-off switching processes for the Si MOSFET and Si IGBT are improved significantly, and the turn-on switching process for the GaN HEMT is improved greatly. In addition to the reduction in static on-state resistances at CT, the dynamic on-state resistances of the GaN HEMTs were also reduced at low temperatures.

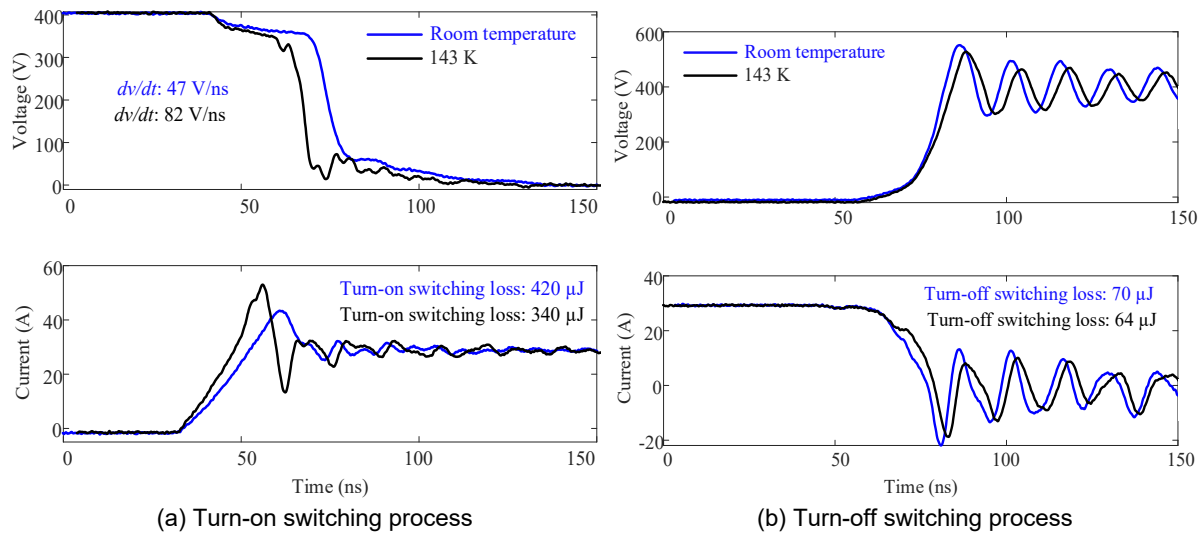


Fig. 169.—Switching process comparisons for 650-V GaN HEMT 2.

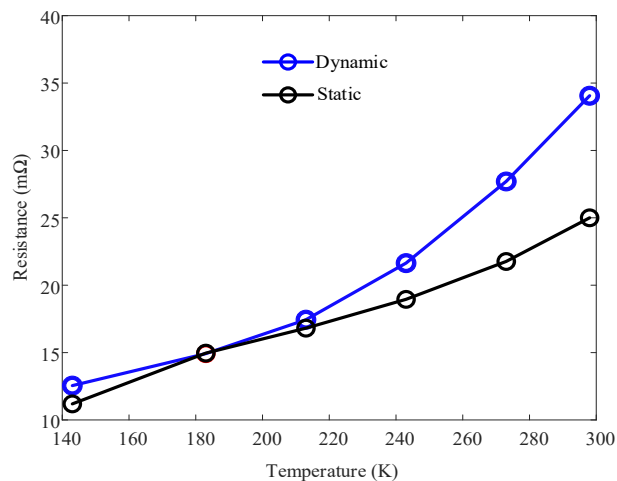


Fig. 170.—Dynamic $R_{ds(on)}$ performance at low temperatures for GaN HEMT 2.

3.2.1.4 Future Work

Based on the evaluation results and literature survey, the future work for semiconductor cryogenic characterizations can be summarized as described here.

3.2.1.4.1 Cryogenic-Friendly Gate Driver

As will be discussed in Section 3.2.2.1, most commercial GDs would not work at CTs, which limits the minimum evaluation temperature for the semiconductors. Research and development on cryogenic-friendly GDs could enable lower-temperature evaluations of semiconductors and also the future safe operation of power converters.

3.2.1.4.2 Comprehensive Characterizations of GaN HEMT

Based on our characterization results and existing literature, we can conclude that GaN HEMT is a very promising candidate for cryogenic applications. However, the understanding of GaN HEMT at CTs is not comprehensive. For example, the threshold voltage and dynamic on-state resistance under CTs are not well explored or explained.

3.2.1.4.3 Reliability of Semiconductors at Cryogenic Temperature

The existing literature focuses on the basic characterizations of semiconductors at CT. However, long-term reliability issues are of great importance for cryogenic power converters. The failure mechanisms of semiconductors at CT should also be investigated.

3.2.2 GaN HEMT Overcurrent Test and Converter Test

The following sections describe the overcurrent test and four-switch buck-boost converter test. The overcurrent test demonstrates that with cryogenic cooling, the GaN HEMT can be operated with much higher current. The maximum current with LN₂ is around 250 A, which corresponds to the saturation current of the device at CT. The experimental waveforms for the overcurrent test are explained. The reduction of saturation current at second pulse is explained by junction temperature rise and off-state electron trapping. Compared with the device RT test, the maximum current is increased from 160 to 250 A, which is contributed by a low junction temperature with cryogenic cooling. The converter test results demonstrate high-efficiency operation at CT due to the reductions of device conduction loss and switching loss.

3.2.2.1 Overcurrent Test

The key elements for a power converter are (1) semiconductor, (2) gate driver (GD), (3) capacitor, and (4) inductor. Based on the characterization results and literature survey, the following conclusions can be drawn:

- A GaN HEMT device can work even down to 4 K with improved performance.
- Designing a GD for CTs is difficult. Most GDs do not function below ~140 K.
- The Al electrolytic capacitor is not suitable for cryogenic applications, although the film capacitor has stable performance under CT.
- A magnetic core has degraded performance at CT.

To maximize the benefits of a GaN HEMT device, a cryogenic power electronics structure as shown in Fig. 171 was proposed. The top-cooled GaN HEMT device (GS66516T) from GaN Systems was selected, which was attached to the cold plate so that the operating temperature was close to the CT. The GD was placed on top of the device so that the operating temperature for the GD was not low. The film capacitor and decoupling capacitor (C0G/NP0 (negative-positive-zero)) were used. Air-core magnetic inductors were used to ensure high-efficiency operation. Fig. 172 shows the DPT setup, which was utilized for the dynamic characterization of the device.

Next, the overcurrent test for the selected GaN HEMT device was conducted, where the first pulse was increased to increase the current flows through the bottom device. Fig. 173 shows the overcurrent test results with 130, 180, and 230 A, respectively. These results demonstrated full operation of the associated GaN HEMT device at CT.

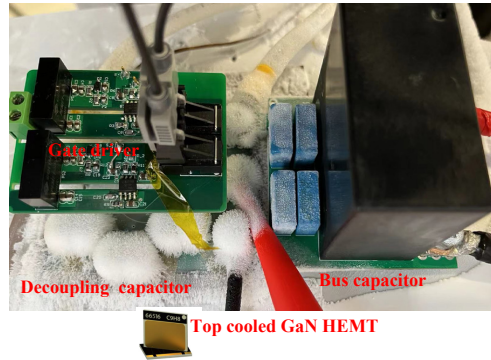


Fig. 171.—Cryogenic power converter structure.

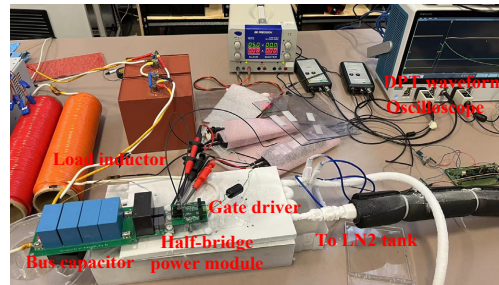


Fig. 172.—Cryogenic DPT test setup.

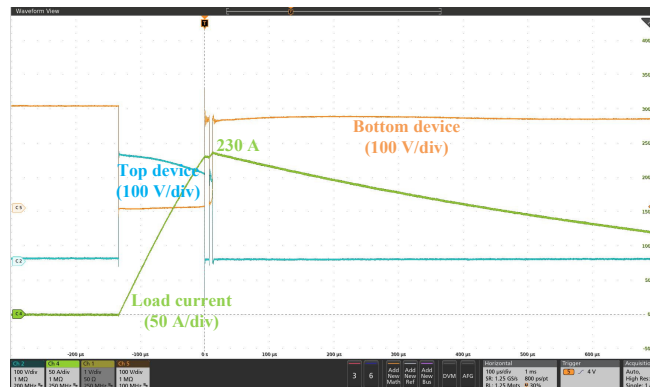
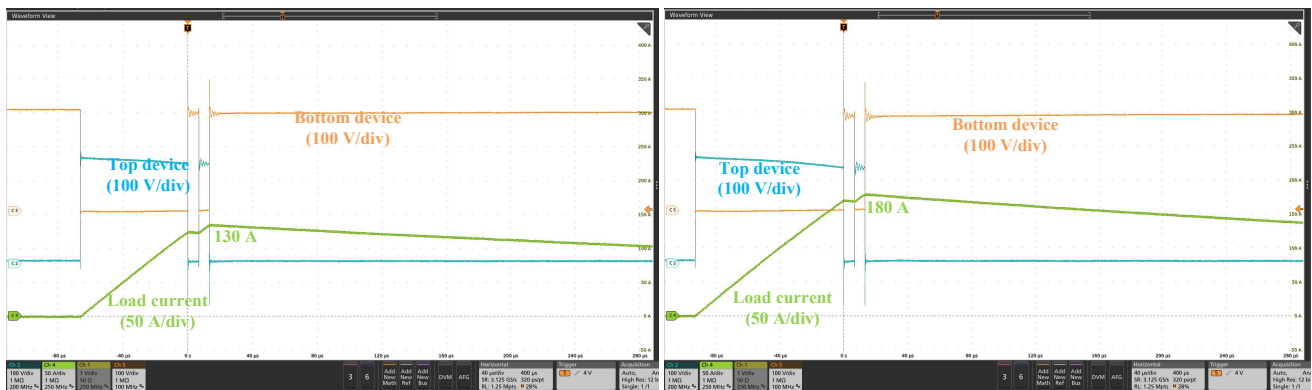


Fig. 173.—Overcurrent test for the 60-A GaN HEMT at CT.

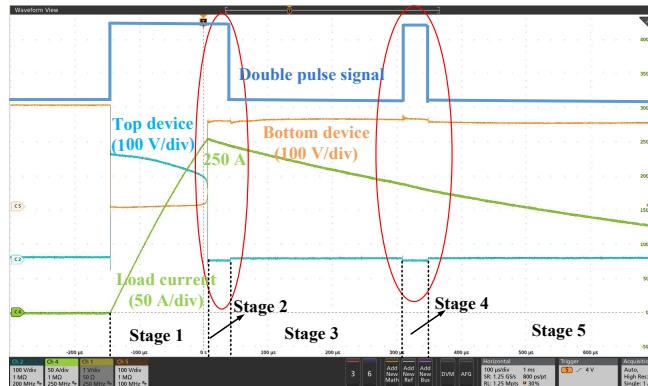


Fig. 174.—Overcurrent test 1.

When the first pulse is further increased and the device current reaches 250 A, the experimental waveform as shown in Fig. 174 is overserved; detailed explanations are given here.

Stage 1: During this stage, by turning on the bottom device, the load inductor is charged up. This stage ends when the device current reaches 250 A. During this stage, the bottom device is operating in the linear region and the top device is in off state.

Stage 2: Although the device gate voltage is still high, the device voltage increases to almost DC bus voltage. The reason is that at 250 A, the device reaches the saturation level and the device voltage starts increasing dramatically. During this period, a great amount of power loss is generated since the device voltage and device current are large. Thus, the junction temperature rises significantly due to the self-heating effect, and the current flow to the bottom device starts decreasing quickly. However, the total load current should remain constant. Thus, a portion of the load current flows to the top device to charge its output capacitor. When the reverse conduction criteria is satisfied, the top device is operating in reverse conduction mode. Due to the large impedance of the bottom device operating in the saturation region, the majority of the load current is freewheeling through the top device, which explains why the load current starts decreasing in this stage. It is confirmed from the top device voltage waveform in this stage that it is operating in reverse conduction mode with a higher voltage drop. In this stage, the bottom device is operating in the saturation region and the top device is operating in reverse conduction mode.

Stage 3: During this stage, the top device is turned on, and instead of reverse conduction operation mode, it is operating in the third quadrant. Since the device current is always lower than 250 A, the top device never enters into the saturation region. During this stage, the load current is freewheeling through the top device. The bottom device is in OFF state and the top device is operating in the third quadrant.

Stage 4: During this stage, the bottom device is turned on again. However, due to the excessive heat generated in Stage 2, the junction temperature of the bottom device is higher than the CT. Thus, the saturation current level is reduced when compared with the initial saturation current (250 A). Once the current reaches the new saturation level, Stage 2 operation is repeated. The bottom device is operating in the saturation region and the top device is operating in reverse conduction mode (which is also confirmed by the top device voltage waveform). The majority of the load current is still freewheeling through the top device. This also explains why the load current always decreases after Stage 1.

Stage 5: This stage is equal to Stage 3, where the bottom device is in OFF state and the top device is operating in the third quadrant.

Fig. 175 shows the experimental waveform with different pulse widths. It is clear that the device enters into the saturation region at 250 A. The inductor current is always decreasing given that when the

second pulse comes, the device is still operating in the saturation region. Fig. 176 shows the experimental waveform by increasing the off-state time so that the junction temperature of the device can be reduced and also the load current is decreasing during the off-state. It can be seen that the bottom device is operating in the linear region initially and that the load current starts increasing and the device voltage starts decreasing. However, when the load current reaches around 170 A, the device enters into saturation again since the junction temperature of the device is higher than the initial case (250 A).

If the off-state time is reduced, the theoretical saturation current level should be decreased since the junction temperature is higher. However, as shown in Fig. 177, by reducing the off-state time, the saturation current for the second pulse is 180 A, which is greater than the saturation current level with a long off-state time. Therefore, junction temperature alone cannot explain this phenomena. In addition to junction temperature, another important feature of GaN HEMT is the electron trapping mechanism. During the off-state, the high-voltage stress on the device will reduce the electron density, and the longer the off-state, the lower the electron density and the lower the saturation current. However, due to the time constant of the thermal network, the junction temperature reduction achieved by increasing the off-state is not obvious. Therefore, the electron trapping mechanism during the off-state is dominating and leads to the increase of saturation current, as shown in Fig. 177. A descriptive summary of the overcurrent test results are plotted in Fig. 178.

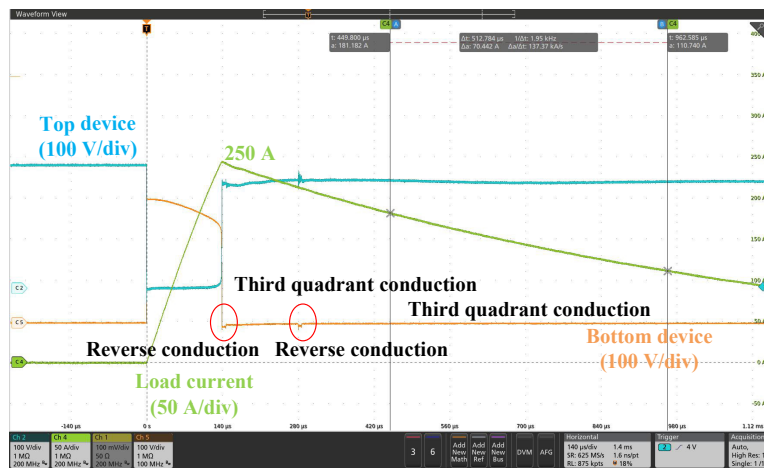


Fig. 175.—Overcurrent test 2.

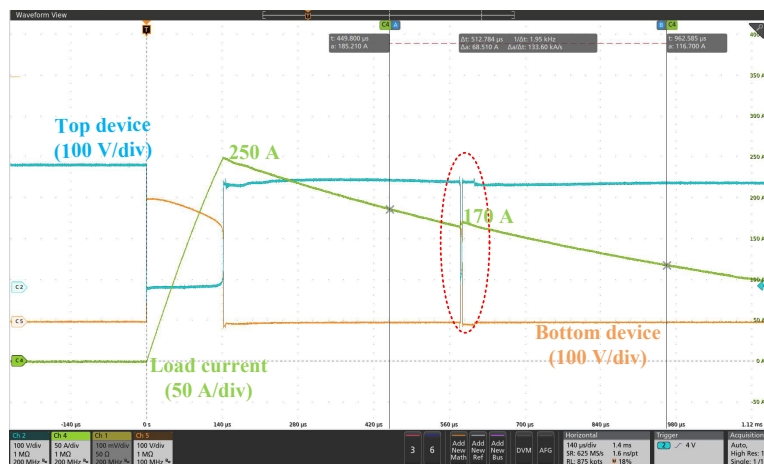


Fig. 176.—Overcurrent test 3.

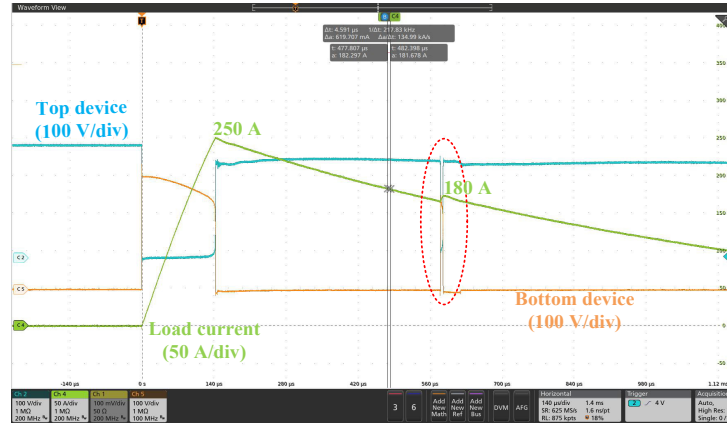


Fig. 177.—Overcurrent test 4.

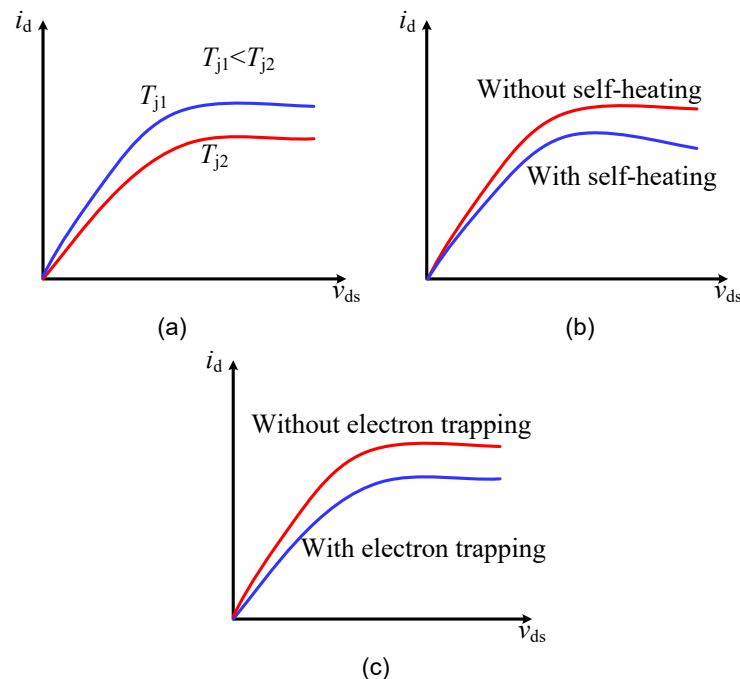


Fig. 178.—Operation mechanisms for the overcurrent test. (a) Junction temperature and saturation current. (b) Self-heating and saturation current. (c) Electron trapping and saturation current.

To make comparisons, the overcurrent test at RT is also performed. Fig. 179 shows the overcurrent test result at RT. The maximum current is around 160 A, which is lower than the value with cryogenic cooling.

3.2.2.2 Converter Test

In this section, converter test results under various conditions are reported. Fig. 180 shows the four-switch buck-boost converter topology. The reason for selecting this topology is that it has symmetric topology and bidirectional power flow capability. To generate the ± 270 -V DC bus voltage, the structure as shown in Fig. 181 is used. Based on the battery charging profile, the battery output voltage is 360 V/540 V (nominal)/605 V. For one converter unit, the output voltage is halved, 180 V/270 V (nominal)/302.5 V.

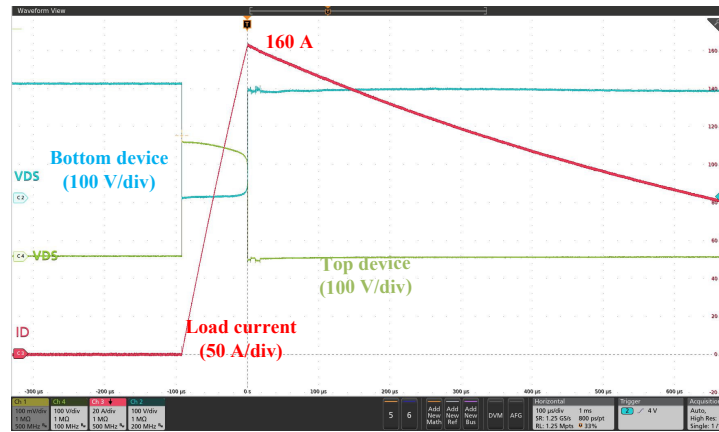


Fig. 179.—Overcurrent test at RT.

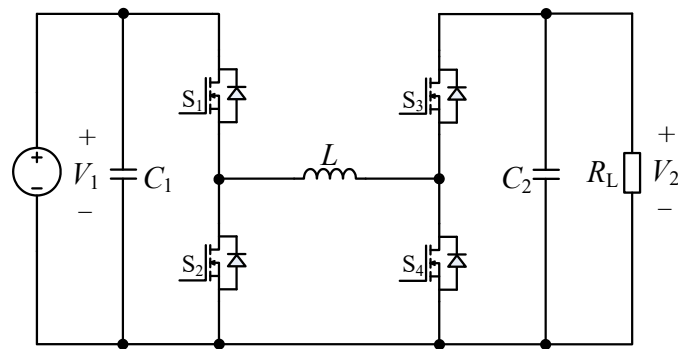


Fig. 180.—Topology of the four-switch buck-boost converter.

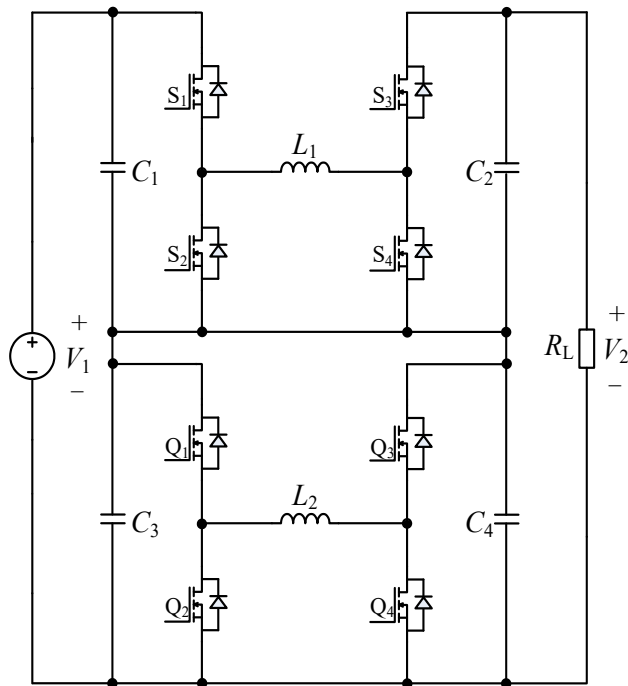
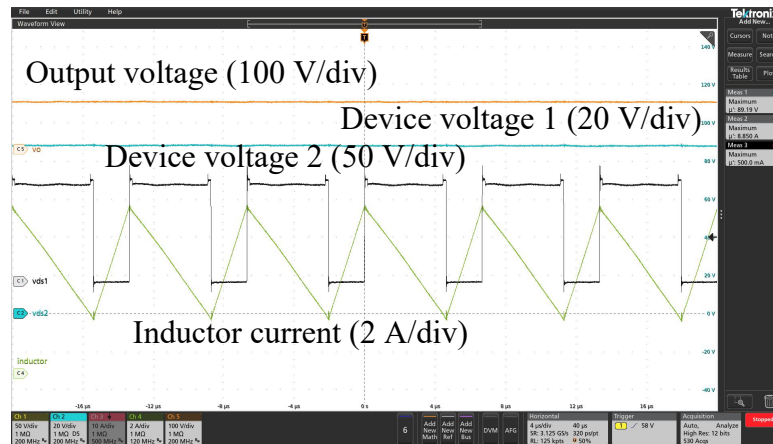
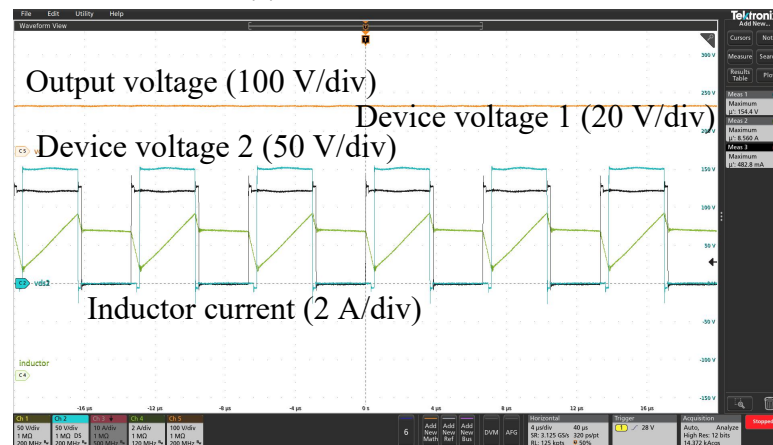


Fig. 181.—Structure to generate ± 270 -V DC bus voltage.

First, the converter is tested at lower power and lower input voltage. The load resistance is 30Ω and the input voltage is halved, 90 V/135 V (nominal)/150 V. The converter is operating in boost mode when the voltage is lower than 120 V. Otherwise, the converter is operating in buck-boost mode. Fig. 182 shows the typical experimental waveforms under two operation modes. Fig. 183 shows the efficiency comparisons between the RT and CT when the output power is around 500 W. The efficiency improvement is achieved at CT. Fig. 184 shows the efficiency performance at CT at 2.2 kW.



(a) Boost operation mode



(b) Buck-boost operation mode

Fig. 182.—Typical experimental waveforms in different operation modes.

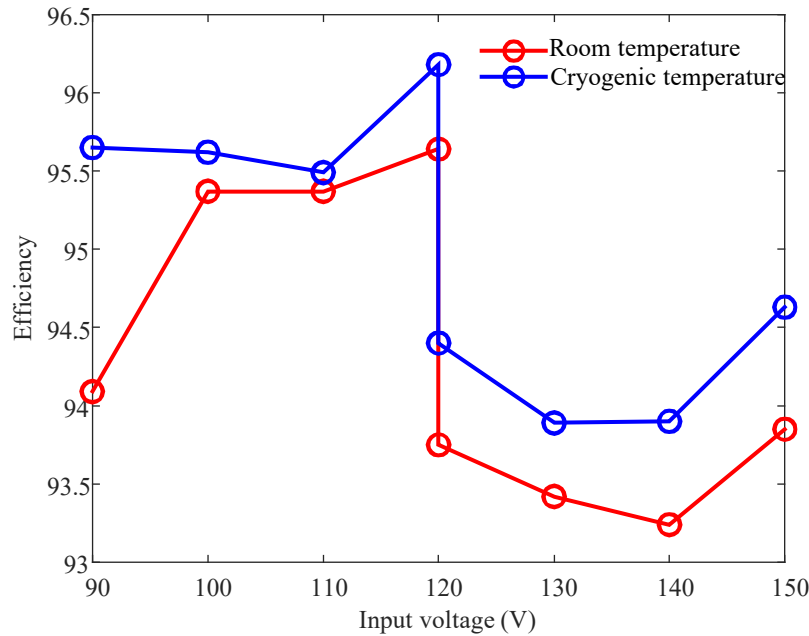


Fig. 183.—Efficiency comparisons at RT and CT when output power equals 500 W.

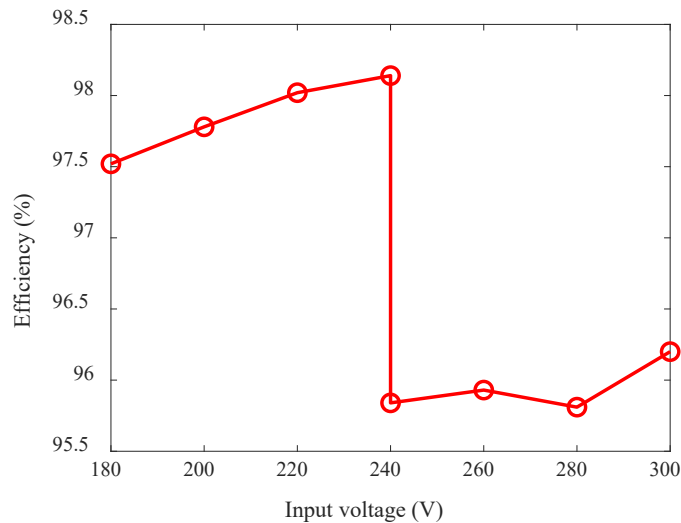


Fig. 184.—Cryogenic converter efficiency performance with 2.2-kW output.

A GaN HEMT-based 5-kW buck cryogenic converter was built and tested. Fig. 185 shows the experimental setup. The conventional magnetic core materials have degraded performance at CT. Thus, an air-core inductor was adopted for the buck converter. The steady-state temperature for the device and inductor winding was around 130 K. Fig. 186 shows the typical experimental waveform with input voltage of 100 V, load resistance of 3.3 Ω , and duty ratio of 0.5. The operating switching frequency was 150 kHz.

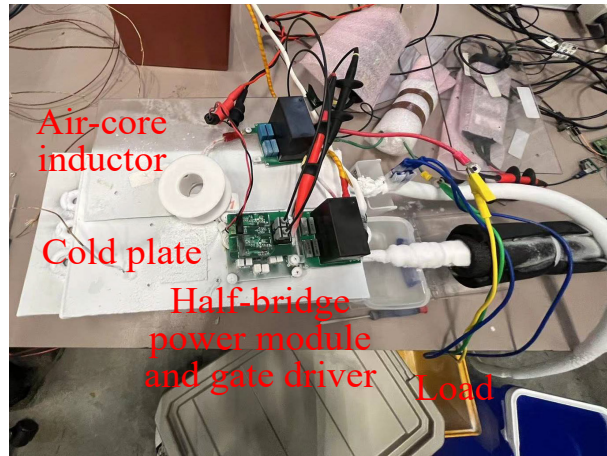


Fig. 185.—Experimental prototype of designed cryogenic buck converter.

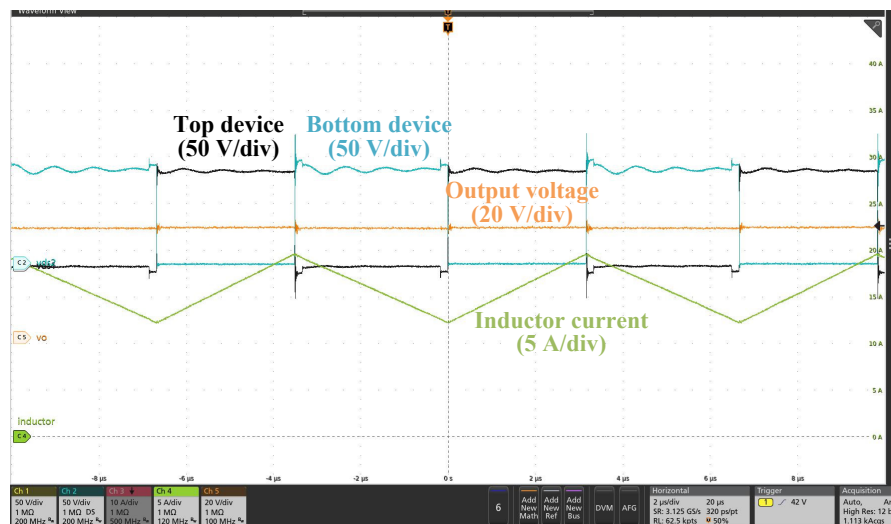


Fig. 186.—Typical buck converter experimental waveform.

Table 25 compares the switching loss for the GaN HEMT under different test conditions for a bus voltage of 200 V. The on-state resistance for the selected GaN HEMT at RT was 25 and 11 m Ω at 130 K. At RT, the DC resistance (DCR) for the inductor winding was 24 m Ω and the AC resistance (ACR) was 260 m Ω . Both DCR and ACR dropped significantly (around 65% reduction) at CT. Fig. 187 shows the measured efficiency comparisons between CT and RT. Around 1% efficiency improvement can be achieved at CT. Fig. 188 compares the loss breakdown for RT and CT for an output power of 2.5 kW. The top device has more loss than the bottom device since the bottom device can achieve soft-switching operation. Significant loss reduction was achieved for the top device due to the reductions in turn-on switching loss and conduction loss for GaN HEMT at CT. The inductor winding loss is dominating the converter power loss. At CT, the winding resistance was reduced significantly, which leads to the reduction of the inductor conduction loss. This is also the benefit of using air-core inductors for CT. Overall, the GaN HEMT and air-core-based power converter demonstrated improved performance for cryogenic applications.

TABLE 25.—SWITCHING LOSS COMPARISONS
FOR GaN HEMT UNDER RT AND CT

Temperature	Turn-on switching loss	Turn-off switching loss
298 K (15 A)	78.65	9.760
143 K (15 A)	66.93	20.250
298 K (30 A)	135.65	29.449
143 K (30 A)	110.30	35.612

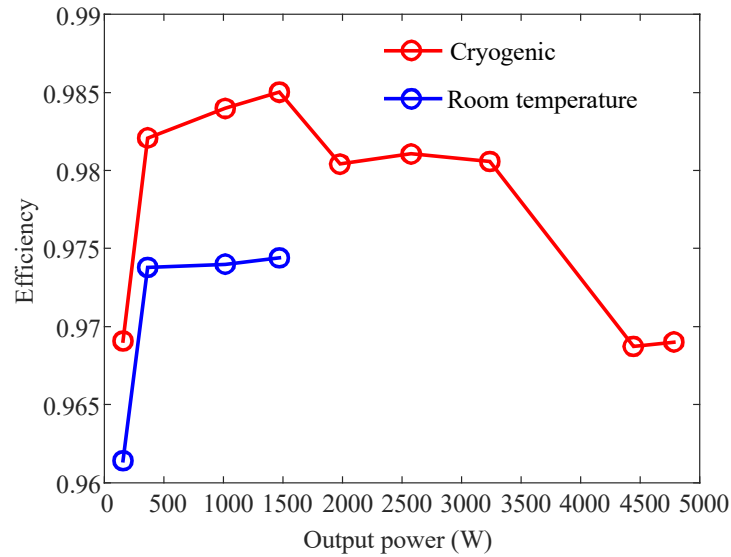


Fig. 187.—Measured efficiency performances under CT and RT.

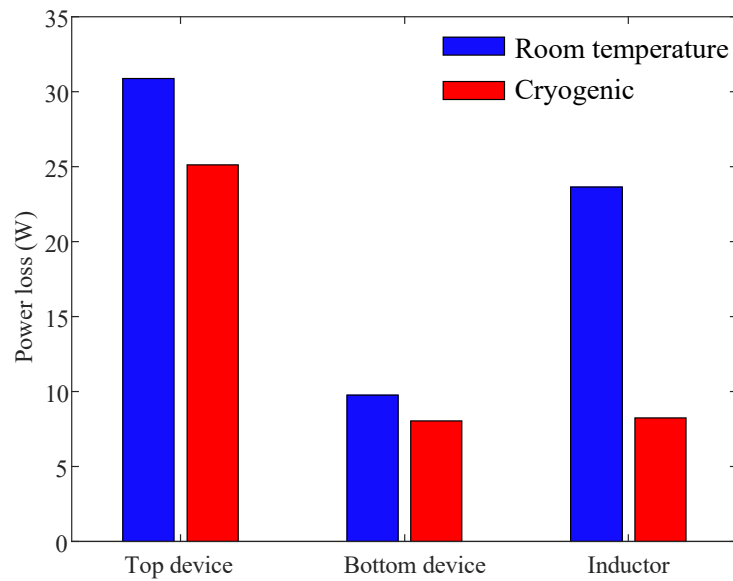


Fig. 188.—Theoretical loss breakdown analysis for the converter at CT and RT when output power equals 2.5 kW.

3.2.2.3 Future Work

We have investigated the overcurrent test and converter test at CT. Improved performances at CT for the GaN HEMT are observed. Future work on cryogenic power converter will look at the following aspects:

- Thermal system design: The current thermal system does not create a vacuum environment, which has icing issues and degrades the converter performance. Future work should include design of a complete thermal system.
- Thermal interface materials (TIM): Cryogenic characterizations of TIM should be made. The cryogenic characteristics of TIM are required to dissipate heat.
- Wire conduction loss: For power electronics applications, wire has AC loss and DC loss, the resistance drops with the decrease of temperature. However, there are many factors that influence the AC loss, such as operating switching frequency and wire construction. Investigations can be made on the wire conduction loss to optimize the air-core magnetics design.

3.2.3 Design and Validation of 20-kVA Cryogenic Current Source Inverter

This section presents the design of a cryogenic power electronics converter (CPEC) using wide bandgap (WBG) devices. A 20-kVA two-level current source inverter (2L-CSI) is designed and validated under both RT and CT of 77 K. CSIs have been neglected because of their physically large and heavy DC link inductor as compared to the DC link capacitor of voltage source inverters (VSIs). However, higher power, together with operation at CT in fully electric aircraft, makes CSIs superior in size, volume, and power quality. The advantage in volume is primarily attributed to utilization of air-core inductors with SC tapes as DC link storage elements. Therefore, a 2L-CSI with both power stage and associated components at CT is designed and evaluated. As part of converter development, commercial off-the-shelf (COTS) products including integrated circuits (ICs), isolated auxiliary power supplies (APSS), and passive components were utilized. Numerous laboratory prototypes of cryogenic subsystems were built to screen out the cryo-compatible components. Based on components screening, converter subsystems, namely the GD board and DPT platform, were also developed. Characterization of these subsystems was performed both at RT and CT, where both individual and integrated testing was carried out.

A cryogenic temperature (CT) is classified primarily as being below 123 K, where a cryogenic environment can be created with the help of LN₂ (77 K), LH₂ (33 K), or liquid helium (LHe₂, 4 K). Considering the increased applications of cryogenic systems in future commercial aircraft, power electronics converters are envisioned to play a significant role. These converters, combined with SC motors, magnetic energy storage modules, and cables, will enable reduced weight and volume, higher power efficiencies, and therefore superior overall performance (Ref. 134). A cryogenic system in which SC machines have multi-MW ratings offers reduced weight and smaller volume, while being relatively inexpensive to operate (Ref. 135); the overall increase in power efficiency is primarily attributed to improved performance of semiconductor devices, as they offer reduced conduction losses and increased switching speeds (Refs. 135 to 138). Furthermore, such systems are not only limited to future commercial and military aircraft, but CPECs find their potential applications in medical diagnostics, multi-MW wind power generation, SC magnetic energy storage (SMES) devices, cargo ships, and instrumentation.

3.2.3.1.1 Power Semiconductor Devices

Characterization of power electronics devices at CT plays an important role in determining the performance of cryogenic converters. Numerous articles have been published discussing characterization and advantages associated in operating semiconductor devices around CT. Among the active devices, Si MOSFETs and IGBTs have been shown to offer reduced on-state resistance and breakdown voltage,

faster switching times, but increased gate threshold voltage (Refs. 135 to 137 and 139). Although the on-state resistance decreases, the reduction in breakdown voltage may ultimately lead to higher device resistance. In contrast, WBG devices have shown conflicting behavior; SiC offers increased on-state resistance and gate threshold voltage as well as stable breakdown voltage and switching speeds, whereas a GaN HEMT shows reduced on-state resistance, faster switching times, and stable breakdown and threshold voltages (Refs. 135 and 139).

3.2.3.1.2 Passive Components

Several papers have reported on properties of passive components with reduction in temperatures (Refs. 140 to 144). From the studies conducted, it can be concluded that most of the capacitors show degradation either in capacitance value, equivalent series resistance (ESR), or displacement factor. Only NP0 and polypropylene sulfide (PPS) based capacitors have been shown to offer stable capacitance and ESR (Ref. 140). Inductors have also been characterized in several published reports (Refs. 145 to 155). As a summary, it can be concluded from these articles that only powdered cores, specifically molypermalloy powder (MPP) core and high flux core (HFC), have been shown to offer stable inductance and increased quality factor with reduction in temperature. A brief review about passive components has been reported in Reference 139.

3.2.3.1.3 Cryogenic Power Electronics Converter (CPEC) Systems

Although much research has been done on characterization of semiconductors and passive components at CT, very few converters are reported for such applications. Furthermore, the converters developed are generally reported to have lower power ratings, and control circuitry placed outside the cryo-environment. In this section, review of both DC-DC and DC-AC converter topologies developed for CTs is provided. The review focuses on converter power ratings, type of semiconductor devices used, and type of filtering components employed.

The majority of the cryogenic converters developed so far are of the DC-DC conversion type. A 175-W pulse-width modulation (PWM) buck-type DC-DC converter was designed and tested at LN₂ temperature in Reference 156. The converter was operated at a 50-kHz switching frequency using Si devices, with an efficiency improvement of 1.2% at LN₂ temperature. MPP core and polypropylene film capacitors were used for filtering purposes, and the power circuit and filter components were placed inside the chamber. Similarly, References 157 and 158 presented the performance of a 60-W three-level buck converter operating down to 77 K at a switching frequency of 50 kHz. The performance of a hard and soft switching 500-W buck-type converter was reported for a temperature down to 20 K in Reference 159. Authors used different power MOSFET devices and diodes for the study and found maximum reduction of semiconductor losses to be 85%. For the experimentation, both the power and GD circuit were placed in CT. A 24-VDC, 20-W full-bridge DC motor drive for ultrawide temperature range (43 to 393 K) was developed in References 160 and 161. The targeted permanent-magnet DC motor drive was developed and tested until 89 K using Si devices. A similar 150-WPWM boost-type converter was reported in References 162 and 163 for a wide operating temperature range with an input of 24 and 48 VDC output. The converter operates with Si devices and has an efficiency of 92.2% at around 90 K, with the power circuitry and the MPP-based input energy storage inductor being placed inside the cold environment. Reference 164 reported the performance difference for a boost converter operating at RT and LN₂ temperature. Authors reported the soft-switching configuration at LN₂ temperature to be the most efficient in terms of semiconductor losses. Of all the DC-DC converters designed and tested for low temperatures, Reference 165 presented the highest power rating of 40 kW for a boost converter using CoolMOS MOSFETs and HTS inductor. HTSs normally behave as superconductors above 77 K up until

~100 K. For the converter developed, 85% reduction in terms of conduction losses and 87% reduction in overall system was reported for operation at LN₂ temperatures compared to RT performance.

In comparison with the available literature on DC-DC converters, much less has been reported on low-temperature DC-AC converters. A 40-kW, three-level active neutral-point clamped (3L-ANPC) converter was designed using Si devices in Reference 166. The converter was designed with 1 kVDC for an output of 3 kHz fundamental frequency, whereas a switching frequency of 140 kHz was adopted. Despite the reduction in breakdown voltage and increase in gate threshold voltage, Si-based devices were used for the intended cryogenic operation of the converter. Furthermore, numerous Si-based devices were put in series to compensate for the lowered breakdown voltage, which leads to increased gate-driving complexity and sophisticated control. An additional SiC-based Schottky barrier diode was configured to control reverse recovery losses. The converter also places the gate-driving circuitry away from the power devices, which may affect device performance and control. Similarly, a single-phase, three-level flying capacitor multilevel inverter was developed for near-cryogenic operating temperature in References 167 and 168. The inverter was designed for a load voltage of 45 V with DC link rated at 150 VDC using 200-V GaN field effect transistors (FETs). Although the efficiency and loss analysis has been provided for temperatures to 133 K, there is no information about the output voltage/current shape and quality at 133 K. Furthermore, the ratings of power and voltage considered are far lower than would be required for the CHEETA aircraft application. The latest development in DC-AC converter design has been reported in the design of a MW-class cryogenically cooled inverter (Ref. 169). Two 500-kW, 1-kVDC, 3L-ANPC inverters have been paralleled through interleaved inductors for an output voltage of 600 V. The inverter was developed using SiC devices operating at a switching frequency of 70 kHz with space vector modulation (SVM) using SiC devices. Although testing of the converter was carried out both at RT and LN₂, utilizing SiC-based devices for lower temperatures is not preferable because of increased conduction losses and gate threshold voltage. Additionally, all the gate-driving and auxiliary components were placed at RT.

From the review of converters, it can be observed that converters designed for cryogenic applications have low power ratings. Also, Si-based devices were used in most of the designs, whereas SiC- and GaN-based devices were used in just one design each. Additionally, all the DC-AC converters employed were primarily VSIs where a DC link storage capacitor was used; no analysis has been conducted for a CSI for such an application. CSIs historically have been neglected because of commercially unavailable reverse-voltage-blocking (RVB) devices and therefore increased conduction losses. Additionally, DC link storage inductors possess greater challenges in justifying the power density of the inverter. Considering the application of converters in cryogenic applications, SC inductors should be thoroughly evaluated in conjunction with a two-level CSI (2L-CSI).

3.2.3.2 Comparative Study and Tradeoffs for Design of CPEC

Before designing the real converter for cryogenic applications, the most feasible converter topology should be down-selected. A comprehensive comparison of the converter in terms of DC link storage, volume of passive components, efficiency, and electromagnetic interference (EMI) should be carried out to find the optimal topology. This section considers three basic converter topologies: (1) 2L-VSI, (b) three-level t-type neutral point clamped VSI (3L-TNPC-VSI), and (c) 2L-CSI. Phase-leg configurations of all three converter topologies are shown in Fig. 189. The power considered for such an analysis should be in multi-MW range as suggested by NASA for electric propulsion (Ref. 170). Therefore, a 2.5-MW converter will be considered as a baseline design in this section. In order to find the optimal topology for such an application, numerous factors were considered together to formulate a figure of merit (FOM) for the converters.

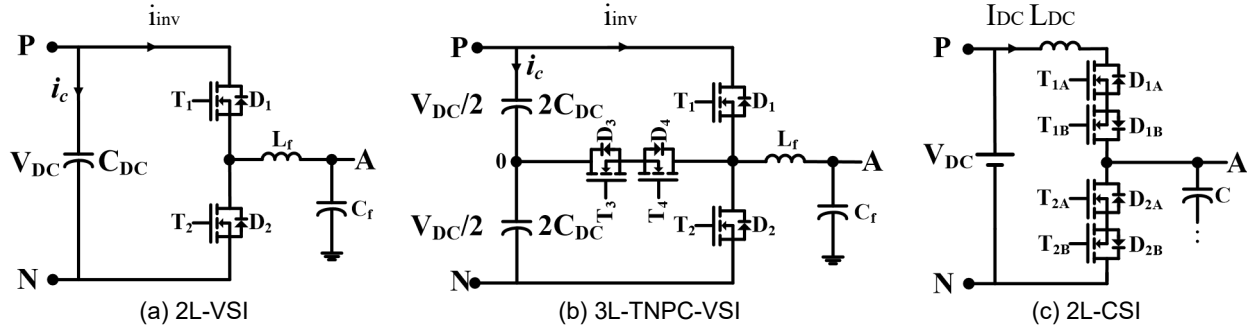


Fig. 189.—Topologies considered for MW-class converter.

3.2.3.2.1 Design Tradeoff for Energy Storage Components in CPEC

As a first index of FOM, energy storage of DC link capacitors and inductors in VSI and CSI is compared. This energy storage will be proportional to the magnitude of capacitance and inductance, and thus the resultant volume as well. Considering the DC link for VSI to be V_{DC} with energy storage capacitance C_{DC} ; and I_{DC} with energy storage to be L_{DC} for CSI; the energy storage for both the converter types can be related as:

$$\begin{aligned} W_{C_{DC},VSI} &= \frac{1}{2} C_{DC} V_{DC}^2 \\ W_{L_{DC},CSI} &= \frac{1}{2} L_{DC} I_{DC}^2 \end{aligned} \quad (51)$$

Further, assuming both converter types to be ideal and feeding a symmetrical load at V_{LL-RMS} for a power of P , and SVM control, the energy storage ratio for the VSI and CSI can be rearranged as:

$$\begin{aligned} \frac{1}{2} C_{DC} (\sqrt{2} V_{LL-RMS})^2 &= \frac{1}{2} L_{DC} \left(\sqrt{\frac{2}{3}} \frac{P}{V_{LL-RMS}} \right)^2 \\ \Rightarrow \frac{C_{DC}}{L_{DC}} &\propto \frac{V_{C_{DC}}}{V_{L_{DC}}} = \frac{P^2}{3V_{LL-RMS}^4} \end{aligned} \quad (52)$$

As can be inferred from Eq. (52), the DC link capacitance for VSI grows quadratically with power, whereas the inductance for CSI grows quadratically with load voltage. Therefore, power and output voltage influence the selection of DC link storage components, and this ratio may help optimally select the converter. It is worth noting that this relation is irrespective of volumetric scaling between DC-link components of both considered converter types.

In order to have a baseline with regard to converter topology, a variable power rating from 1 to 2.5 MW was considered for load voltage (V_{LL-RMS}) of 600 and 800 V. As can be seen from Fig. 190, the energy storage ratio exceeds unity for all the powers considered for 600 V, whereas it exceeds 1.25 MW for 800 V. This ratio indicates a VSI to be offering greater volumes for DC link storage elements for MW-class inverters, therefore favoring a CSI from the perspective of volumetric density.

Apart from lower load voltage favoring a CSI, a lower DC link voltage also supports a CSI from the perspective of smaller energy storage. Considering a fixed energy/power content, a higher DC link current is required at reduced voltages, resulting in a lower value of L_{DC} for a CSI. For inductor development, numerous core materials can be considered; however, an air-core inductor together with SC tapes can be employed for high power/current applications, especially at CT. It is worth mentioning that the loss in relative permeability (μ_r) while using an air-core inductor is compensated for by the capability to conduct much higher currents CT (Ref. 155), resulting in reduced volume of DC link energy storage for a CSI.

Furthermore, the problems associated with different core materials in terms of hysteresis and core losses as well as limited saturation flux density can also be avoided using air-core inductors.

3.2.3.2.2 Volume Estimation for Passive Components in CSI and VSI

This subsection discusses the dimensioning of filter components. Given that filter components occupy 30% of the volume of any converter (Ref. 171), it is important to have an appropriate estimate of their magnitude and volume. These filter components primarily composed of different configurations of inductors and capacitors, where input storage capacitors are required at the DC link for VSIs, and output filter capacitors are required for CSI. Conversely, input storage inductors are required at the DC link for CSIs, and the LC-type output filter is considered for VSIs to have comparable total harmonic distortion (THD). For the volume estimation, converter ratings of 2.5-MW generating an output V_{LL-RMS} of 600 V are considered. The equivalent DC link voltages of 1,000 VDC and 540 VDC are considered for VSIs and CSIs, respectively, with SVM as a modulation technique (Ref. 172). To validate the design of passive components, simulation models were developed in MATLAB® (The Mathworks Inc.) and LTspice® (Analog Devices, Inc.), and design specifications were verified.

3.2.3.2.3 Sizing of DC Link Storage in CSI and VSI

An estimate of input DC link capacitance based upon ripple current can be made from Reference 173; however, actual values may be limited by the current rating (Ref. 174). DC link storage capacitors for VSI can be identified from Fig. 189(a) and Fig. 189(b). Similarly, DC link storage inductance for CSI can be estimated from Reference 175 and can be identified in Fig. 189(c). Both components can be dimensioned using Eq. (53), where f_{sw} is the converter switching frequency, and ΔV_{DC} and ΔI_{DC} represent the ripple in DC link voltage and current, respectively. For the calculations of DC link storage elements, a voltage ripple and current ripple percentage of 5% was used for VSI and CSI, respectively. A summary of DC link sizing for all converter types considered is presented in Table 26, where nearest COTS components are tabulated.

$$\begin{aligned} C_{DC} &= \frac{P}{(V_{DC}\Delta V_{DC} \pm 0.5\Delta V_{DC}^2)f_{sw}} \\ L_{DC} &= \frac{V_{LL-RMS}}{4f_{sw}\Delta I_{DC}} \end{aligned} \quad (53)$$

3.2.3.2.4 Sizing of AC Link Filters in CSI and VSI

The output filter combination for the converter depends upon the type and sensitivity of the load. Although a fully SC machine for CT results in a 3.5-times higher power-to-weight ratio than other configurations, it is very sensitive to AC losses (Ref. 176). Therefore, converter output voltage should have lower THD and better power quality. To determine the output filter, a voltage THD limit of 3% was considered for the analysis. To achieve this, an inductor-capacitor (LC)-type filter combination is considered for VSIs and can be designed using Eq. (54), whereas the already existing dV/dt filter should be sufficient for CSI. In the equations, ΔI_{max} represents the peak-peak ripple current, whereas f_{res} is the resonance frequency.

$$\begin{aligned} L_f &= \frac{V_{DC}}{3\sqrt{2}\pi f_{sw}\Delta I_{max}} \\ C_f &= \frac{1}{4\pi^2 f_{res}^2 L_f} \\ C &= \frac{I_{DC}}{4f_{sw}\Delta V_{LL-RMS}} \end{aligned} \quad (54)$$

A summary of AC link filter sizing for all converter types considered is provided in Table 26.

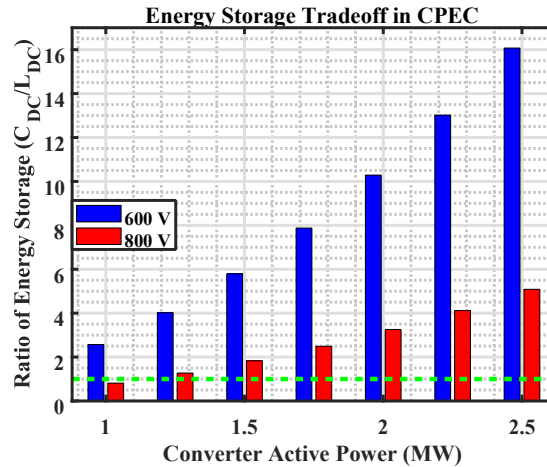


Fig. 190.—Comparison on basis of energy storage.

TABLE 26.—SIZING OF DC LINK STORAGE AND AC LINK FILTER COMPONENTS

Property	2L-VSI	3L-TNPC-VSI	2L-CSI
C_{DC}	1500 μ F, 1150 VDC	2 $\times$ 3000 μ F, 630 VDC	-----
L_{DC}	-----	-----	25 μ H, 3290 A
C_f	3 $\times$ 65 μ F, 400 VAC	3 $\times$ 60 μ F, 400 VAC	-----
L_f	6.15 μ H, 3400 A	4.61 μ H, 3400 A	-----
C	-----	-----	3 $\times$ 430 μ F, 400 VAC

3.2.3.2.5 Overall Volume Comparison of DC Link Storage and AC Link Passive Components

Having determined the magnitudes of different filtering components, comparison among the topologies was extended to determination of the volume occupied by them. To determine the volume for inductors, numerous inductor types commonly used for CT applications were considered. From the literature, it can be concluded that ferrite and amorphous materials not only lose their permeability, but their core losses also increase, whereas powdered cores show stable operation (Ref. 139). Considering the MW range of power, numerous powdered cores would need to be paralleled to justify area product A_p (Ref. 177), resulting in huge volume and weight. Instead, coreless inductors with SC wires/tapes were considered for the analysis and were found to be more suitable for such low-temperature and high-power applications. Coreless inductors offer not only zero core losses but also no limitation on saturation flux density (Ref. 155). Numerous SC tapes such as magnesium diboride (MgB_2) or yttrium barium copper oxide (YBCO) can be utilized for such inductor-building applications; YBCO-type superconductor was considered in this application, with a conservative selection of current density of 200 A/mm². This superconductor was wound in a multilayer cylindrical orientation to furnish the inductors specified in Table 26. As part of inductor optimization process, single- and double-layered windings were wound around various diameters of Al cylinders (serving as inductor former). The optimization process for L_{DC} is shown in Fig. 191. As can be seen from Fig. 191(a), winding the SC tapes around a 2-in.-diameter inductor former using a double layered structure not only offers the least number of turns but also results in a superconductor of the shortest length, as shown in Fig. 191(b). Therefore, L_{DC} will give a volume of around 0.2157 L by having 31 turns in a double-layered structure, as shown in Fig. 191(c). The approach

was extended to determination of volume for output filter inductors of VSIs. Optimized volume with this approach is displayed in Fig. 192(a).

Similar to inductor optimization, optimization of capacitors was also carried out. PPS capacitors are considered for the comparison because of their negligible variation in capacitance and reduced ESR at CT (Ref. 139). Capacitors from numerous vendors were selected, and the one offering the best combination of volume and ESR is being considered in the analysis. Based upon the manufacturer-provided datasheet (Ref. 178), variation of capacitor volume was studied and plotted in Fig. 192(b) for all three topologies, whereas overall converter volume (sum of inductors and capacitors) is shown in Fig. 192(c). It can be seen from the overall volume comparison that 2L-CSI occupies the least volume in terms of passive components.

3.2.3.2.6 EMI Performance Comparison Between 2L-VSI and 2L-CSI

In this section, a comparison between 2L-VSI and 2L-CSI will be carried out based on conducted EMI; 3L-VSI is omitted from the comparison as Fig. 192(c) suggests the highest volume for passive components for this topology. For the analysis, LTspice-based simulations were carried out for the two converter topologies with numerous parasitics elements as shown in Fig. 193(a) and Fig. 193(c).

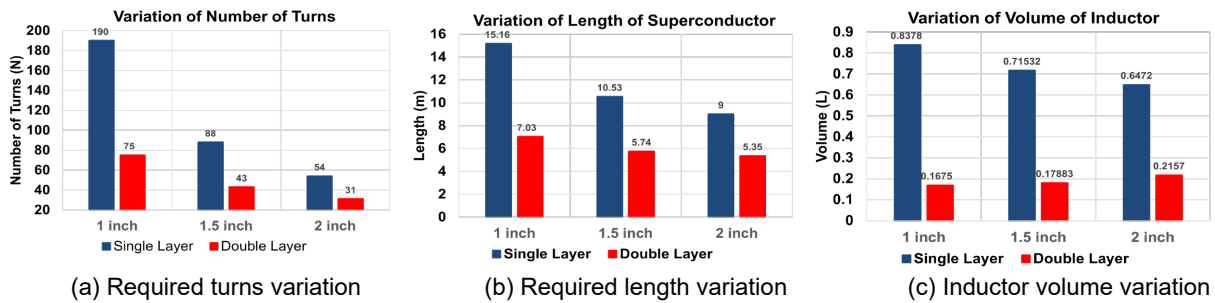


Fig. 191.—Optimization of L_{DC} for 2L-CSI.

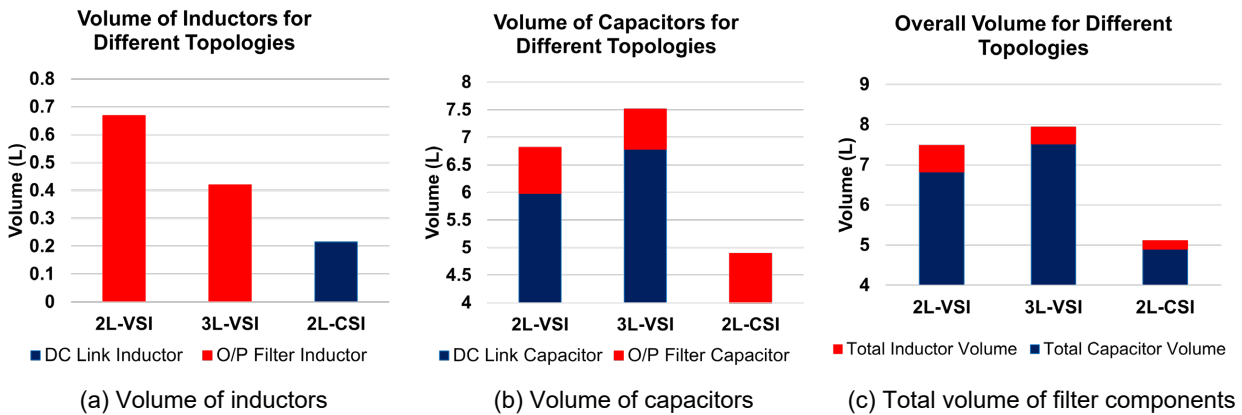


Fig. 192.—Variation of volume with topology.

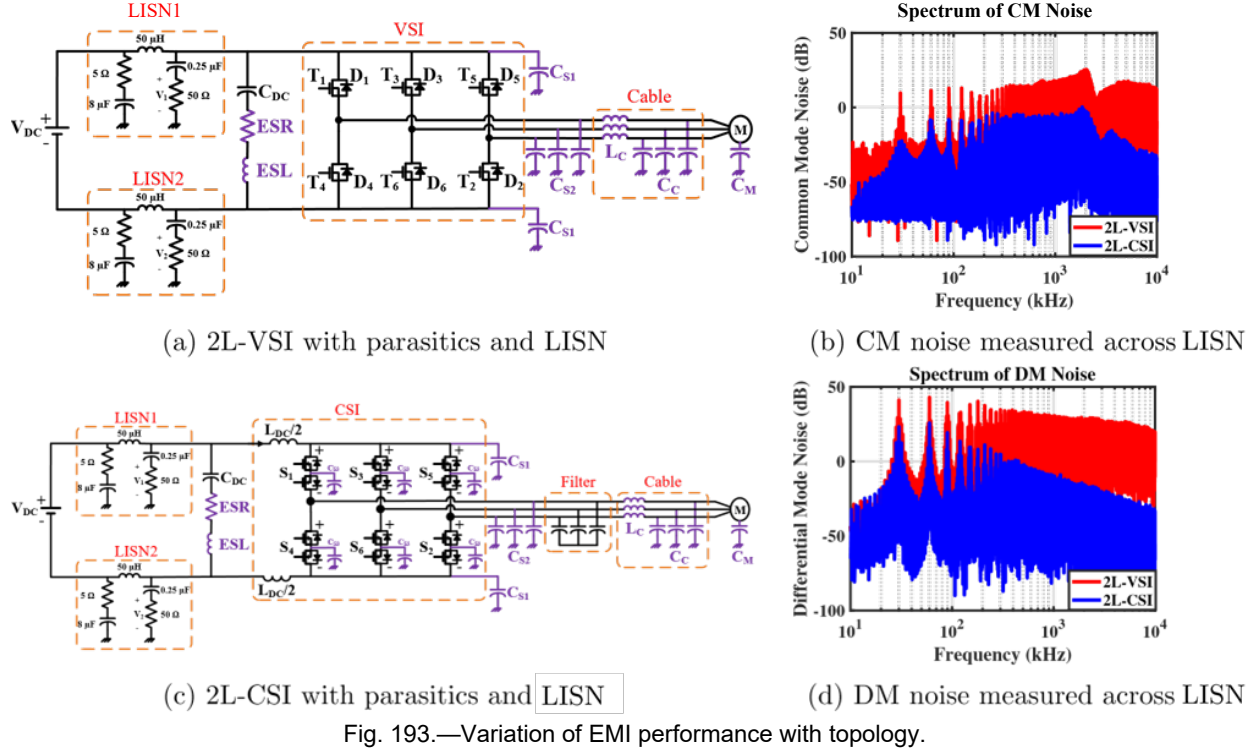


Fig. 193.—Variation of EMI performance with topology.

Typical parasitics considered for the comparison are shown in Fig. 193(a) and Fig. 193(c). Values of 200 and 560 pF were considered for the stray components $C_{S1}/C_{S2}/C_{S3}$ and C_M , respectively, whereas 100 nH and 50 pF were adopted for L_c and C_c , respectively, for cable length of 1 m. The simulations were carried out for output power of 2.5 MW at 600 V_{LL-RMS} while using real device models from the manufacturer. For the DC link, values calculated in Section 3.2.3.2.3 were utilized. Furthermore, switch node parasitic capacitances, stray capacitance from DC link to ground, cable parasitic elements, and motor model were also simulated and were kept uniform for both converters as discussed in Reference 179. Both common mode (CM) and differential mode EMI spectra were determined and plotted as shown in Fig. 193(b) and Fig. 193(d), respectively. For the analysis, a MIL-461-based line impedance stabilization network (LISN) was considered and was connected at the input side of the converter.

The EMI spectrum for a 2L-CSI was further extended by including equivalent parallel capacitance (EPC) of L_{DC} . It was found that EPC forms a self-resonating parallel LC tank, which can contribute to the conducted noise. It was also found that a 2L-CSI still offers lower noise spectrum even after consideration of EPC. However, the value of EPC needs to be optimized. The methods to limit EPC of inductors can be found in References 177 and 180.

3.2.3.2.7 Summary of Design Tradeoffs for CPEC

From the conducted analysis, it can be easily observed that a 2L-CSI offers advantages in terms of smaller DC link storage inductor size as compared to capacitor size in VSI. The collective volume of passive components both at the DC and AC link was found to be least with 2L-CSI while using an air-core inductor with SC material. An estimate of conducted EMI was also made, which supports 2L-CSI in terms of lesser interference generated, hence smaller sizes of required filters. Furthermore, a CSI inherently offers better power quality and THD at the output, thus helping in reducing AC losses in SC machines (Ref. 179). The disadvantages of higher conduction losses can be neutralized by utilizing monolithically integrated bidirectional RVB devices (Ref. 181).

Considering these factors, a 2L-CSI was determined to be more feasible for MW-scale power conversion systems for cryogenic applications. Therefore, a 2L-CSI converter will be designed, analyzed, and tested in the rest of this section.

3.2.3.3 Design and Demonstration of CPEC Subsystems

From the perspective of power electronics converters, a majority of the converters designed have utilized Si devices (Refs. 159, 164, and 166). Despite the optimal operating performance of GaN devices at CT (<123 K), only single-power electronics converters have been built using GaN FETs (Ref. 167), and these have been tested until 133 K only. One of the reasons for the uncommon emergence of GaN converters at CT may be the nonavailability of GDs at such low temperatures (Ref. 182). To date, none of the GDs have been reported to operate successfully at CT. Given that GaN-based devices are extremely sensitive to fast switching signals and layout (Ref. 183), operating them at low temperatures further increases their switching speeds (Ref. 184). This adds to the already-existing complexity levels, and therefore the design of a reliable, repeatable, and resilient GD configuration for such devices will be extremely important for upcoming applications (Ref. 182).

3.2.3.3.1 Design of Gate Driver at CT

In almost all of the converters built for CT applications, GDs and auxiliary components have been placed outside the cryogenic environment. This type of arrangement can perform satisfactorily if switching devices are Si based, whereas making GaN devices work with such an arrangement will be extremely difficult. Considering the superior performance of GaN devices at CT, together with the absence of any low-temperature GD, this section targets successful implementation of a LN₂ immersed GD. Implementation of a gate-driving PCB for such low temperatures is susceptible to numerous challenges, including (1) choice of passive components, such as decoupling capacitors, resistors, and CM chokes; (2) choice of ICs, such as a gate-driving IC; (3) choice of diodes, such as Zener diode or rectifier; (4) choice of manufacturing material, such as PCB material, soldering material, or interconnects; and (5) selection of isolated APS.

A noise-tolerant and LN₂-temperature-resilient GD was designed to overcome the challenges discussed previously. The GD was successfully tested at RT as well as at CT under both unloaded and loaded conditions. In the proposed CT GD, all the components will be placed inside a low-temperature environment. A typical GD consists of a buffer IC, a GD IC, and an APS; the overall configuration for the proposed LN₂ immersed GD is shown in Fig. 194.

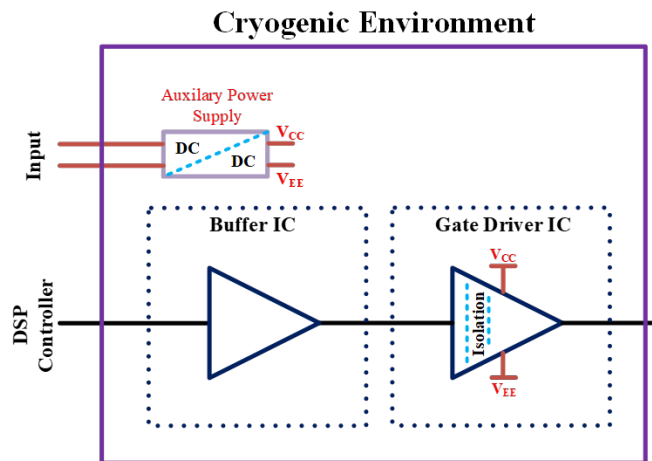


Fig. 194.—Structure of the GD.

3.2.3.3.2 Selection of Gate Driver Components at CT

For a well-functioning GD, both the ICs and APSs should be compatible with extreme low temperatures. For this purpose, numerous COTS APSs were tested and characterized in laboratory at both RT and LN₂ temperatures. Since most of the COTS APSs available in the market use ferrite-based core materials, it is nearly impossible to find a COTS APS that can work at extremely low temperatures. This may be attributed to increased power losses, reduced inductance, and decreased quality factors at extreme low temperatures (Ref. 139). A buffer stage is also added to offer high impedance to the microcontroller, whereas a gate-driving stage is designed to source and sink high gate-source currents for GaN switching devices. Considering the ICs required in a GD board, numerous technologies, such as bipolar junction transistors (BJTs) or complementary metal-oxide-semiconductors (CMOSs), are commercially available in abundance. For the GD, CMOS devices are good candidates, as they offer improved performance in switching speed (Ref. 185), whereas BJT is not preferred, as these devices suffer from a significant decrease in current gain (Ref. 186). In the design of the GD, passive components were also selected to be CT compatible; for example, nanocrystalline-based inductors and CM chokes were used because of their increased permeability and saturation flux density with reduced temperatures (Ref. 139). Additionally, ceramic-based NP0 and X5R decoupling capacitors were used because of their negligible variation of capacitance and dissipation factor with temperature. For the resistors, thin film material was selected for prototyping the board. Normal PCB material consisting of FR-4 (flame retardant 4) was selected for the fabrication of the board, and Sn42/Bi57.6/Ag0.4 solder paste was used to solder various components.

3.2.3.3.3 Laboratory Prototyping of CT Gate Driver and Test Results

Four-layered PCBs were designed for prototyping five different GD ICs with similar board layout and auxiliary components. Two versions of the boards were later developed. For the first version, R05P209S APS was used, assuming its successful operation at CT. Fig. 195(a) shows the first version of GD prototyped with dual-stage isolation.

A conduction-cooled cryogenic setup was used for cryogenic testing. An 18-in. stainless steel chamber was integrated with a single-stage cryopump (CryoMech A230). The chamber was evacuated to 10⁻⁶ Torr by a Pfeiffer HiCube 80 Eco turbomolecular pump system. The system reached the 25-K base temperature after 4 h of cryocompressor operation. The assembled GD board was thermally anchored to a 2-mm thick copper plate by a thermally conductive epoxy, StyCast 2850FT. A 50-μm-thick polyimide film was bonded to the copper plate to insulate the GD circuitry. The copper plate was tightened with the baseplate of the chamber, where the cold/base-plate temperature was measured by a Si-diode CT sensor (LakeShore 420 model) as shown in Fig. 195(b). To prevent heat loss by radiation, the GD assembly was encompassed by a copper radiation shield and 10 layers of polyethylene terephthalate (PET) MLI, as shown in Fig. 195(c). Finally, the GD was tested to a temperature of 77 K, where successful operation at 200 kHz and 50% duty cycle is shown on the oscilloscope in Fig. 195(d).

All the measurements for characterization of this setup were taken outside the chamber with a Tektronix TBS 2000 B oscilloscope. During the test setup, it was ensured to have minimal length of measurement connections. The complete test setup, including the cryogenic chamber alongside measurement equipment, is shown in Fig. 196.

After repetitive testing and characterization of GD board components at CT, two gate-driving ICs were found to be working up to 77 K. However, only a single CMOS-based buffer IC (ISO7820) was found to be promising at LN₂ temperature. Only one of several combinations of COTS APSs showed promising operation. A brief summary of the performance of GD ICs and APSs, with temperatures, is provided in Table 27 and Table 28, respectively.

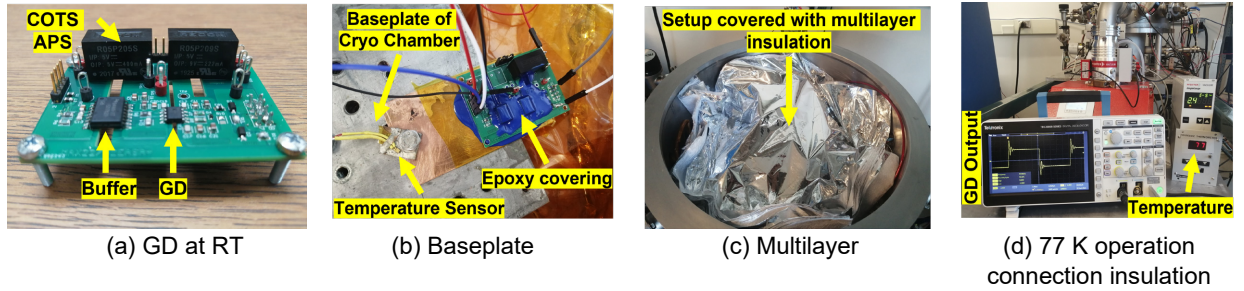


Fig. 195.—Various steps during characterization of GD.

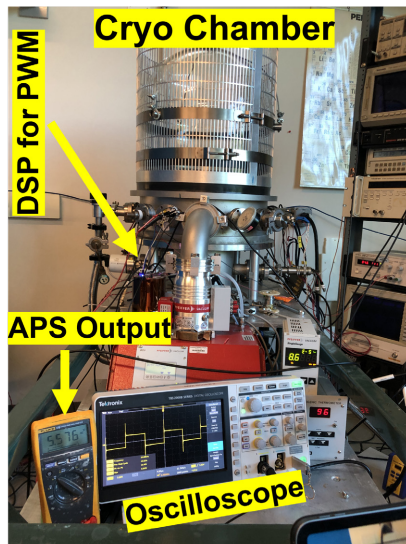


Fig. 196.—Outside view of cryochamber

TABLE 27.—SUMMARY OF GATE DRIVER ICS PERFORMANCE WITH TEMPERATURE

Property	SI8271	SI8235AD	UCC21540A	MAX22702	ADUM4221
Fabrication process	—	—	BiCMOS	BiCMOS	Monolithic
Technology	SiC	Si	—	SiC	—
Minimum temperature	77 K	~130 K	~93 K	~130 K	77 K

TABLE 28.—SUMMARY OF COTS APS PERFORMANCE WITH TEMPERATURE

Part number	Manufacturer	Power, W	Performance at LN ₂	Efficiency at LN ₂
PEM2-S5-D5-S	CUI INC	2	Consistent	Reduced to 30%
PEME1-S12-S12-S	CUI INC	1	Consistent	Reduced to 34%
MDS01M-12	Meanwell Web	1	Consistent	Reduced to 24%
MDS01L-12	Meanwell Web	1	Consistent	Stopped working
TRV 1-1212M	Traco Power	1	Consistent	Stopped working
RV-0509S	Recom Power	2	Sometimes malfunctions	Negligible reduction
R05P209S	Recom Power	2	Inconsistent	Negligible reduction

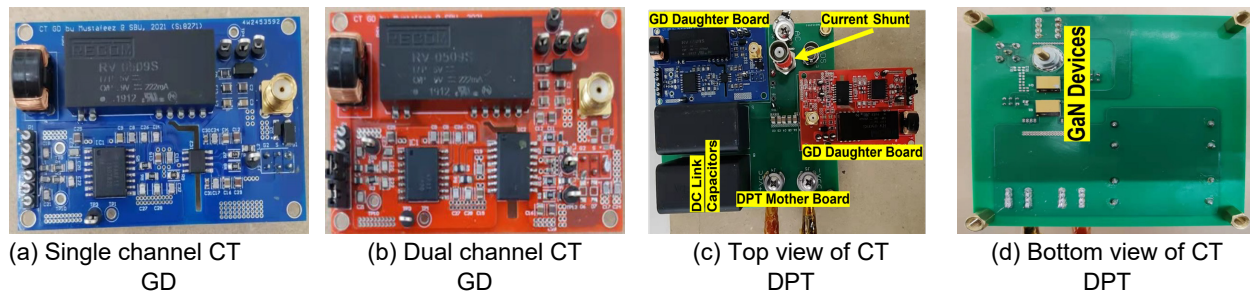


Fig. 197.—Development of converter subsystems at CT.

3.2.3.3.4 Cryogenic Double-Pulse Test (DPT) Platform for Validation of Gate Driver and Device Characterization

Based upon the observations and experimentation with the first GD board, a second version was developed specifically for LN_2 applications. This version incorporates all the components already tested to be operating fine at reduced temperatures.

Having developed a successfully operating GD board at LN_2 temperature, a DPT platform was also developed for GaN HEMT. Not only would GD boards together with a DPT board represent the real switching behavior, but the switching properties of the device can also be determined. For this purpose, GaN systems 650-V, 60-A device GS66516T was used on the DPT board, and a finalized version of single- and dual-channel cryogenic GD boards were prototyped as shown in Fig. 197(a) and Fig. 197(b), respectively. The GD (daughter) boards are designed to be modular and can be attached to a DPT (mother) board, as shown in Fig. 197(c). For the DPT board, PPS capacitors by Vishay were employed as main DC link capacitors, while X5R high-voltage decoupling capacitors were used due to the nonavailability of NP0 capacitors at such high voltages.

3.2.3.3.5 Dynamic Characterization of GaN Device and Comparison Between RT and 77 K Operation

This subsection compares different quantities for both RT and LN_2 tests performed. RT testing of the designed DPT and GD boards was conducted before subjecting them to LN_2 temperature, as shown in Fig. 198(a). DPTs performed at RT were repeated at LN_2 , as shown in Fig. 198(b), with similar turn-on and turn-off times to compare the performance of the switching device. For consistency in measurements, similar probes were employed both for RT and LN_2 testing. A 1-GHz bandwidth oscilloscope by Tektronix was used for the measurements, and active probes of 200-MHz bandwidth (labeled as THDP0200) were used for gate-source and drain-source voltage measurements. For the gate-source voltage measurements, subminiature version A (SMA) based coaxial cable with 50 Ω external termination was used to avoid inserting active probes inside the LN_2 dewar. A 2-GHz, 101-m Ω current shunt (labeled SSDN-10) was also placed onto the DPT board to monitor the device current. During the measurements, it was ensured the BNC cables for gate-source voltage and device current were terminated properly. Furthermore, numerous combinations were adopted to measure drain-source voltage from inside the LN_2 dewar; inserting jaws of active probe inside the dewar was found to be a workable solution. It was observed that jaws of active probes become stiff inside LN_2 but are still functional, unlike passive probes. Therefore, only active probes (with limited bandwidth) can be used for drain-source voltage measurements inside the dewar.

Utilizing the CT setup shown in Fig. 198(b), DPT tests inside LN_2 polystyrene foam buckets were performed for numerous DC link voltages and load currents. Based upon the DPT performance, differences in device-switching characteristics were observed. As seen from Fig. 199(a) and Fig. 199(b), the GaN-based switching device turns off quickly at LN_2 . As a result, the peak overshoot voltage and the

device dV/dt and dI/dt at turn-off were recorded to be higher as compared to RT; this trend was found to be generic for all the voltages considered for a device current of 30 A. Similarly, faster turn-on behavior of switching devices can also be observed from Fig. 199(d) and Fig. 199(e) for low temperatures. The time improvement for turn-off and turn-on was measured to be around 100 and 20 ns, respectively, and no significant variation was observed with change in operating voltage. This improvement in turn-on and turn-off time will directly influence the switching performance of the device.

Based upon the switching performance determination of the GaN device at RT and LN₂, switching loss calculations were performed using a 10% I_{DS}/V_{DS} and 90% V_{DS}/I_{DS} criterion (Ref. 187). A method of V-I alignment with deskew adjustment was adopted for precise calculations of the losses, as elaborated in Reference 183. It is worth mentioning that V-I alignment with deskew adjustment was repeated inside the LN₂ dewar before measuring the actual circuit parameters to properly calibrate the system and avoid any errors. The turn-on and turn-off switching losses at two temperatures and blocking voltage of 400 VDC are presented in Fig. 199(c) and Fig. 199(f), respectively. It can be observed that the increase in switching speed resulted in a significant reduction in losses. In the past literature, dynamic characterization of GaN HEMTs has only been reported until 133 K (Refs. 169 and 184), and loss analysis below this temperature is being reported for the first time.

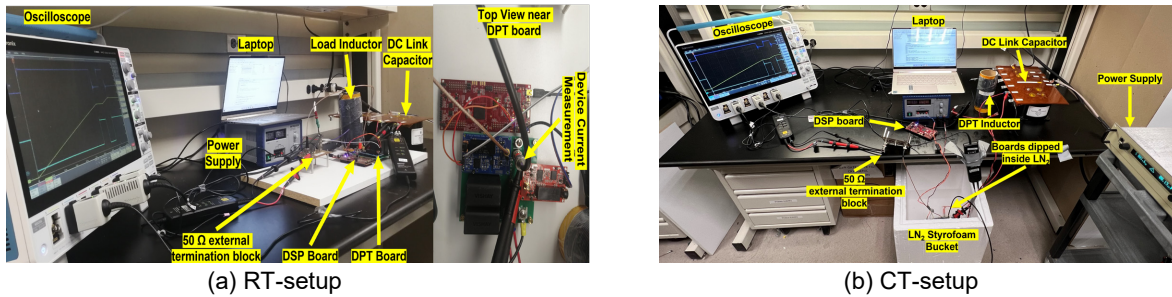


Fig. 198.—Double-pulse test setups at different temperatures.

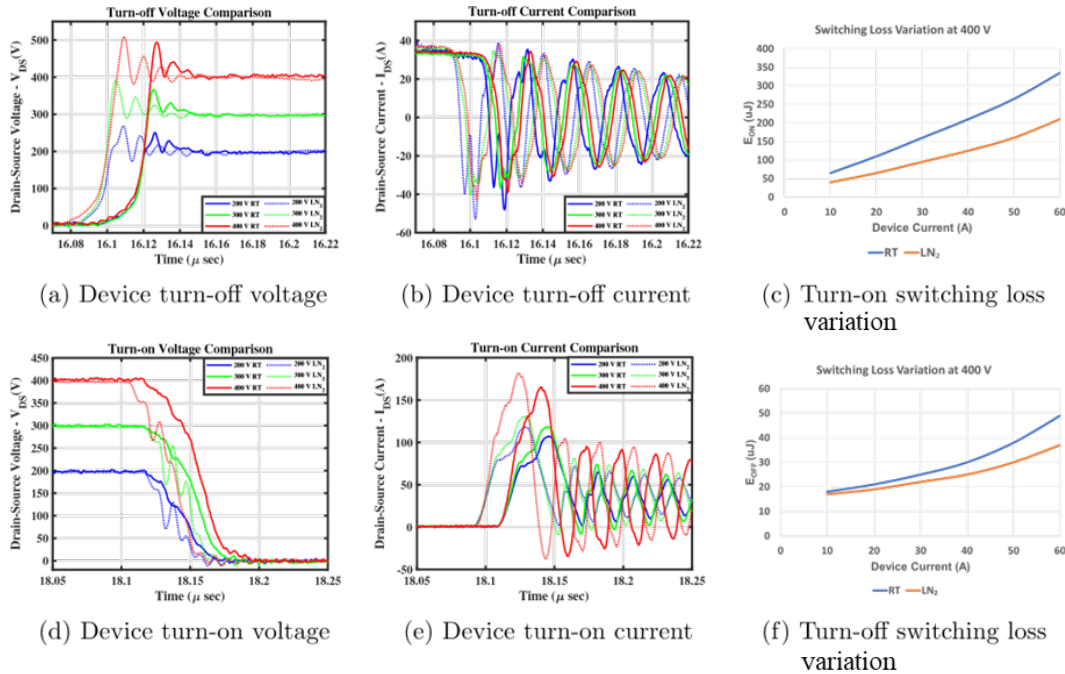


Fig. 199.—RT and LN₂ comparison of switching device.

TABLE 29.—DETAILS OF COMPONENTS USED IN FINALIZED DESIGN

Component	Part number	Specification
Gate-driving ICs	Silicon Labs Si8271 Analog Devices ADuM4221	200 V/ns CMTI 150 V/ns CMTI
Buffer IC	Texas Instruments ISO7820x	CMOS
Auxiliary power supply	Recom Power RV-0509S	5 to 9 V converter
Common mode choke	Würth Elektronik 7448011008	
Decoupling capacitors	NP0 or X5R	Numerous voltage levels
Resistors	Thin Film	Least temperature coefficient
Zener diodes	SMAZ6V2-13-F, DDZ9685-7	1 W, 0.5 W
Solder paste	Sn42/Bi57.6/Ag0.4	
DC link capacitor	MKP1848C62580JP4	Polypropylene
GaN switches	GS66516T	650 V, 60 A

3.2.3.3.6 Summary and Details of Components for CPEC Subsystems

As observed from the results presented, the designed GD does not only work at extremely low temperatures (e.g., LN₂) but also controls the switching device quite amicably since no mistriggering and/or jittering was observed during the DPT tests conducted. Therefore, the proposed GD can successfully be employed in future device characterization, as well as CPEC systems. Details about the important components used in the GD and DPT board are presented in Table 29.

3.2.3.4 Fully CPEC Design, Integration, and Demonstration

This section focuses on designing and demonstrating a 2L-CSI inverter for cryogenic applications. The choice of topology has been primarily dictated by advantages in terms of lower volume, better source and load-side power quality, and lower magnitudes of conducted EMI, as presented previously. For a 2L-CSI, the advantage in volume is influenced by air-core inductors using superconductor materials, whereas better source and load-side power quality affects the operation and performance of SC transmission lines and machines, respectively.

3.2.3.4.1 Design of a Three-Phase 2L-CSI CPEC

The design for a 20-kVA three-phase CSI with an output voltage of 300 V_{LL-RMS} is described in this section. Since device-blocking voltage for CSI is dictated by peak output voltage, an output voltage of 300 V was selected, as limited by the availability of commercially available discrete devices that are suitable for cryogenic applications.

3.2.3.4.2 Selection of Semiconductor Device for 2L-CSI

Based upon the device characterization results reported in literature (Refs. 135 to 139), only Si MOSFET and GaN HEMT show better performance towards CT. Therefore, a GaN-based discrete device, GS66516T, is used for the analysis. As of this writing, this device has the highest voltage and current rating of any commercially available GaN product that also has been characterized to perform better at reduced temperatures (Ref. 184). Its dynamic characterization at LN₂ was performed for the first time in the previous section, as shown in Fig. 199. Because a 2L-CSI needs a bidirectional current and RVB switch, two devices were connected in an anti-series configuration known as common source (CS) configuration. CS is simple to configure with commercially available GDs, hence reduced gate-driving complexities, and bidirectional current carrying capability (Ref. 188), whereas it has similar

semiconductor losses as common drain (CD) configuration. However, a monolithic CD bidirectional switching device can be designed to have the same on-state resistance as a standard switch (Ref. 189).

In order to optimize the performance, a parametric sweep with frequency, device number and modulation technique was conducted for CS configuration using Reference 190). A combination of minimum of two devices per position results in efficiencies above 99% with SVM as shown in Fig. 200(a). Therefore, two devices were paralleled per switching position in the converter layout to achieve 99+ % efficiencies, while the switching frequency was limited to be 30 kHz to fulfill the minimum power efficiency at RT.

3.2.3.4.3 Power Loop Design and Optimization for 2L-CSI

Considering the sensitivity of GaN devices to converter layout and parasitics involved, switching cell of 2L-CSI needs to be optimized as it affects the transient voltages and device performance (Ref. 191). Furthermore, power loop inductance of all three cells should be symmetrical to reduce the circulating currents. In order to fulfill both the objectives, all the switching cells were placed on a ‘Y’ shape configuration whereas vertical power loop placement was used to reduce the board inductance. A six-layer PCB containing devices on the bottom side and decoupling capacitors on top side was designed where forward current was distributed in even layers while return path was provided by the odd layers. This type of configuration resulted in maximum cancellation of flux, hence an optimized power loop inductance alongside reduced CS inductance was achieved for better operating performance. To validate the concept, the layout was exported into Ansys Q3D and parasitics analysis was performed. The power loop inductance between switching cells S1-S3, S3-S5, and S5-S1 was not only symmetrical but also resulted in the lowest inductance value reported for any CS inductor, 3.85 nH. The designed PCB with switching devices soldered is shown in Fig. 200(b), where switch numbering is similar to Fig. 200(c).

3.2.3.4.4 Selection of Passive Components for 2L-CSI

A 2L-CSI involves primarily two types of passive components: a DC link storage inductor L_{DC} and an output filter capacitor C , as labeled in Fig. 189(c). An air-core inductor design using SC magnetic tape YBCO was considered for the DC link inductor, and PPS capacitors by Vishay were used for the output filter capacitor.

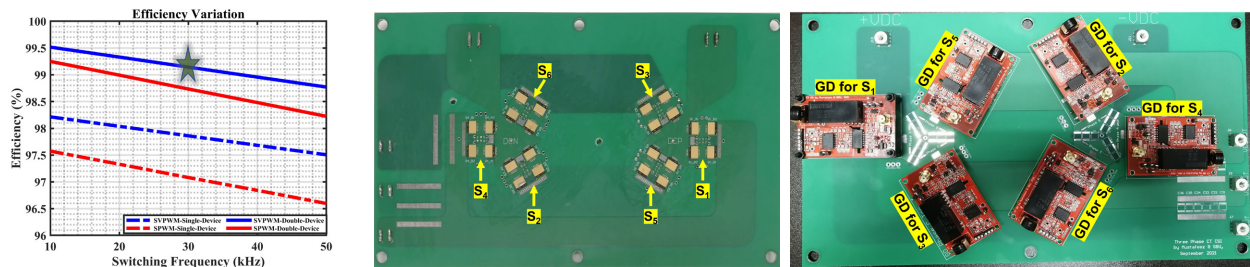


Fig. 200—2L-CSI converter design and development. (a) Selection of device modulation technique. (b) Bottom side of power board containing GaN devices. (c) Top side of power board number and containing GD daughter boards.

3.2.3.4.5 Converter Layout Considerations

Having successfully characterized the switching device and GD, the designed subsystems were employed for the design of the whole converter. The envisioned three-phase inverter consisted of the following: (a) a power board containing switching devices, (b) six GD boards containing cryo-compatible components, and (c) a filter capacitor board containing an output filter capacitor. Considering the bidirectional RVB switches to be in CS configuration as shown in Fig. 193(c), the overall converter structure requires six dual-channel GD boards. A GD board with IC ADuM4221 can be used to drive two channels independently; therefore, the same GD board has the feature to control two CS-configured switching devices. All the boards have been configured to be identical so they can replace each other in a DPT board or power board. The assembled power board is presented in Fig. 200(c) alongside the GD daughter boards.

3.2.3.4.6 Room Temperature Testing of 2L-CSI

To carry out testing of the whole inverter, all the boards mentioned were populated with respective components and tested individually. Collective testing was subsequently carried out as described in the following sections.

3.2.3.4.7 Double-Pulse Evaluation of Switching Cells

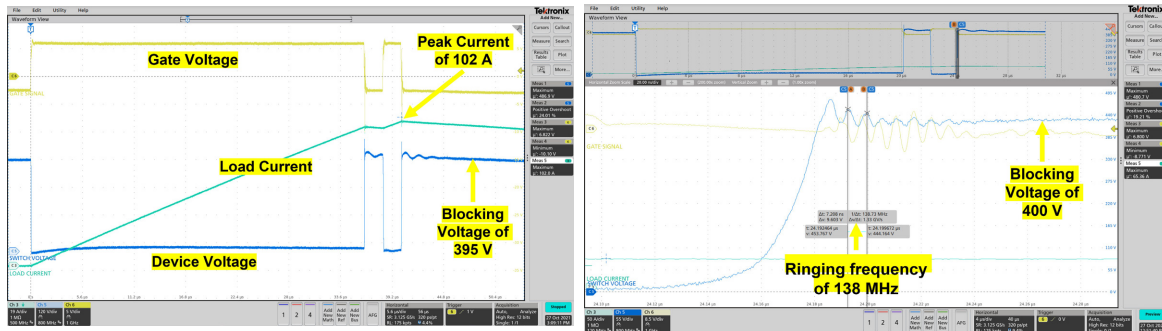
Before carrying out the inverter testing, DPT was performed on all switching cells as identified in Reference 192 under rated conditions. Since two 60-A devices are paralleled per position, the peak current of 120 A must be pushed at the rated voltage. This not only helps in characterizing the power loop inductance, but information about collective switching losses and gate-driving capability was also analyzed. For the accuracy of measurements, a passive probe (TPP0850) of 800-MHz bandwidth and 1,000 V was used for measuring device voltage, and a Rogowski coil (TRCP0600) was used for load current measurement. As seen from Fig. 201(a), a peak load current of around 102 A was pushed at 395 V. Fig. 201(b) shows the ringing frequency of the device voltage to be 138 MHz. Considering C_{oss} of 135 pF at 400 V, the power loop inductance can be calculated to be 5.2781 nH, which closely agrees with the Ansys Q3D extracted value. The relatively higher measured value of inductance is related to additional inductance being introduced by the packages of four pairs of GaN devices in a switching loop (Ref. 193). Similar DPTs were extended to all six switching cells to ensure that the switching performance is identical and that there is no influence on nonswitching devices.

3.2.3.4.8 Integrated Testing of 2L-CSI

Having performed the DPT on all the switching loops, the whole inverter setup was integrated as shown in Fig. 202(a). For the evaluation of the inverter, a 391- μ H MPP core inductor was utilized as DC link energy storage (because of nonoperation of superconductors at RT). The converter feeding a three-phase resistive load was driven by a laboratory power supply. To test the converter continuously under higher operating voltages, a commercially available heat-sink was used to keep the devices under safe operating temperatures.

A three-phase unified SVM was used as a modulation technique (Ref. 194), and a cyclone IV field-programmable gate array (FPGA) was used for the generation of 12-PWM signals. For the safe transition of current from phase-phase, an overlap time of 650 ns was configured among the alternate phases. Similarly, a delay time of 1.3 μ s between two switches of all the CS configurations was established to ensure a unidirectional current flow during the commutation process, hence preventing phase-phase short circuit across output capacitors (Ref. 195). Using the switch convention shown in Fig. 193(c), the PWM for a CS configuration has been divided into top and bottom switch and is shown in Fig. 202(b).

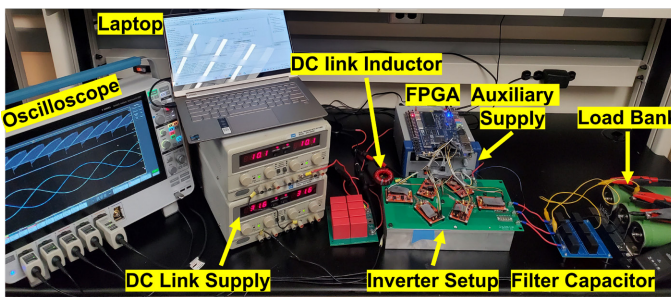
Integrating the system together, the inverter was run continuously at DC link voltage of 62 V, and V_{LL-RMS} of 110 V was recorded. The three-phase output voltage, switch node and input DC link voltage, and inductor current are presented in Fig. 203(a) and Fig. 203(b). Converter output voltages, input DC link voltage, and DC link current during start-up were also recorded as shown in Fig. 203(c). It is worth mentioning that testing a CSI is different from testing a VSI, as most of the commercially available power supplies can be high voltage but not high current. Therefore, testing a CSI ideally requires a constant current supply, which is uncommon. This limitation restricts rated voltage testing of the designed converter, as it requires a current supply capable of delivering ~ 75 A.



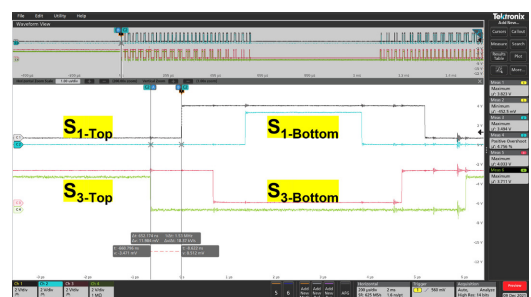
(a) DPT on a switching cell with two paralleled

(b) Ringing frequency of device voltage devices

Fig. 201.—Characterization of CS configured switching cell with two paralleled devices

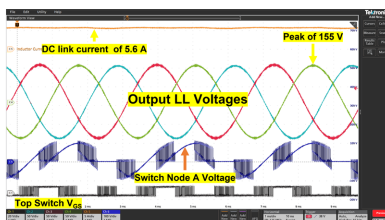


(a) Whole inverter setup

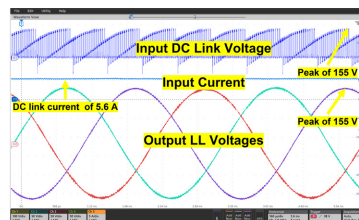


(b) PWM generation for top and bottom switches for a CS configuration

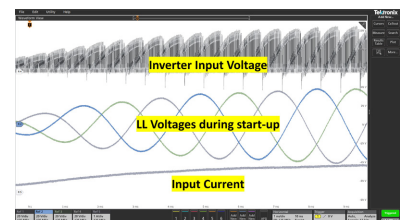
Fig. 202.—2L-CSI converter setup and PWM waveform



(a) Converter switch node voltage



(b) Converter output and DC current



(c) Converter response during and DC link voltage start-up

Fig. 203.—Test results of inverter at RT

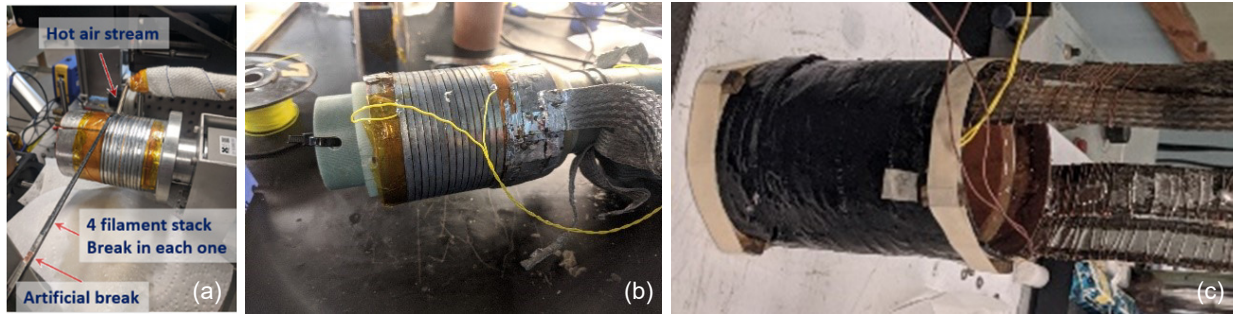


Fig. 204.—Development of SC inductor. (a) Coil winding process. (b) Wound coil with voltage drop sense leads. (c) Test coil after epoxy impregnation and mounting current leads.

3.2.3.4.9 Development of Superconducting Inductor

To demonstrate a SC inductor, L_{DC} was assumed to be split into two equal inductors ($12.5 \mu\text{H}$ each), as shown in Fig. 193(c). A copper tube of 3.93 in. (10 cm) OD was selected to prototype the initial version of the inductor. The tube was cut in half lengthwise and crosswise to avoid the SC current shunting. For the SC cable, 2-mm-wide exfoliated YBCO coupons manufactured from a standard second-generation tape supplied by American Superconductor Corp. were used (Ref. 196). The exfoliated tapes, initially 10-mm wide, were sliced into 2-mm wide filaments by a fiber laser (STI 80-W model). The filament can be described as a 1.2-mm-thick YBCO layer attached to 75-mm-thick copper foil. For inductor forming, 14-turn single-layer coil was accomplished, resulting in measured inductance of $13 \mu\text{H}$ with bode 100 vector network analyzers. The inductor value matched well with the designed value ($12.5 \mu\text{H}$ each), and two such inductors can be configured with opposing polarity relative to the DC link supply to make a current source configuration. The process of winding the coil is shown in Fig. 204(a), and Fig. 204(b) shows a wound coil with voltage-drop sensing leads. The finished coil was vacuum impregnated with Stycast 2850FL epoxy, as shown in Fig. 204(c), to have good thermal conduction. The inductor developed was successfully tested at 77 K for a minimum current of 600 A and thus can successfully be employed for converter applications.

3.2.3.4.10 Cryogenic Temperature Testing

Using the cryogenic converter setup illustrated in Fig. 205, converter tests were repeated under conditions similar to those used for testing at RT. For the experimentation, a polystyrene foam bucket was used to contain LN_2 , in which the output filter capacitor and the heat-sink were submerged. Switching devices of the power stage were mounted onto the heat-sink through a silicone elastomer TIM. While operating the inverter at similar input and load conditions, the overall inverter efficiency was measured to increase by 0.3%, which is expected to improve when the loading percentage increases. Using the efficiency measurements, the actual device junction temperature can be estimated using thermal resistances of all the series interconnects: R_{th-js} , R_{th-tim} , and R_{th-sa} . By using the manufacturer-provided values, device junction temperature under these conditions was calculated to be $\sim 82.34 \text{ K}$. The whole inverter, including GD board, DC link inductor, and output filter capacitor, was not completely immersed inside the LN_2 . This was done deliberately, as otherwise the process of debugging and taking measurements would have been extremely difficult. The difficulty is associated with draining and repouring of LN_2 back and forth into the polystyrene foam bucket while the converter components are already dipped inside. Furthermore, after every alternate cycle of RT and LN_2 exposure, the converter components go through excessive icing and deicing, which can lead to short circuiting of components if not properly attended, and also takes considerable time to dry up.

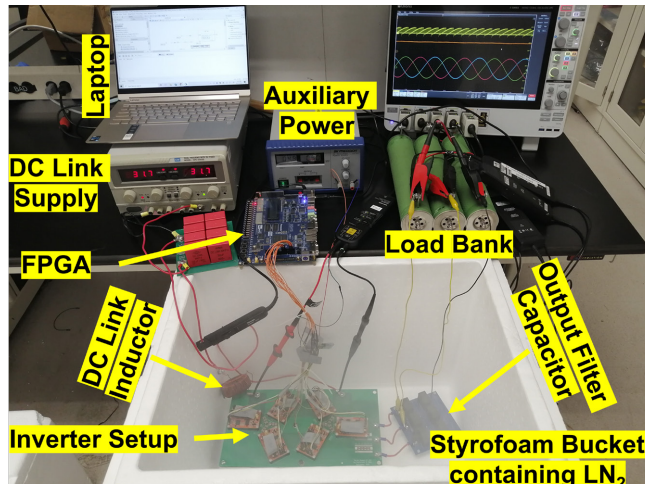


Fig. 205.—Cryogenic setup of inverter.

3.2.3.4.11 Discussion of Converter Performance at RT and CT

A three-phase 2L-CSI was developed and tested successfully at RT and CT with similar source and load conditions. An efficiency improvement of 0.7% was observed at load demand of 300 W. Although the idea of utilizing COTS components for a fully cryogenic converter was implemented, numerous challenges and avenues need to be addressed. From the experience of working with LN₂, it was realized that either cryo-chambers or cold plate (carrying liquid cryogen) must be designed for testing a cryogenic system. Such a setup will not only ensure reliable connections of components and easy measurements and debugging, but will also be critical from the perspective of safety. Therefore, proper design and sizing of cooling mechanisms must be considered in further development of cryogenic converters.

3.2.3.5 Discussion of Challenges in Developing CPEC

Although CPEC systems are expected to offer better efficiency and density, development of fully cryogenic converter systems is in the initial phases and needs considerable development and experimental characterization. The advantages associated with cryogenic operating temperatures remain attractive for power electronics systems, but such a configuration also poses numerous challenges that should be considered. Based upon the experience in designing a cryogenic converter, a few of these challenges are summarized here.

3.2.3.5.1 Device Selection for CPEC

Among active switching devices, Si devices have been shown to offer improved performance at CT and are also available in abundance, yet the reduction in device blocking voltage limits the device operating voltage. In comparison, GaN HEMTs have shown better performance among WBG devices in all the variables of interest, but the highest voltage COTS device characterized for CT is limited to just 650 V (Ref. 197). Considering this limit on device voltage, series cascading of devices could be a possible solution, but it leads to numerous other challenges. A few of these challenges include a larger power loop, asymmetrical dynamic performance of devices, and complex gate-driving circuitry. Therefore, the development of CPEC systems is limited to lower voltages until suitability of higher voltage GaN devices gets proven for cryogenic applications.

3.2.3.5.2 Uncertainty in Operation of ICs and APS at CT

One of the biggest challenges is availability and development of CT-compatible GD circuitry. Since the GD boards themselves need numerous components, careful selection and characterization of all the individual building elements must be given thorough consideration. The first and foremost challenge is to find CMOS ICs, as most ICs work on bipolar CMOS (BiCMOS), double-diffused metal-oxide semiconductor (DMOS), or other technologies. It is also very difficult to know which fabrication process and technology was used in IC manufacturing, as this information is not commonly available on device datasheets. The only way of finding the correct option is by carrying out rigorous testing and evaluation.

Finding a CMOS IC is only half the solution, as some ICs can possibly underperform or may not work at all for the intended CT operation. From the authors' experience, it was observed that not all similar components perform well at CT. For example, state-of-the-art isolated GD Si8271 was found to be working until 120 K in Reference 169, whereas it was observed to be working until 133 K in References 184 and 182. During the characterization of GD ICs, similar experiences were encountered by the authors. Upon replacing and recharacterizing, every other Si8271 IC gives the opportunity to be workable as low as LN₂ temperature. By comparison, the other GD ADuM4221 and buffer IC ISO7820x were found to be operational most of the time.

Similar characterization issues occurred with APS RV-0509S, as it has been found to malfunction sometimes. To further understand the operational boundaries of this device, the individual characterization at CT was repeated for at least three times for an extended duration of time spanning from 6 to 8 h. While characterizing is important, test conditions should also be extended to numerous load conditions, with and without being exposed to cold start operation. This characterization approach helped authors find the APS that can work together with GD boards constituting a full three-phase converter.

3.2.3.5.3 Inaccurate Temperature Assessment

Another challenge is inaccurate assessment of temperature, particularly with conduction-cooled cryogenic chambers. From the experimentation, it was found that if any of the components is left uncovered by epoxy coating, it might not have the same temperature as the actual baseplate of the chamber. This difference in temperature is governed by the power loss inside the equipment under test (EUT) and might be negligible if there is not much heat loss by the EUT. To avoid misreading, the entire PCB must be fully epoxy hardened to ensure thermal conduction of the entire EUT with the baseplate of the chamber. However, this temperature assessment is not an issue if liquid cryogenics are used for cooling purposes, such as immersion of EUT within large volumes of LN₂, which maintains a temperature of 77 K.

3.2.3.6 Challenges in High-Voltage Measurements at CT

One of the most important challenges is related to measurement of fast-rising/falling high-voltage electrical signals at CT. The challenge becomes more critical when the CPEC systems incorporate high-switching GaN devices. A typical example of this is measuring a device voltage V_{DS} inside a cryo-chamber or LN₂ dewar. Since not much has been reported on cryogenic measurements, numerous compensation networks were tried, but signal shape and integrity were found to be getting lost in this whole process. Finally, the jaws of an active probe were inserted inside the LN₂ dewar to take the measurements, but a better method for this should be researched.

However, low-voltage signals such as V_{GS} can easily be measured by employing SMA connectors, BNC cables, and 50 Ω external terminations. External terminations play an important role by damping the reflection and refraction of waves.

3.2.3.7 Cryogenic Inverter Design: Conclusions

In this section, the design, analysis and operation of a cryogenically cooled 2L-CSI inverter for fully electric aircraft was presented. The importance of CPEC systems in fully electric aircraft was emphasized and the converter design tradeoffs for such applications was reviewed. From the considered parameters, such as volume, power quality, and energy content, a 2L-CSI was found to be more appropriate when considering SC inductors. Therefore, a 20-kVA, 300-V, three-phase 2L-CSI with the most suitable WBG GaN device was designed and tested at RT and CT. The minimum device junction temperature was estimated to be ~ 82.34 K. The converter was successfully run at V_{LL-RMS} of 110 V at both RT and LN₂ temperature, and an efficiency improvement of 0.7% was measured under light load conditions.

As part of the converter development, numerous cryogenic subsystems, including GD boards and DPT platforms using COTS components, were also developed. Both boards were integrated, and their performance was successfully tested at RT and inside a LN₂, 77-K environment. Using the designed subsystems, dynamic characterization of GaN HEMT was performed for the first time, where the time delay for turn-off and turn-on was reduced by 100 and 20 ns, respectively, and an overall switching loss reduction of as much as 62% was observed at the device-rated current. This section provided a complete list of the COTS components utilized in the design and briefly discussed the challenges encountered in the development of a cryogenic inverter.

As part of future work, converter testing under rated conditions is ongoing, and results with SC inductors will be provided in future publications. Furthermore, the authors would like to extend the testing to even lower temperatures, such as ~ 20 K.

3.2.4 Design of an Improved Cryogenic Auxiliary Power Supply

As shown in Fig. 194, a typical GD consists of passive components, ICs, and an APS. The APS needs to be placed close to the other GD components and offer minimal isolation capacitance to maximize their CM transient immunity (CMTI). Therefore, the APS must be placed in the same cryogenic environment as the converter for optimal operation (Ref. 198). Numerous passive components and materials have been tested and documented at CTs; however, the same cannot be said for commercial APSs. In light of this, testing of APSs was carried out, and their failure necessitates the design of an isolated cryogenic auxiliary power supply (CAPS). This section proposes a design to address this issue using a CAPS. The goal of the proposed implementation is to verify the operation of the power supply individually at CTs. The CAPS implementation will be verified at RT and at CT. The goal for this implementation is to have CAPS integrated directly onto GD boards to minimize any transients in the design. A DPT of a GS66516T GaN HEMT will be used to compare operation of each implementation at RT and CT. The design of the DPT board and the GD circuit will need to be verified at CTs so that it can be used to test the CAPS design as seen in this section (Ref. 199). This GD and DPT board will be used to validate both implementations of the CAPS design. Ultimately, the updated implementation could eventually be used to run a GaN-based power electronics converter at CT.

A sampling of commercial power supplies from different manufacturers was carried at CT, and their performance and change in efficiency is presented in Table 30. As a result of experimentation, many APSs have reduced efficiencies or fail to operate at CTs. Those with reduced efficiencies mostly fail to have cold start capabilities, which means that the CAPS cannot be switched off at CTs. Therefore, a CAPS is needed that can match the performance of traditional RT supplies. The initial separate CAPS solution proposed is an isolated 5-W open-loop 5-9 V DC-DC converter that can serve as isolated power supply for a GD board. The GD board can be utilized to operate a GaN HEMT converter for CT. The proposed solution needs to meet characteristics of any APSs at RT, including stable operation, high efficiency, and low isolation capacitance alongside a small form factor. The initial solution comprises a fly-back converter operating at a 200-kHz frequency in an effort to shrink the magnetic design, with a 50% duty cycle. The fly-back topology was chosen because of its simple design and lower component count. To meet stable cryogenic operation, active and passive components that have beneficial or at least stable performance with temperature were used. For best isolation, a transformer winding optimization process was also carried out to enhance CMTI. A resistor-capacitor-diode (RCD) snubber was placed on the primary side of the winding to limit voltage ringing across the switching node. As part of the design goal, there was an effort to keep the volume of the CAPS close to that the COTS APS, which can be verified from Fig. 206.

TABLE 30.—COMPARISON OF COTS APS PERFORMANCE AT CRYOGENIC TEMPERATURES

COTS APS	Efficiency at LN ₂
PEM2-S5-D5-S	Reduced to 30%
PEME1-S12-S12-S	Reduced to 34%
MDS01M-12	Reduced to 24%
MDS01L-12	Failure
TRV 1-1212M	Failure
R05P209S	Inconsistent

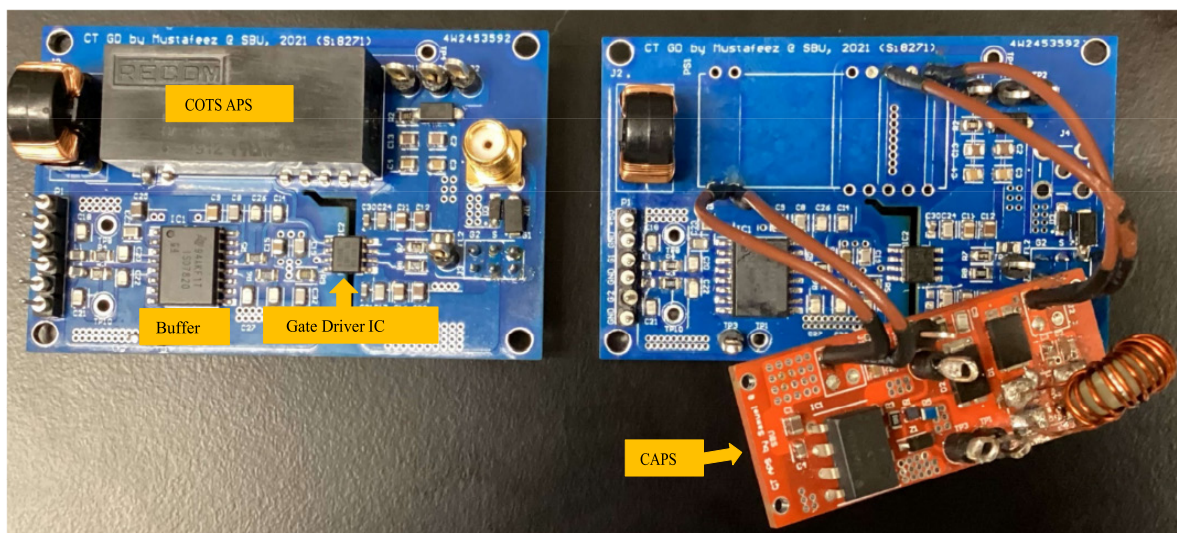


Fig. 206.—APS structure.

3.2.4.1 Components Selection and Design

To find components compatible with low temperatures, a literature review was carried out on control ICs, transformer core material, and associated active and passive components. The performance of Si and SiC MOSFETs and GaN FETs was taken into consideration while operating at low temperatures. Although Si MOSFETs perform well at CTs, they have lower switching speeds, whereas SiC MOSFETs have higher on-state resistance. GaN FETs have reduced on-state resistance and increased switching speeds at CTs (Ref. 139). Therefore, an 80-V GaN FET (EPC2039) was selected as a switching device. For the control circuit, CMOS-based ICs are most likely to operate at CTs because of their increased carrier mobility and saturation velocity (Ref. 169). Numerous control ICs were reviewed for their cryogenic operation. The TLC555CP from Texas Instruments was successfully operated at CTs and was verified to have cold-start capabilities. The test setup for cryogenic characterization is shown in Fig. 207. In addition to the control IC, thin-film resistors, ceramic NP0 capacitors, and Si Schottky diodes were selected for the control logic, snubber design, and output filtering. For the transformer core, a high-flux powdered core material was selected because of its stable permeability and unchanged losses at low temperatures. Throughout the design, all the constituent components were selected based upon their proven performance at CTs as found in the literature (Ref. 139).

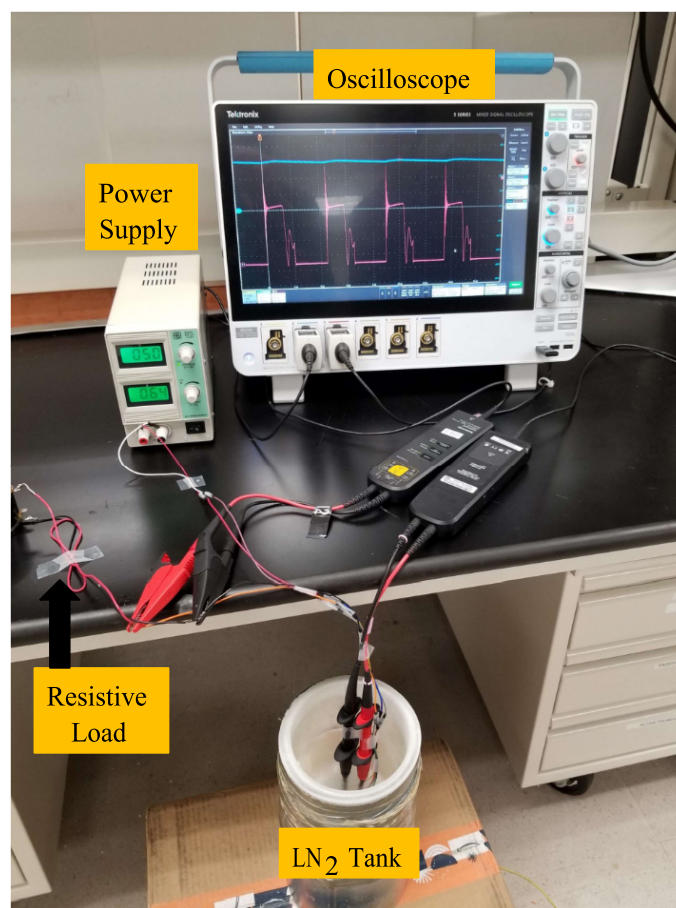


Fig. 207.—APS cryogenic testing setup.

3.2.4.2 Optimization

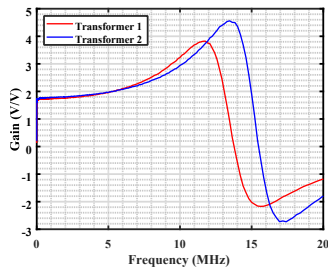
To minimize the switching device stress and leakage inductance, numerous snubber designs were tested and many transformer winding techniques were adopted. An optimized snubber design leads to lower switch node voltage ringing and thus lower device stress. For transformer optimization, two versions were found most suitable for the application. Transformer 1 had both primary and secondary winding layered on top of each other, while transformer 2 had both windings separated on opposite sides of the core. Part of the transformer optimization process involves the tradeoff between leakage inductance and winding isolation capacitance; these control the switch node ringing and CMTI, respectively. To better understand this phenomenon, the gain versus frequency response for both transformers, presented in Fig. 208(a), was measured using a vector network analyzer (Bode 100 by Omicron Labs). Results showed that higher leakage inductance leads to better CMTI, and vice versa. Taking both constraints into account, transformer 2 with separate winding was found to offer the lowest inter-winding capacitance. This capacitance is important, as it plays a vital role in increasing CMTI.

3.2.4.3 Hardware Results

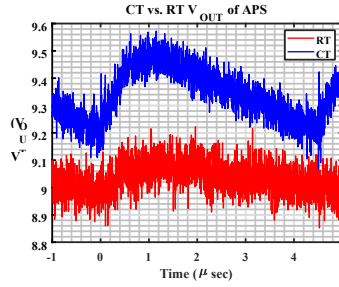
To test the performance of the CAPS design, both RT and cryogenic measurements were performed using the setup in Fig. 207. The output voltage V_{OUT} was measured using a resistive load to simulate the GD board. Fig. 208(b) shows that V_{OUT} increases both in value and ripple at CT, implying reduced device conduction losses and change in filter capacitors. Additionally, the duty cycle and switching frequency generated by the PWM control circuit can also be observed to vary with temperature. A pictorial depiction of that can be seen in Fig. 208(c). This change is explained by the variations in the TLC555CP's internal characteristics, timing resistors, and capacitors. The GaN FET's switching speed and dV/dt also increased at CTs. To justify the CAPS design for gate-driving applications in GaN-based converters, a DPT platform was established, as depicted in Fig. 209(a), to investigate the performance of the CAPS. The DPT board consists of a four-layer PCB implementing a vertical power loop with GS66516T HEMTs from GaN Systems Inc. as switching devices. NP0 decoupling capacitors and PPS DC link capacitors were also utilized in the DPT board, as they are compatible at CTs (Ref. 198). The complete DPT board, together with the proposed CAPS and gate board, is highlighted in Fig. 209(b). Device current measurements were taken using the SSDN-10 current shunt, and voltage measurements were taken using the TPP0850 passive probe with a bandwidth of 800 MHz and a MS056 oscilloscope. This DPT platform performed well up to the device's recommended voltage and current of 400 V and 60 A, as shown in Fig. 209(c).

3.2.4.4 Combined Implementation

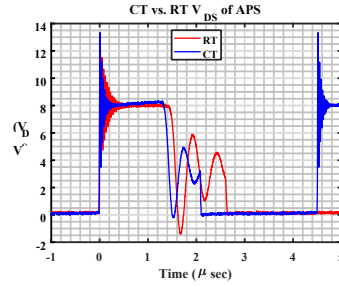
The combined implementation takes the separate CAPS/GD design and places both the CAPS and the GD circuit on the same board as seen in Fig. 210(a). This design should improve CMTI, as any leads connecting both boards are now minimized. To further improve efficiency, the switching frequency of the CAPS will increase to 1 MHz. This faster design is meant to further reduce the effective current seen by the primary side of the converter, which should decrease power loss from the core and the windings. The updated test setup for this design can be seen in Fig. 210(b).



(a) Transformer optimization

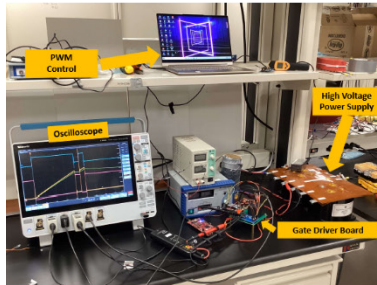


(b) Output voltage

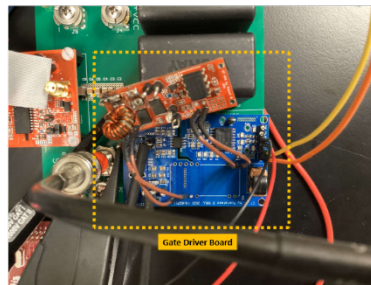


(c) Switch node voltage

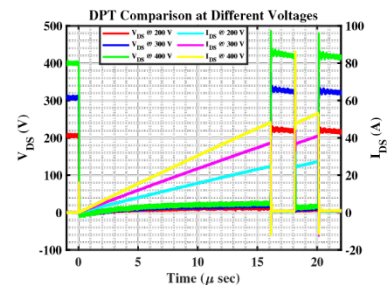
Fig. 208.—APS at RT and CT.



(a) DPT platform

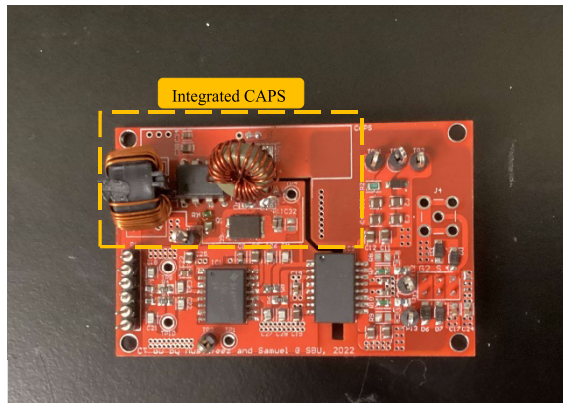


(b) CAPS with DPT board

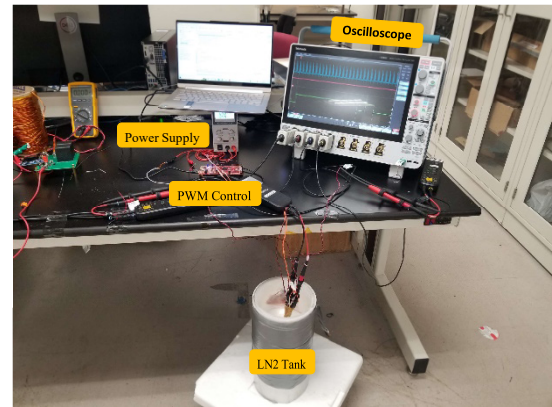


(c) DPT Results

Fig. 209.—RT characterization.



(a) CAPS and gate driver together



(b) Updated APS cryogenic testing setup

Fig. 210.—Implementation of combined APS.

As a result of the increased frequency, the RCD snubber needs to be optimized to handle the changes in voltage overshoot. Three different combinations of resistors and capacitors are tested with the goal to see how each component affects the voltage overshoot of the GaN FET. The results are compared to voltage overshoot of a GaN FET operating without an RCD snubber. The data show the snubber resistor is inversely related, and the snubber capacitor is proportional to the maximum overshoot. For example, halving the snubber resistor value and doubling the snubber capacitor value can lead to a lower maximum voltage overshoot, as seen with the combination tested in Fig. 211(a). The final values chosen for the RCD snubber were 750 Ω and 100 nF, as this was the combination that clamped the maximum voltage overshoot at 15 V. Transformer 2, with a separate winding, was chosen for this updated implementation, as it still provided the most CMTI through its lowest interwinding capacitance.

3.2.4.5 Updated Hardware Results

To test the performance of the updated CAPS implementation, a new testing setup was devised as seen in Fig. 210(b). Room and CT results were performed and plotted in Fig. 212. In both scenarios, the converter was driving the same resistive load that simulated a GD board. Similarly to the initial proposed CAPS/GD implementation, the V_{OUT} decreases at CT, as seen in Fig. 212(a). This larger change in V_{OUT} is because of the TLC555CP's increased frequency and duty cycle at CTs, shown in Fig. 212(b). As in the proposed CAPS/GD implementation, the PWN from TLC555CP has increased the frequency from 1 to 1.2 MHz and the duty cycle has increased from 50% duty cycle to around 60%. The reason for a larger voltage deviation in this implementation is due to the bandwidth limitations of TLC555CP at 1 MHz. Since 1 MHz is near the upper limit of the maximum frequency of the TLC555CP, small changes to the timing resistors of the TLC555CP lead to sizeable changes in the frequency and the duty cycle. The same variations in the internals of the TLC555CP, the timing resistors, and the timing capacitors, as seen in the proposed implementation, explain the changes in the time measurements at CTs.

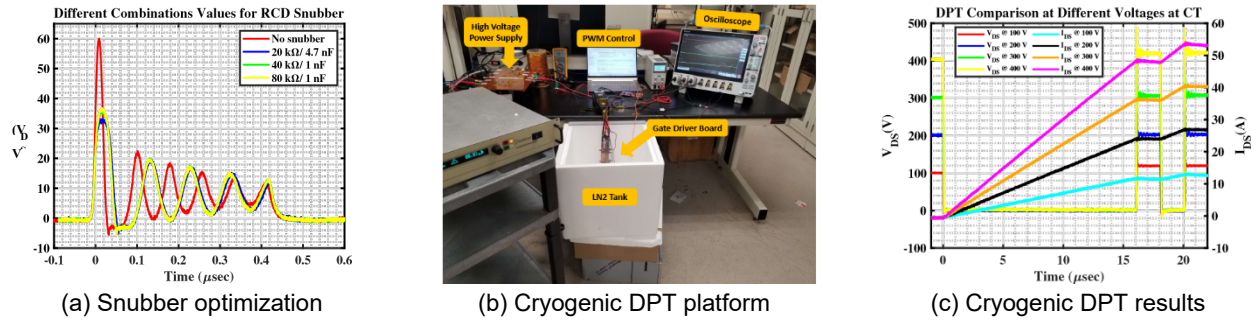


Fig. 211.—Cryogenic characterization.

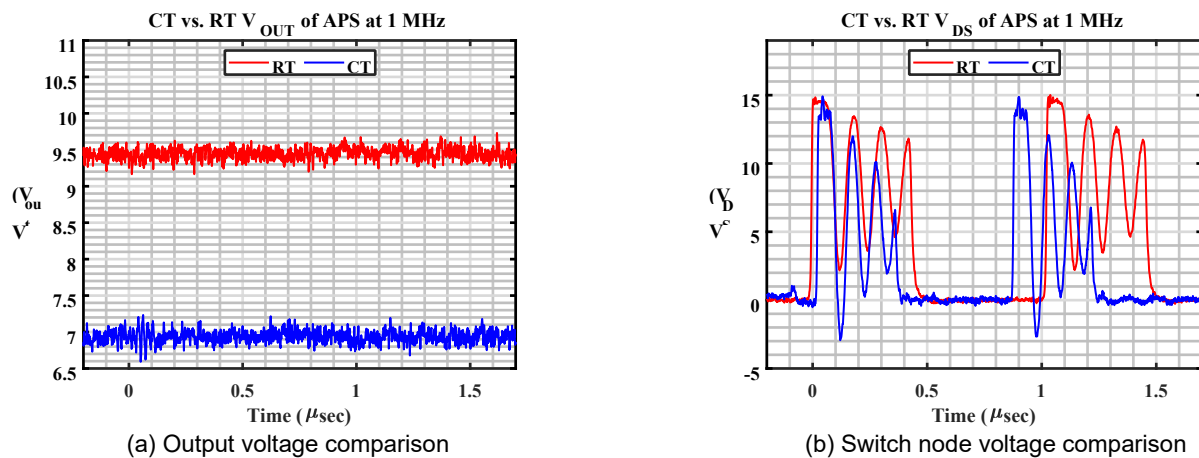
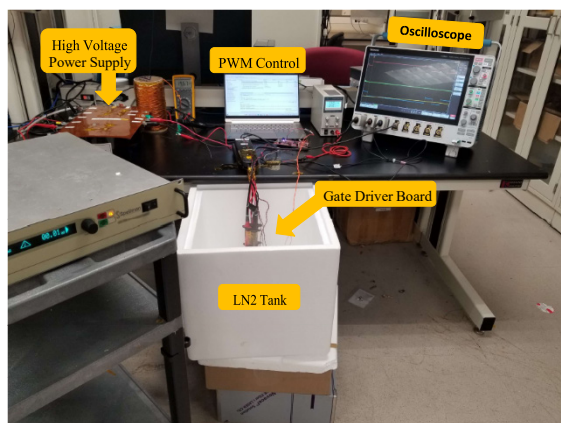


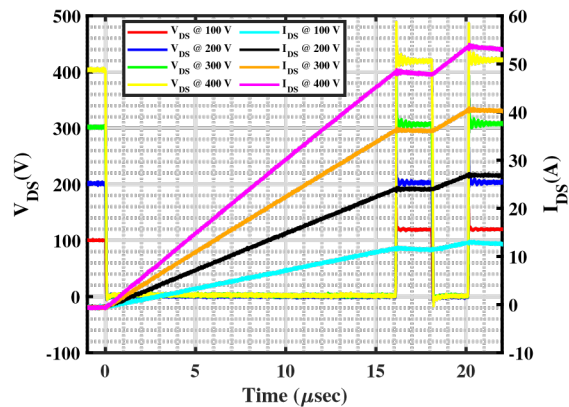
Fig. 212.—APS operating at 1 MHz at RT and CT.

The new DPT testing setup used for the updated implementation of the CAPS design is seen in Fig. 213(a). The DPT platform used in Fig. 213(a) uses the DPT test board as in the proposed implementation and therefore shares the same vertical power loop with the GS66516T HEMTs. The same NP0 decoupling capacitors and PPS DC link capacitors are used, and measurements for device current and voltage are taken using the same SSDN-10 current shunt and probes mentioned previously. The same MS056 oscilloscope is used for both RT and CT results. The DPT results at CT can be seen in Fig. 213(b). The DPT platform successfully performed up to the GS66516T's recommended limits of 400 V and 60 A at CTs. The rise and fall time of the GS66516T increases when compared to the RT results, as seen in Fig. 214(a) and Fig. 214(b). Faster rise and fall times for V_{DS} implies less crossover with the device current, leading to smaller switching losses at CTs.

The improved rise and fall times also allow for faster switching for the device, which leads to smaller packages and higher power density. These results validate the idea that both the initial proposed implementation and the updated implementation of an isolated CAPS design is able to replace many commercially available APSs for gate-driving applications in cryogenic environments.

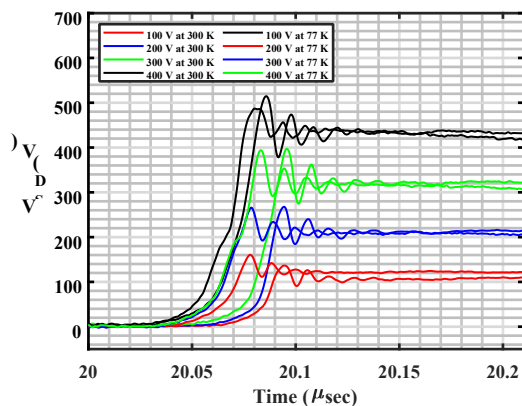


(a) Cryogenic DPT platform testing setup

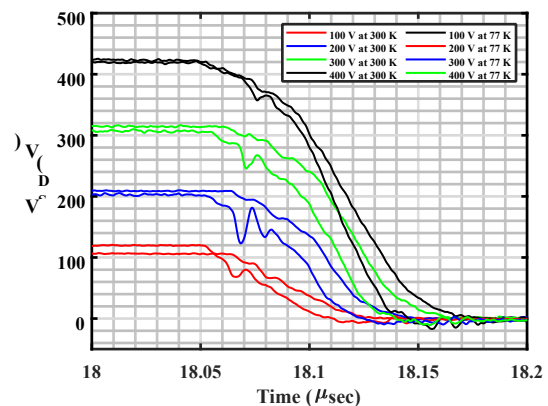


(b) Cryogenic DPT results

Fig. 213.—Cryogenic characterization.



(a) RT and CT turn-off differences



(b) RT and CT turn-on differences

Fig. 214.—Switching characterization.

3.2.4.6 Cryogenic Auxiliary Power Supply: Conclusions

This section proposed two implementations of a CAPS design using the fly-back topology for cryogenic applications (77 K). Each implementation was validated through a DPT of a GS66516T GaN HEMT. Both implementations were successfully run up to 400 V and 60 A at RT, and the updated CAPS design was validated up to 400 V and 60 A at CTs. This design can be improved through closed-loop operation, and further development and operation of the CAPS will enable wider adoption of cryogenic power electronic converters in the aviation industry.

3.3 Superconducting Power System

The use of a SC system provides an enabling, innovative approach to achieving high-efficiency power transmission at scales on the order of >10 MW. Owing to the zero ohmic losses produced by superconductors, the transmission current can be pushed to kA-scale levels across ultralight cables. Doing so also allows transmission voltages to be decreased to current standards, alleviating the challenges of partial discharge. The sizing, configuration, and circuit protection of the SC power system of the current study are described throughout this section.

3.3.1 Superconducting Power System Configuration

Power distribution system mass and heat parameterization studies were conducted for the CHEETA aircraft electric drivetrain, with relevant cable lengths shown in Fig. 215. This configuration supplies 22.56 MW to nine propulsor motors with a total cable to/from length of 151 m. Batteries are provided to boost the power required during takeoff and to cover transient loads; the fuel cells provide electric power for cruise and can recharge the batteries during cruise, if desired. High-ampacity power transmission is achieved with metal busbars, coaxial SC or metallic cables, and various connectors, which deliver current from the fuel cells to the batteries and between the batteries and the motors. The LH₂ serves a dual purpose, not only supplying the fuel cells but also providing cooling for the HTS lines, the HTS motors, and the power electronics.

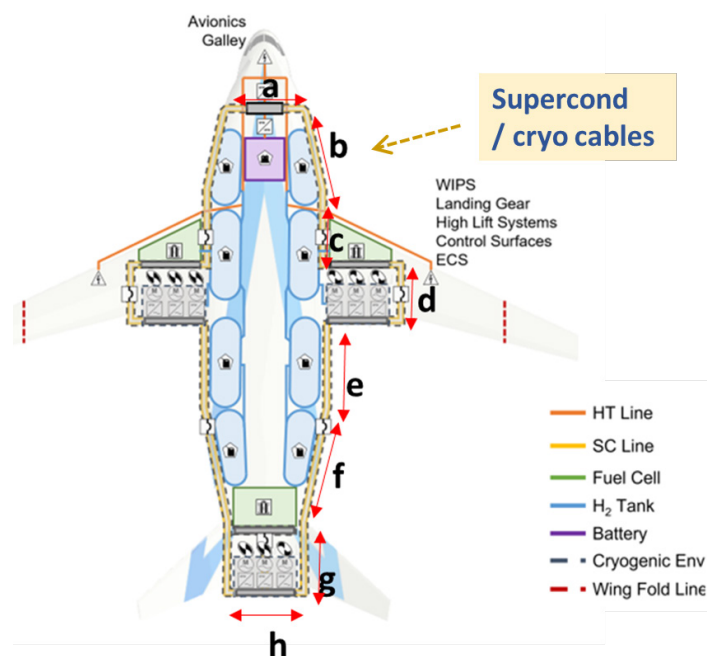
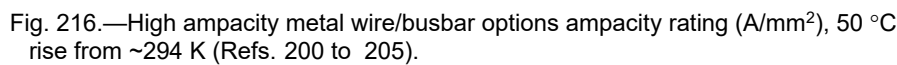


Fig. 215.—Power system configuration with superconducting cable lengths labeled.



The whole-wire current density of COTS metallic conductors is shown in Fig. 216 (noninsulated), which gives the maximum current the wires can carry in steady-state “on” condition. The current density is also considered the ampacity rating provided by the wire manufacturers, and it is determined and limited by the heat generation from Ohm’s law: $Q_{loss} = RI^2$, where R is resistance (Ohms) and I is current level (A). The ampacity rating of the wires is established as the values at which the ohmic heating can be controlled by ambient cooling without thermal runaway occurring and varies slightly depending on the maximum operating temperature that is allowable to be reached; typically this is 30 or 50 °C. The values of current density in Fig. 216 for varying ampacity were used in later sections for parameterization tables and full system analysis.

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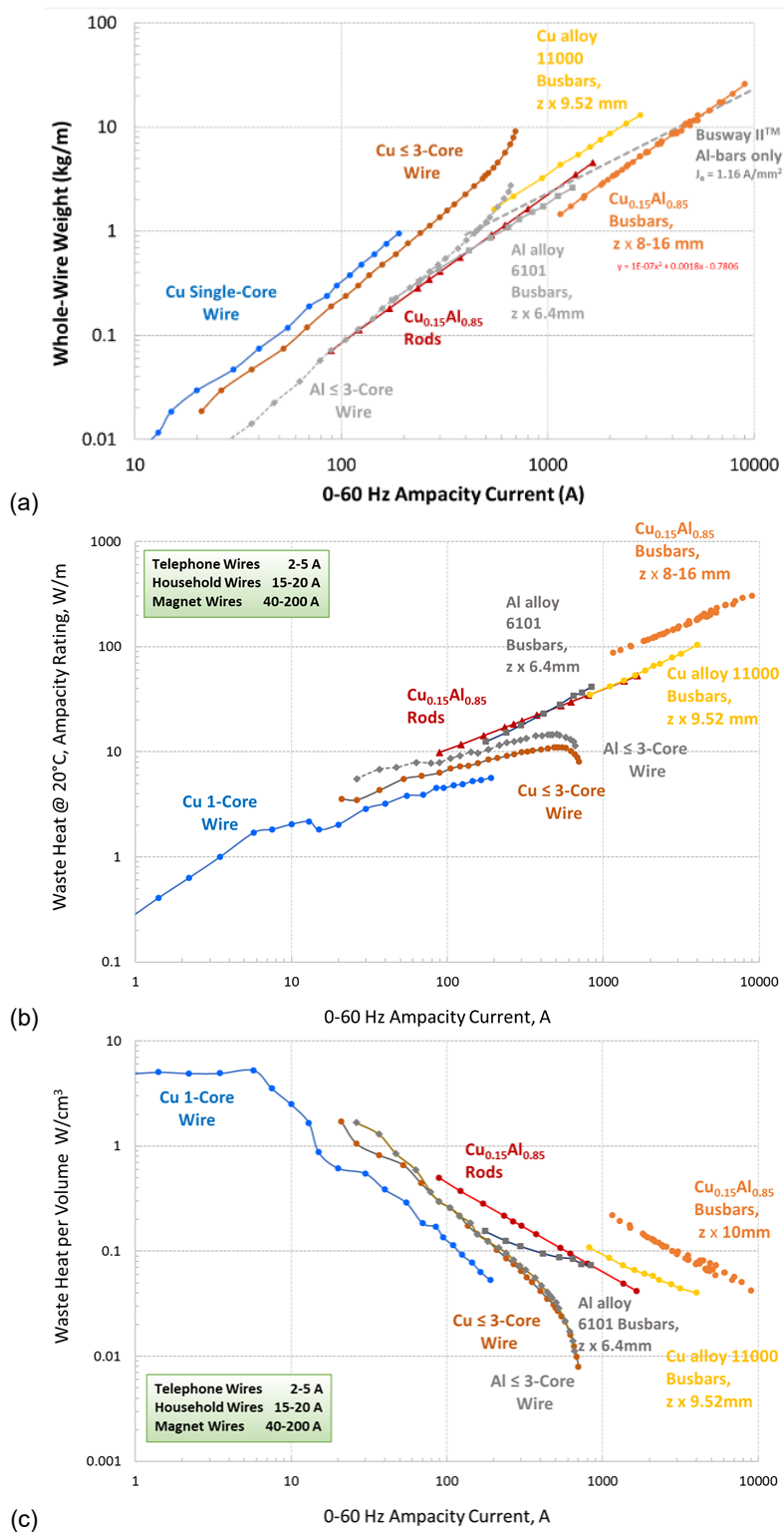


Fig. 217.—High ampacity metal wire/busbar options: (a) Mass (kg/m), (b) Waste heat (W/m) and (c) (W/cm³), with heat values calculated at maximum ampacity rating at 294 K (Refs. 200 to 205).

The effect of adding electrical insulation to wires and full cables is provided in Fig. 218 for voltage levels of about 0.6 to 35 kV. For 5-35 kV insulation, the weight of insulation can reach values as great as 60% of the entire cable, which must be considered. For electrical insulation, the thickness layer can be found by finding the minimal break strength for varying insulation layer thickness. And for true safety, generally the insulation thickness is increased to 5 times that amount for the final thickness. The weights of the I-line and busway systems commercially available are also shown in Fig. 218. The I-line and busway systems generally increase linearly as ampacity levels increase. However, at 35 kV, a linear increase is not observed, and the weight is mostly not dependent on ampacity of the wires, at ampacity values of about 1,000 to 5,000 A.

3.3.1.2 Power Distribution System Components: Electric Wires and Cables at 20 to 294 K

The technology option of the wire and cable technologies operating at 20 K for cryofuel H₂ are provided in Fig. 219 and are compared to standard Cu and Al wires at ambient temperatures (294 K) in Fig. 216 to Fig. 218. Hyper-Al is an interesting option at 20 K, with ultralow weights, almost as low as superconductors with Al stabilizers for quench protection, as shown in Fig. 219. However, hyper-Al wires carrying high currents still have significant ohmic heat loss from $Q = CI^2$ heating that would need to be cooled with liquid H₂ contained in cryoflex tubing. The cooling loads are substantial for 20- to 30-m cable lengths, and the options of cooling the cables with gaseous H₂ or He are not considered viable because of the high heat loads.

Large-current-capacity cables of Y-Ba-Cu-O achieve amazingly high values of $J_e \sim 5,000$ to $10,000 \text{ A/mm}^2$ at 20 K; the values at ~ 50 and $\sim 65 \text{ K}$ would be about 3 times and 5 times lower, respectively. Even higher values of $J_e \sim 100,000 \text{ A/mm}^2$ are achieved in single tapes. The performance and properties of a YBCO Conductor on Round Core (CORC[®], Advanced Conductor Technologies LLC) is provided in Fig. 220 and shows $\sim 8,900 \text{ A}$ can be achieved at 20 K with a cable of only 3.63 mm OD, and weighing only 0.036 kg/m. This value was calculated assuming an inner core of Al-99.999% wire for quench protection.

3.3.1.3 Power Distribution System Components: Current Leads at 20 to 294 K

The use of current leads is required in the drivetrain to bring high currents from batteries at RT down into the cryogenic environment. The design of current leads has been studied in great detail for varying conditions, as provided in Fig. 221 (Refs. 206 to 207). The heat loss of the current leads is plotted in Fig. 221 for varying steady-state on current levels and number of stages. If desired, the heat loads can be reduced significantly by increasing the number of stages. However, it is unknown if the added complexity and cost would be wanted by the electric power system or aircraft manufacturer.

The heat loss shown is ohmic heat loss generated in the metal conductors entering the cryogenic vessel that must be accommodated by the cooling system. This is typically done with liquid N₂ on the outside of the current lead and, for some designs, by evaporative cooling of cryogenic fluids such as LN₂, LHe, and LH₂.

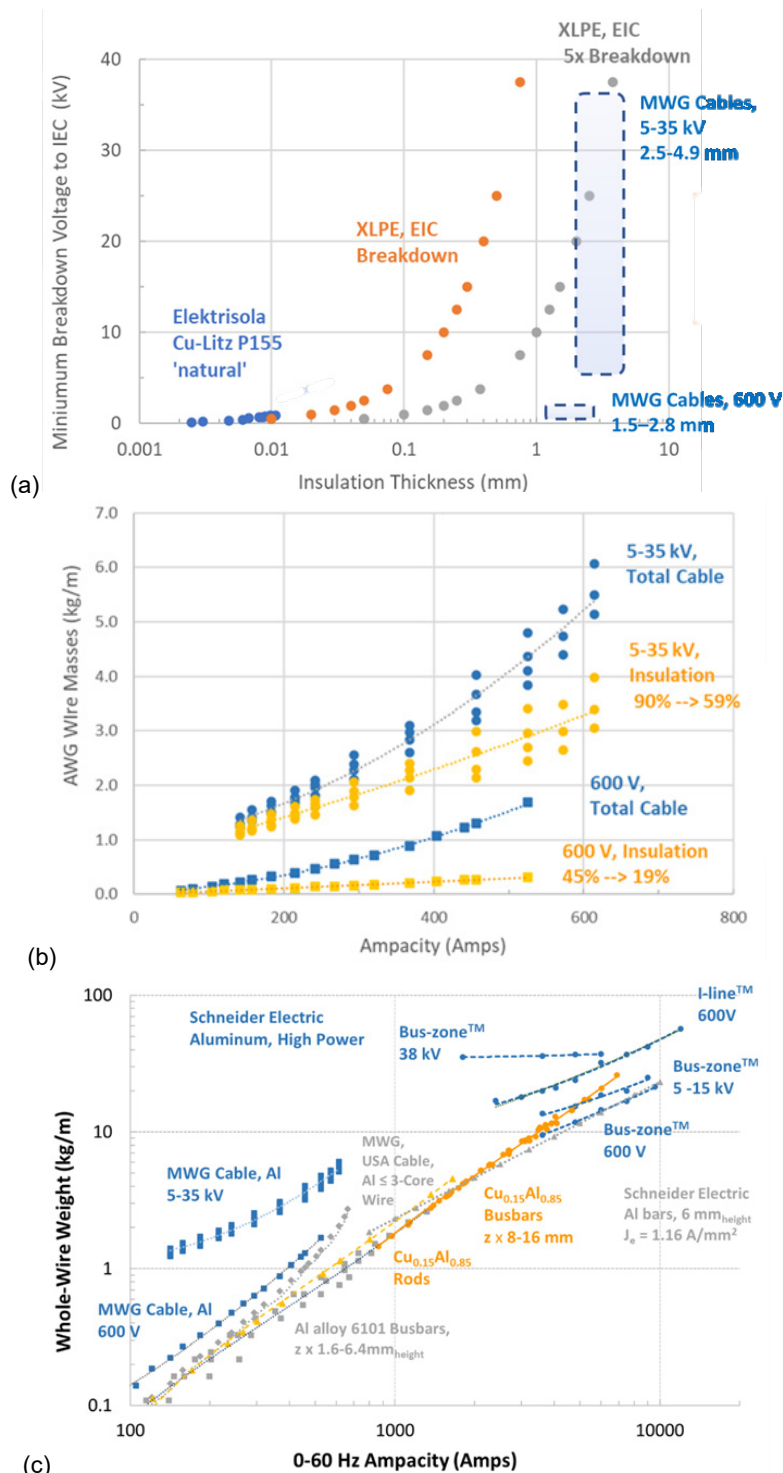


Fig. 218.—High-ampacity metal wire/busbar options at 294 K: insulation, wires and cables, and integrated systems for varying voltage levels 0.6 to 38 kV: (a) electric conductor insulation thickness required to prevent breakdown, (b) mass of high-voltage Al cables, and (c) wire weight of high-power Al busbars.

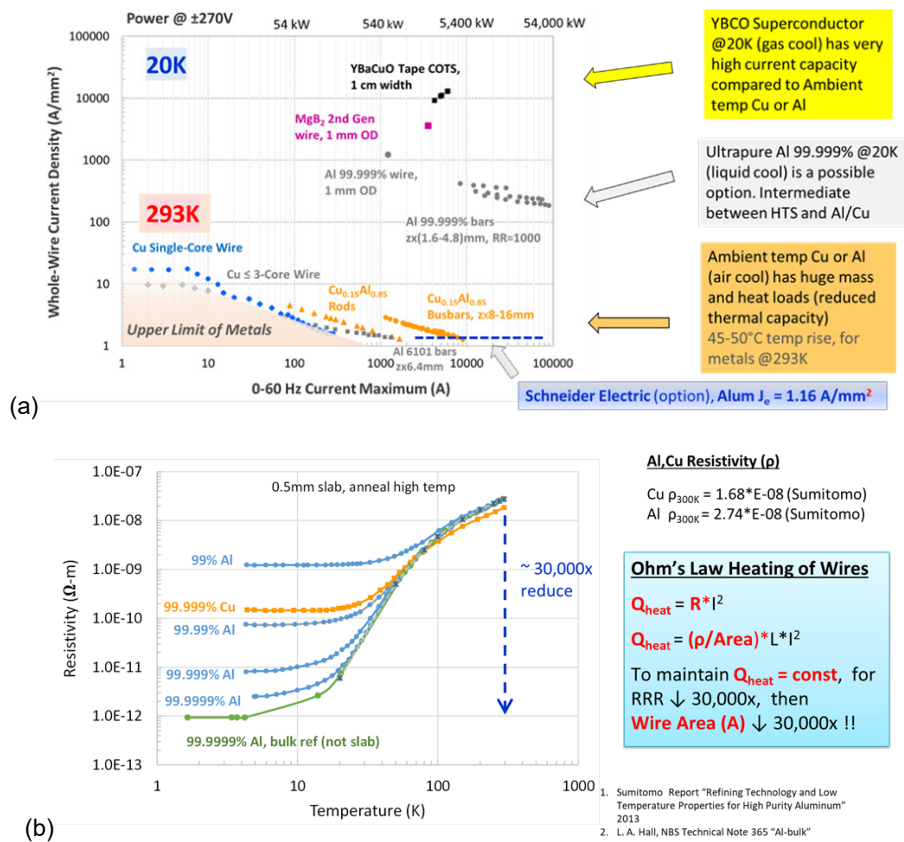


Fig. 219.—Electrical and physical properties of select wire and cable technologies, to operate at 20 to 77 K (Refs. 200 to 202), (Refs. 208 to 212). (a) Ampacity (A/mm^2) of busbar options. (b) Resistivity ($\Omega\cdot m$) of metal wire with reduced temperature.

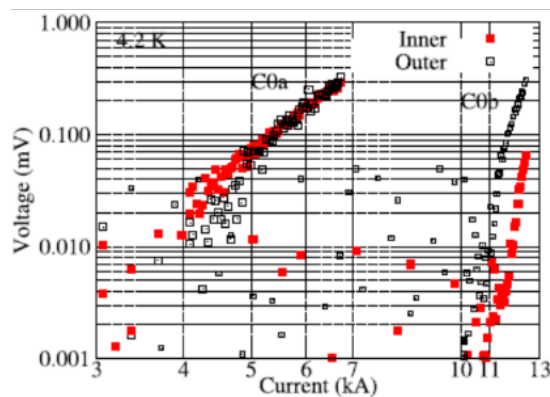


Fig. 220.—Electrical and physical properties of Y-Ba-Cu-O.

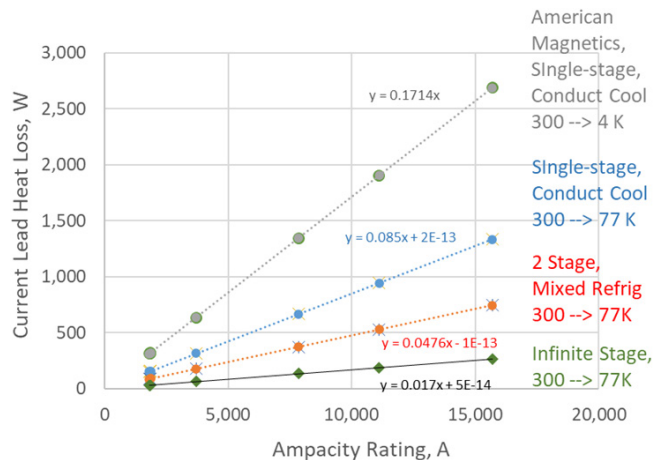


Fig. 221—Current lead heat loss versus ampacity rating for a single pair comparing single stage and 2 stage conduction cooling down to 77 K and 4 K (Refs. 206 to 207).

3.3.1.4 Electrical Wiring Interconnect System (EWIS) – System Level Calculations

The summary weight and heat loss parameterization for various components of the power distribution system are provided in Fig. 222, consisting of SC cables, current leads, Tee-connectors, and cryoflex tubing. A summary of mass and heat loss for various power distribution components of fixed and variable length for the CHEETA configuration can be seen in Fig. 223. By far, current leads consisting of either Cu-clad Al (Cuponal™, Cu_{0.2}Al_{0.8} or Cu_{0.15}Al_{0.85}) or Cu and cables consisting of either Cuponal™ or hyper-Al (purity ≥ 99.999%) account for the majority of the weight and heat loss, for a cryogenic cable system.

Table 31 and Fig. 224 compare the three technology choices for the cable: Cuponal™ at 294 K, hyper-Al at 20 K, and SC options of YBa₂Cu₃O_{7-x} (YBCO) at 20 K. The reduction of mass and loss for hyper-Al and YBCO at 20 K, is quite dramatic compared to Cuponal™ at 294 K, with the SC cables resulting in approximately 30 to 100 times reduction of both weight and heat loss. Estimates of the other properties of the system, including cost, system complexity, TRL level, and protection risks, are also provided in Table 31, which shows that the benefits of lower weight and less heat loss when utilizing YBCO components does come at increased cost, with progress needed in advancing TRL levels for aircraft application. Fig. 224 visually compares the mass and heat loss for bus materials of Cuponal™, hyper-Al, and SC YBCO, with YBCO achieving significantly lower weight and heat loss for these three options over a wide range of voltages from 0.6 to 35 kV. It is important to note that the mass of the YBCO or Al-hyperconducting systems can be approximately 4 to 5 times lower, even at ±270 V, than the 35-kV cable system made with Al/Cu. The full mass and heat loss of full cable systems at 20 K or ambient temperatures at 294 K have not been fully studied and are not available yet. However, the initial analysis outcomes provided in Fig. 224 demonstrate the large sensitivity of mass and thermal properties of the power transmission system to superconducting and hyperconducting material use.

Battery, Fuel-Cell → Motor

Pink = large mass, heat loss

Power Transmission Cable Component	Mass Parameterization	Heat Loss Parameterization	Reference, Comments
Metal Cable + Conduit ~ 300K, single	Al + Busway™ Mass(kg/m) = 0.0019*Wire(A)+2.529**	Heat-Loss(W/m) = 0.0437*Wire(A)+45.14	www.se.com , busway specs
Current Lead 300K → 77K	Cu ₂ Al _{0.8} pair Mass(kg/m) = 0.0062*Wire(A)-3.8606		1st estimate, Cuponal current lead
Current Lead 300K → 77K	Cu, pair mass not provided	Heat-Loss(W) = 0.085*Wire(A)	1-stage, R.McFee, Rev Sci Instr. 30 , 98 (1959), asked for mass Cu/Bronze
Supercon Lead 77K → 20K	BSCCO ?, pair Mass(kg) = 0.00066 *Wire(A)	Heat-Loss(W) = 0.00034*Wire(A)	HTS 110, private communication
Metal to Supercond Joint	Cu to YBCO, pair ~ zero, cable plugs insert into lead	Heat-Loss ~ 3-10W, 5-20 mOhm@20K, 15kA	Adv Cond Tech., EUCAS 2019
HTS Cable, single, "to"	YBCO CORC™ Mass(kg/m) = 4.0438 E-6 *Wire(A)	DC loss = 0, AC Loss Transients?	X. Wang, SuST 31 , 045007 (2018), info.,
HTS Cable, single "from"	YBCO CORC™ Mass(kg/m) = 4.0438 E-6 *Wire(A)	DC loss = 0, AC Loss Transients?	" 8.9kA@20K, 3.63mm OD, 2.56 mm ID; Al
Hypercond Al Cable "to"	HP Al Mass(kg/m) = 1.94 E-5 *Wire(A)	Heat-Loss(W/m)=0.007155*Wire(A)	C. Oberly, IEEE Tr. Magn. 27(1) , 458 (1991)
Hypercond Al Cable "from"	HP Al Mass(kg/m) = 1.94 E-5 *Wire(A)	Heat-Loss(W/m) = 0.007155*Wire(A)	C. Oberly, IEEE Tr. Magn. 27(1) , 458 (1991)
T-Connector, HTS to Metal	CORC™ to Cu/Al Mass(kg) = 0.0006*Wire(A)	Heat-Loss(W) = 0.00123*Wire(A)	AFRL Calc. @30K; 50K is ~4x higher
Cryoflex Tubing	OD=34 mm, ID= 14 mm Mass = 0.50 kg/m	Heat-Loss = 0.5-0.63 W/m @ 77K → 20K	Nexans COTS, other options possible

13,900 Amps for one loop of 22.5 MW, and $\pm 270V$

Fig. 222.—Mass and heat loss parameterization for various power distribution components.

	Power Transmission Cable Component	Bus Power (kW)	Bus Voltage (V)	Bus Current (A)	Piece Length (m)	Piece Mass (kg/m)	Piece Heat (W/m)	Piece Mass (kg)	Piece Heat (W)
Depends on Length	Metal Cable + Conduit ~ 300K, single "to"	AI with Busway™	7500	540	13,889	75.46	28.9	652	2,182 49,206
	Metal Cable + Conduit ~ 300K, single "from"	AI with Busway™	7500	540	13,889	75.46	28.9	652	2,182 49,206
	Metal Cable Conduit, pair		7500	540	13,889	75.46			
	HTS Cable, single, "to"	YBCO CORC™	7500	540	13,889	75.46	0.06	0	4 0
	HTS Cable, single "from"	YBCO CORC™	7500	540	13,889	75.46	0.06	0	4 0
	Hypercond AI Cable "to"	HP AI	7500	540	13,889	75.46	0.27	99	20 7,499
	Hypercond AI Cable "from"	HP AI	7500	540	13,889	75.46	0.27	99	20 7,499
	Cryoflex Tubing	OD=34 mm, ID= 14 mm	7500	540	13,889	75.46	0.50	1	38 48
				Subtotals		59.0	1,504	4,451	113,458
					# Pieces	kg/piece	W/piece		
Fixed (pieces)	Current Lead 300K → 77K	Cu ₆₂ Al ₃₈ Cuponal pair™	7500	540	13,889				82.3
	Current Lead 300K → 77K	Cu, pair	7500	540	13,889				1,181
	Supercon Lead 77K → 20K	BSCCO 7, pair	7500	540	13,889				9.17 3.57
	Metal to HTS Joint	Cu to YBCO CORC™, pair	7500	540	13,889				0.00 3.57
	T-Connector, HTS to metal	CORC™ to AI/Cu @30K	7500	540	13,889	18	1,600	3,075	28.80 55.35
				Totals	Cu ₆₂ Al ₃₈ Cuponal™ Cable			4,364	98,413
					AI Hyperconductor Cable			199	16,288
					YBCO CORC™ Cable			166	1,291

CHEETA V03
 22.5 MW, +270 V,
 13.9 kA, 150.3 m

Almost
 'zero' mass

>70% of
 supercond/cryo
 system

Heavy,
 Hot!

Fig. 223.—Summary of mass and heat loss for various power distribution components of fixed and variable length.

TABLE 31.—COMPARISON OF MASS AND HEAT LOSS FOR CUPONAL™, HYPERCONDUCTING AL, AND SC MATERIAL OPTIONS

Parameter	Al Busway™ at 294 K	Al > 99.999% hyper- conductor at 20 K	YBaCuO or MgB ₂ super- conductor at 20 to 65 K
Weight	4,365 kg	199 kg	166 kg
Waste heat	98 kW	16.3 kW	1.29 kW
Cost	Medium	High	High
Complexity	Low	Medium	Medium
TRL	9	4	4 (aircraft) 9 Organisation Européenne pour la Recherche Nucléaire (CERN)
Protection risks	High	High	Medium (Fault current limiter, FCL, intrinsic)

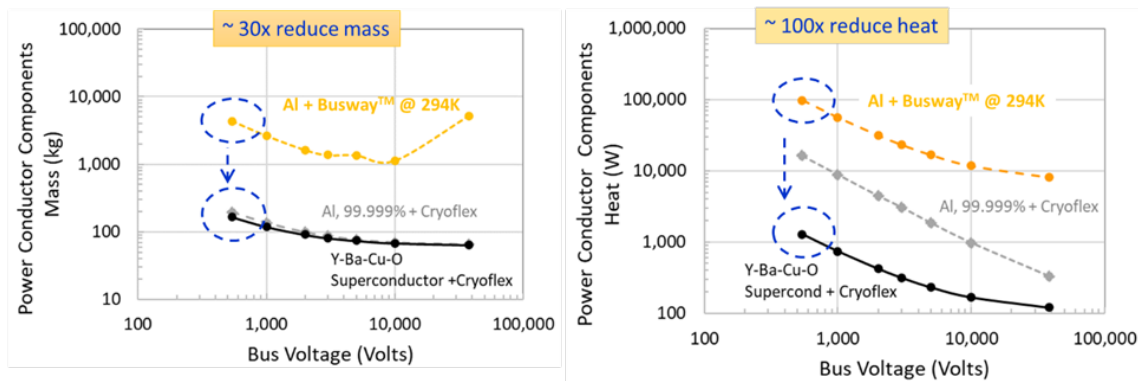


Fig. 224.—Comparison of mass and heat loss of Cuponal™ versus YBCO superconductor for bus materials.

3.3.1.5 Superconducting Power Distribution: Conclusions

The current study provided an initial understanding of the system-level properties of electrical power distribution systems for CHEETA for a 22.5-MW power level. This program considered most of the major power components and also included circuit breakers (see the following sections). Some important lessons learned are provided as follows, with the specific properties of CHEETA V03 architecture considered:

- Differences of weight and performance of components can be relatively easy to study and understood; however, the differences of full systems are not as easy to estimate a priori without full analysis and system integration.
- The use of superconductors at 20 K for electric wires becomes arguably the best technology option once the complexity and performance of the thermal-management system (TMS) are better understood. The total lack of ohmic thermal losses even for ultrahigh ampacities makes it much easier to design the TMS, and only gas cooling is required to handle the 1-kW heat loads resulting mostly from the current leads. The safety and quench protection systems of superconductors are the main challenges.
- The use of an Al-99.999+% hyperconductor at 20 K is considered the next best technology option. It is ultralight, which is very similar to SC options. However, the heat loads of the Al-hyperconductor vary from 16 to 29 kW, which is not trivial to manage in the long length of small-diameter cryoflex tubing. Generally, larger diameter cryoflex tubing is needed to allow the large coolant volumes and flows that also increase significantly as the cable length increases, and extra pieces are required to mitigate or control possible second-phase cooling. The added complexity and system weight to satisfactorily cool ~16 to 29 kW of waste heat from Al-hyperconductors are provided Section 3.3.2.2.
- The use of Al/Cu composites at 294 K (ambient temperatures) is the heaviest and highest loss option for the electric cables. Waste heat loads were very high, from 98 to 155 kW, and the TMS system to handle this amount of low-grade, low-temperature waste heat load has not even been roughly estimated yet. Also, the weights range from 4,400 to 10,000 kg, not including system pieces needed including insulation, conduits, connections, and very large volume considerations. Those added pieces could increase the system-level weights by roughly 2 to 3 times.
- Operation at ± 270 V at 20 K is considered to be a viable option for superconductors and Al-hyperconductors. The first system-level calculation showed approximately 30 to 100 times reduced weight and heat loss as compared with Cu/Al at 294 K.
- Scaling for all components to higher powers is not necessarily linear – the designer and operator must be aware of the operation complexities of each component (e.g., current leads).

- The design, operation, and integration of components also have complexities. Many trade options are usually possible, for example, to reduce waste-heat loss by increasing mass, or vice versa.

Further work is recommended as follows:

- Additional work is recommended to develop lower mass cryoflex tubing, which dominates the mass.
- Additional work to better understand and optimize current leads and junctions would be helpful.
- The full integration of the TMS is needed to understand how this added complexity will affect the system-level design, weight, volume, and performance.

3.3.2 Electrical Wiring Interconnection System (EWIS)

With the previous analysis of various materials for high-power transmission across the CHEETA vision vehicle concept, the integration of these wire materials into an interconnected power system requires additional consideration of electrical, thermal, and mechanical properties of the configuration. This work includes analysis of electrical insulation systems, terminals and terminations, and cryostat installation requirements.

3.3.2.1 EWIS Background

Both SC and non-SC (cryoresistive) materials have been investigated for use in the power distribution system. Both of these options require a different paradigm of thermal management and interconnections. Many decisions have been made during this program on different layouts of the EWIS, and systems masses have changed accordingly.

Shown in Fig. 225, the fuel cell power generation is spatially divided into three subgroups (each 7.5 MW), and each of these fuel cell sections powers three 2.5-MW motors. Also, there is a main distribution network capable of distributing 22.5-MW throughout the propulsion systems. Using standard Cu metal conductors, the maximum permissible current density of force air-cooled bus bars does not exceed 3 A/mm². This would mean the metal in a single-pole, ± 270 -VDC, 22.5-MW main distribution cable would have a cross-sectional area of $\sim 28,000$ mm² and a linear mass density of ~ 250 kg/m. Insulation thickness can be calculated by the following equation, which includes a safety factor, dielectric breakdown strength, and outer radius of the metal conductor.

$$\text{Insulation Thickness (m)} = r \times \left(e^{SF \times V / r \times BDS} - 1 \right) \quad (55)$$

Because the voltage for the CHEETA configuration is low at 270 VDC (unlike other proposed designs that can reach 10 kV), the insulation required is thin. For ethylene propylene diene monomer (EPDM) rubber and a safety factor of 100 $\times$, the linear mass density of the insulation for 270-VDC 2.5-MW cable is only ~ 1.3 kg/m. As shown in Eq. (55), the operational voltage and safety factor are in the numerator in an exponential and the cable radius is in the denominator. As a result, if 22.5-MW main feeder cable is designed with a much higher 10-kV operating voltage (2,250-A cable, 6.7-kg/m conductor), the cable radius would shrink and the insulation thickness would increase dramatically. For a 100 $\times$ safety factor, the linear density of the insulation would be over 72 kg/m. For a safety factor reduced to 10 $\times$, this would be a more reasonable 1 kg/m (and 3 mm thick).

This, of course, only tells part of the story because of changes in mass in components (e.g., protection systems) outside of the cable with changes in voltage. A NASA report from 2015 (Ref. 213) talks about the mass changes in other components in an electric aircraft with a majority SC EWIS. Lower voltages are desirable for a multitude of safety reasons, whether partial discharge) or insulation degradation (whether from environmental factors or partial discharge). At higher altitudes, the breakdown strength of

air decreases to a minimum before increasing again as vacuum is reached. This is known as a Paschen curve; see Fig. 226 for example curves as a function of gap distance between two electrodes. The increase in breakdown voltage at very small gaps is related to the mean free path of an electron in the air and the number of collisions required for an avalanche breakdown.

It is clear 270 VDC is near the minimum of the Paschen curve over a range of gap sizes and altitudes, and therefore will be much safer than any higher voltage. 270 VDC also has been proven in multiple aircraft in operation today, including the Boeing 787 and certain military aircraft (Ref. 214). If an arc does occur, due to the extreme power of these cables, the damage would be catastrophic.

During previous iterations of the CHEETA aircraft EWIS during this program, we investigated various voltages and currents. With the latest design of ± 270 VDC, the electrical current of different power cables is determinable. For the V03 architecture, the current is 9,260, 27,780, and 83,330 A for 2.5, 7.5, and 22.5 MW main feeder cables, respectively. If the current density to these power distribution cables and buses is not increased, the mass of the EWIS is unsustainable. Therefore, integration of cryoresistive metals and/or SC composites has been paramount for this program. Nano-carbon consolidations, such as carbon nanotube yarns, have been disregarded for this program as they have yet to reach the mass specific resistivity required (Ref. 215).

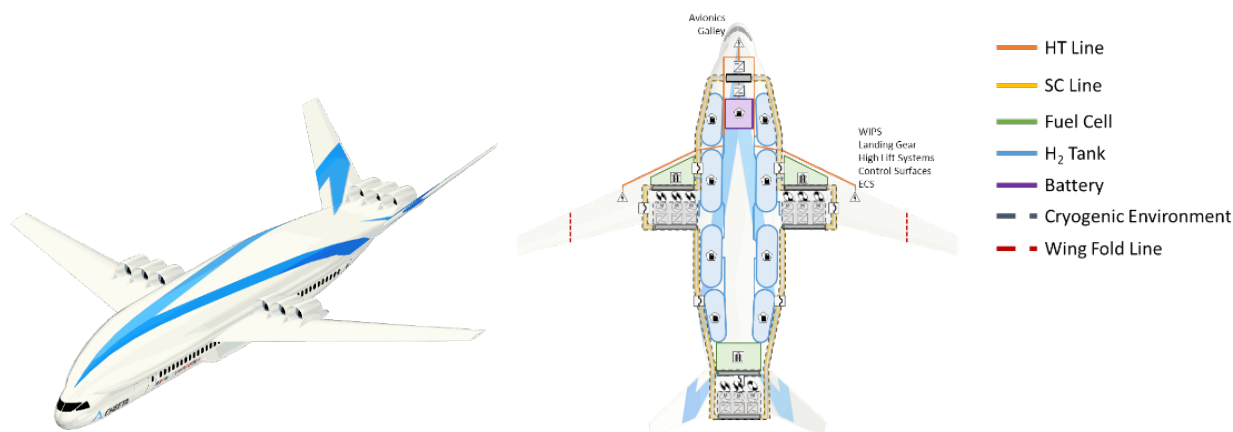


Fig. 225.—Exterior and interior cryogenic, power, and fuel systems of the CHEETA aircraft.

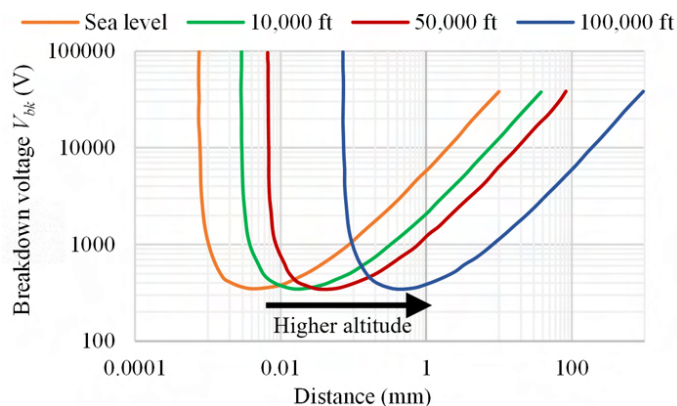


Fig. 226.—Paschen curve; decreased breakdown voltage of air at different altitudes (Ref. 216).

3.3.2.2 Cryoresistive Cable and Thermal Management

3.3.2.2.1 Benefits of Cryoresistive Cable

The electrical resistivity of non-SC metals decreases with temperature and will eventually reach the residual resistivity limit. The residual resistivity limit is determined by the crystallinity (dislocations, grain boundaries, vacancies) and purity of the metal, and the limit decreases to lower resistivities for higher crystallinity and higher purity metals. Residual resistivity ratio ($RRR_{273/T}$) is defined as the ratio of the resistivity at RT divided by the resistivity at another temperature shown as T . For very high RRR metals, boundary effects can also increase resistance, as these will function as effective scattering sites for high mean free path electrons. Fig. 227 and Fig. 228 show the resistivity of some relevant metals versus temperature and the mass specific resistivity (mass density multiplied by electronic resistivity; small is good) respectively.

AlBeMet 162H was included in this investigation because of its high performance at intermediate CTs, making it an excellent choice for lightweight current leads that extend out of the cryogenic environment. The “Bruggeman Calc” shown in Fig. 228 is for the theoretical value for a homogenous Al-Be composite of the same mass ratio as that in AlBeMet 162H nanocomposite. Note that high-purity Al conductors (5-9s purity) have lower resistance than 1,000 RRR Cu at 20 K, and because of previous developments by the U.S. Air Force in the 1970s to the 1990s in making ultrahigh RRR Al composites, we have selected HPAL (high-purity Al, known also as hyperconducting Al or hyper-Al) as our cryoresistive cable material.

Cryoresistive materials can safely carry current densities nearly commensurate with those of conventional SC composites (Ref. 217) and are less prone thermal instability (Ref. 218). However, unlike SC composites, they will generate significant amounts of Joule heating. Because of this, a high current density cryoresistive cable will require direct cryogen fluid cooling. This involves utilizing a portion of the specific heat (if subcooled or slush LH_2), latent heat of vaporization, and para-ortho conversion heat of cryogenic LH_2 . Currently there are commercial aerospace regulations that prohibit the spatial proximity of fuel and electrical systems. These regulations could be changed in future work, or to follow them, a closed-loop HEX system could be used throughout the power distribution network. Suitable primary coolants could be Ne, Ne-He-Ar mixtures, and pressurized He (with LH_2 as the secondary coolant).

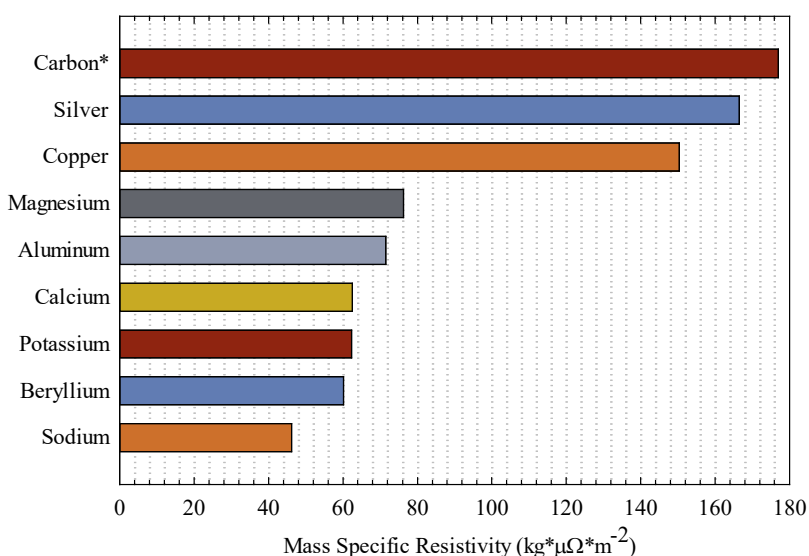


Fig. 227.—Mass specific resistivity is a useful metric for aerospace electrical conductors (equal to mass density multiplied by resistivity) and is shown for some of the best materials known. “Carbon*” denotes useful carbon consolidations, not individual nano-carbon structures (Ref. 219).

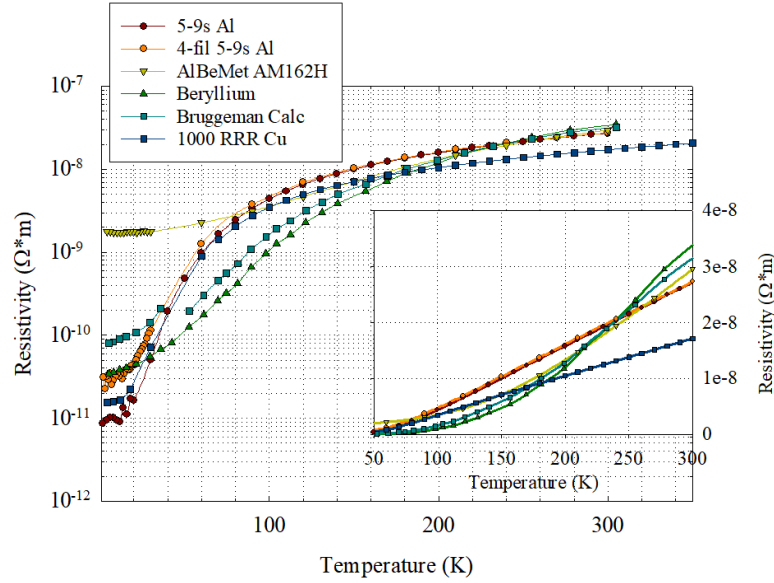


Fig. 228.—Electrical resistivity versus temperature (at zero applied magnetic field) for different high-performance electrical conductor options. Medium purity Be is superior to other materials down to low CTs but is difficult to shape and bond to. AlBeMet 162H takes advantage of a HPAL matrix to make a machinable and weldable Al-Be nanocomposite (Ref. 219).

3.3.2.2.2 Cryoresistive Cable Designs for Previous and Current CHEETA Aircraft

Direct flowing cryogen on the cryoresistive cable surface is required to effectively remove power from Joule heating. Assuming an infinite bath of flowing cryogen, one could calculate the ampacity of a cryoresistive metal by limiting the allowable temperature increase to that of the nucleate to film-boiling transition, otherwise known as the departure nucleate boiling (DNB) limit. At this transition, the HTC rapidly drops due to the formation of a large and rapidly replenishing vapor layer, and this transition therefore determines thermal runaway of many systems designed to push their limits.

Starting from a dynamic thermal power equation shown in Eq. (56) and changing to the steady-state equation, one can determine the heat transfer between the cable thermal loads and the LH₂ cryogen.

$$L_{unit}A_{x-cable}\delta C_p(T)\frac{dT}{dt} = I^2 \frac{\rho(T)L_{unit}}{A_{x-cable}} + Q_{cryostat} - h(T)(T_{cable} - T_{LH_2})L_{unit}P_{cable} \rightarrow$$

$$steady\ state, \frac{dT}{dt} = 0 \text{ when } I^2 \frac{\rho(T)}{A_{x-cable}} + Q_{cryostat} = h(T)(T_{cable} - T_{LH_2})P_{cable} \quad (56)$$

where L_{unit} is unit length (1 m), $A_{x-cable}$ is cross-sectional area of the cable (m²), δ is mass density (kg/m³), $C_{p(T)}$ is specific heat capacity as a function of temperature (J/kgK), dT/dt is change in temperature with time, I is transport current (A), $\rho(T)$ is electrical resistivity as a function of temperature, $Q_{cryostat}$ is cryostat heat loss per unit length (W), $h(T)$ is the HTC between the cable surface and liquid cryogen (W/m²K), T_{cable} is the temperature of the cable (K), T_{LH_2} is the bulk temperature of the liquid cryogen (K), and P_{cable} is the total of all of the cable's individual strand perimeters (m).

The resulting cooling per surface area in LH₂ as a function of ΔT is shown in Fig. 229. The $T_{cable} - T_{LH_2 \text{ at Standard Pressure}} (\Delta T)$ before the DNB is approximately 5 K and $h(T)$ is a function that can be described by the simplified function shown in Fig. 229 for heated plates and exteriors of heated tubes (Ref. 220). Note, heated fine wires will have different $h(T)$ that might not describe our situation well.

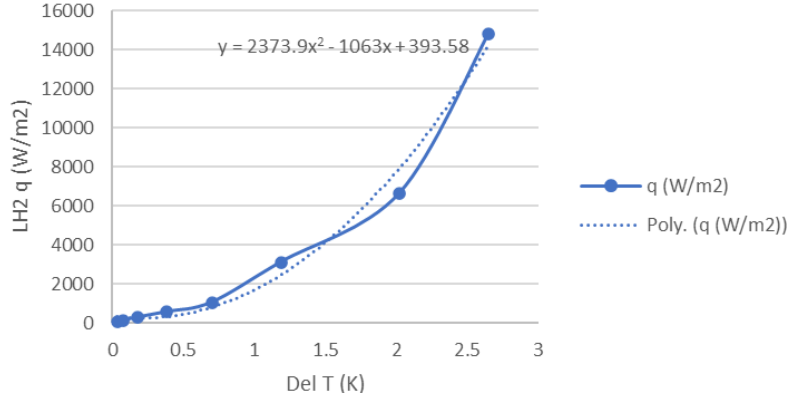


Fig. 229.—Cooling per surface area in LH₂ versus ΔT ; $h(T)$ is taken by dividing $q(T)$ function [$q(T) = 2,373.9T^2 - 1063T + 393.58$] by ΔT .

An increased surface area (i.e., stranding) of the cable greatly benefits heat transfer as well as cable flexibility. Also, subcooled fluid at the inlet of the cable is effective at increasing the ΔT window, but this heat capacity is relatively small compared to the para-ortho conversion and latent heat of vaporization of LH₂. Eventually the system is limited by the cooling power available, which is dynamically determined by the fuel consumption rate and any other thermal dampers, such as phase-change materials and the pressure limit of the cable conduit/cryostat. The pressure of the conduit is important because large channels are undesired due to increases in mass and system volume, but with cable stretches tens of meters long, the two-phase fluid pressure drop can easily exceed pressure ratings of the system components. Therefore, there exists both an optimal cable cross-sectional area (for cable cooling limit) and an optimal conduit/cryostat size (for the pressure limit).

The maximum cooling limit is determined first by an examination of the LH₂ fuel consumption rate and the exhaust from the cable or exhaust from the HEX when LH₂ is a secondary coolant. It is important to note that with the use of LH₂ as a secondary coolant and an inert/nonfuel primary coolant, the HEX effectiveness will at best be 50%, thereby reducing cooling available. The T increase of subcooled LH₂ is determined by the time a slice of flowing cryogen experiences Joule heating output (Eq. (57)):

$$dT_{\text{LH}_2} = \frac{Q}{V \times C_p} dt \rightarrow dT_{\text{LH}_2} = \frac{q \times dx}{dx \times A \times C_p} dt \rightarrow$$

$$T_{\text{inc}} = \frac{q}{A \times C_p} \times \frac{L}{v} \rightarrow T_{\text{exit}} = T_0 + T_{\text{inc}}, \text{ where } T_{\text{inc}} = \frac{q \times L}{\dot{v} C_p \left[\frac{\text{J}}{\text{m}^3 \text{K}} \right]} \text{ or } \frac{q \times L}{\dot{m} C_p \left[\frac{\text{J}}{\text{kgK}} \right]} \quad (57)$$

where T_{inc} is subcooled LH₂ temperature increase (K), $\dot{m}$ is mass flow rate (kg/s), $\dot{v}$ is volumetric flow rate (m³/s), Q is total heat generation = $I^2 \frac{\rho(T) \times L}{A_{x-\text{cable}}} + Q_{\text{cryostat}} \times L$ (W), q is heat generation per meter (W/m), V is the system volume (m³), C_p is specific heat of the cryogen (J/m³K or J/kgK), L is cable length (m), T_{exit} is the cable exhaust temperature (K), and T_0 is the initial LH₂ temperature (K).

If the subcooled LH₂ reaches the vaporization temperature, the Joule heating will go toward boiling LH₂ into H₂. Assuming the para-ortho conversion is kinetically limited (no catalyst), this heating limit is limited by the maximum allowable volumetric vapor quality χ of the cable conduit or HEX before slug-plug flow or excess two-phase pressure drop. Because vapor density (and therefore volumetric vapor quality) is also controlled by the system pressure, the system pressure for calculations was chosen as the maximum working pressure of the conduit, a standard Nexans flexible cryostat, and a lightweight fluid-fluid shell-tube HEX (~200 psi). In the case of LH₂ as the primary coolant, the pump is assumed to be part of the mass of the fuel system. In the case of LH₂ as the secondary coolant, for future work, the circulation pump mass must be considered during this analysis because of the discharge pressure and flow

rates involved. The volumetric vapor quality limit was arbitrary chosen as 0.3 (30%), and the conduit and cryostat sizes and maximum pressure drop were chosen around this.

A pressure of 200 psi is nearly the critical pressure of H₂ in the P-T diagram, and if this pressure existed throughout the cable length, there would be no vapor other than superheated LH₂. It must be noted that this pressure exists at the inlet of the conduit, and at the end of the conduit the pressure is considered to be 1 atm. Therefore, any heat put into the system in the superheated liquid length will be extracted as vapor once the pressure decreases, and an equivalent calculation is assuming the Joule heating all goes toward vaporization at 1 atm. Again, when investigating a slice of moving fluid, the vapor quality is determined by the following equation:

$$\begin{aligned} d\chi_{\text{LH}_2} &= \frac{Q}{V \times L_v \times \rho_{\text{gas}}} dt \rightarrow d\chi_{\text{LH}_2} = \frac{q \times dx}{dx \times A \times L_v \times \rho_{\text{gas}}} dt \rightarrow \\ \chi_{\text{LH}_2} &= \frac{q}{A \times L_v \times \rho_{\text{gas}}} \times \frac{L}{v} \rightarrow \chi_{\text{LH}_2} = \frac{q \times L}{\dot{v} (L_v [J/kg] \times \rho_{\text{gas}})} \end{aligned} \quad (58)$$

where L_v is the latent heat of vaporization of LH₂ (J/kg) and ρ_{gas} is the mass density of the gas (kg/m³).

These equations were configured and solved for different cable lengths (10, 20, and 30 m), different electrical power ratings, and fuel flow rates, and cable diameters were optimized to reach $\chi = 0.3$. The results shown in this report assumed atmospheric pressure LH₂ on the inlet that was not subcooled and initially had no vapor fraction. However, this method can also be used to numerically solve for other scenarios using the analytical equations shown for subcooled LH₂, which is intended for future work. Likewise, this method can also be used to consider strategies that utilize superheated LH₂ created with the application of a net system pressure with an equal reduction in allowable pressure drop along the cable length.

Once this cable cross-sectional area was calculated, a compatible conduit size was selected to ensure that the associated pressure drop was under 200 psi. The pressure drop of two-phase flow was determined using the Lockhart-Martinelli equation, assuming the vapor fraction 0.3 existed along the entire length of the cable, adding a fairly large safety factor since the vapor fraction would increase along the length of the cable. The difference between single-phase and two-phase calculations was investigated and is shown in this report. To calculate the Reynolds number, it was assumed the annular area around the cable created an effective diameter open pipe.

$$\begin{aligned} Re_{\text{vapor}} &= \frac{v D_{\text{conduit}} m_{\%-\text{vapor}} \delta_{\text{vapor}}}{\mu_{\text{vapor}}}, Re_{\text{fluid}} = \frac{v D_{\text{conduit}} (1 - m_{\%-\text{vapor}}) \delta_{\text{LH}_2}}{\mu_{\text{LH}_2}} \\ \text{and } f_{\text{vapor}} &= \frac{0.046}{Re_{\text{vapor}}^{0.2}}, f_{\text{LNG}} = \frac{0.046}{Re_{\text{LH}_2}^{0.2}} \end{aligned} \quad (59)$$

where D_{conduit} is the dimension of the cable fluid conduit taken as the diameter if the annular fluid area were to form a circle of equal area; μ_{vapor} and μ_{LH_2} are the dynamic viscosities of the H₂ vapor and LH₂, respectively. In nearly all our cases, the Reynolds number is greater than 2,000 and thus turbulent, which allows us to calculate the friction factor f_{vapor} and f_{LH_2} as shown in Eq. (59). Next, the frictional pressure drops for each individual phase are calculated.

$$\begin{aligned} \frac{dp}{dL_{\text{vapor}}} &= \frac{2 f_{\text{vapor}} v^2 m_{\%-\text{vapor}}^2 \delta_{\text{vapor}}}{D_{\text{conduit}}}, \frac{dp}{dL_{\text{LH}_2}} = \frac{2 f_{\text{LH}_2} v^2 (1 - m_{\%-\text{vapor}})^2 \delta_{\text{LH}_2}}{D_{\text{conduit}}} \\ \text{and } X_{L-M} &= \sqrt{\frac{\frac{dp}{dL_{\text{LH}_2}}}{\frac{dp}{dL_{\text{vapor}}}}} \end{aligned} \quad (60)$$

where X_{L-M} is the Lockhart-Martinelli parameter. Next, an LH_2 and vapor multiplier K_{LH_2} and K_{vapor} , respectively, are calculated as shown in Eq. (61) for $\text{Re} > 1500$. The final pressure drop of the two-phase fluid through the pipe is then determined by the larger pressure drop multiplied by the respective multiplier.

$$K_{\text{vapor}} = 1 + 20X_{L-M} + X_{L-M}^2, K_{\text{LH}_2} = 1 + \frac{20}{X_{L-M}} + \frac{1}{X_{L-M}^2}$$

$$\boxed{\frac{dp}{dL_{2\text{-phase}}} = K_{\text{vapor or LH}_2} \frac{dp}{dL_{\text{vapor or LH}_2}}} \quad (61)$$

After solving the inner diameter of the conduit of a single-pole direct fluid cooled cable, the cable cryostat (CC) and total mass can be determined by picking a suitable arrangement. The arrangement was chosen to be a two-pole per cryostat, with the poles in a side-by-side configuration. The conduit chosen was cross-linked polyethylene (XLPE)-Al-XLPE composite tube, with the Al layer serving as the ground shield. XLPE has a proven track record of being an excellent insulation for cryogenic cable applications (Ref. 221). An image of one of the conduits and the layout of the entire two-pole cryoresistive cable is shown in Fig. 230. If LH_2 is used as a secondary coolant, the fluid flow will run in parallel down both conduits and return to the HEX and circulation system anti-parallel outside of the conduits (still within the inner area of the cryostat). If LH_2 is used as the primary coolant, the conduits will carry the fluid flow parallel down both conduits toward the other thermal management systems (TMSs). In all cases, the flow is parallel in both conduits to increase the cable current carrying capacity, so one pole is not receiving exhausted fluid flow from the initial pole.

Previous HPAL conductors utilized an Al-alloy sheath that functioned to strengthen the otherwise extremely soft conductor. The Al-alloy sheath was unique in that it was strong, but also the alloying elements would not contaminate the HPAL during low-temperature annealing to remove atomic defects introduced from previous deformation processing (thereby boosting RRR dramatically) (Ref. 222). However, the Al-alloy possesses a very low RRR, and therefore the RRR of the complete composite generally falls near 500 (Ref. 223).

The mass of the cryostat can be determined by finding the required inner diameter and utilizing size and weight data from Nexans (Paris, France), shown in Table 32, with an empirical sizing determined by regression, shown in Fig. 231. It is important to note it is possible to lightweight Nexans cryostats by removing the armoring (~50% reduction) and changing from stainless steel to titanium or other materials. Shown are some examples of cable system masses for different power 270 VDC cables, fuel consumption rates, and cable lengths with the cooling limit and pressure limit taken into account. These cable system mass values are compared with the best data of COTS Cuponal™ busbars (insulation mass negligible) (Ref. 224).

When considering two-phase flow, the cryoresistive cable systems did become heavier. This is due to the enlarged cryostat space required to fit the increased gap between the cable surface and the inner wall of the conduit. As expected, increases in system mass were particularly large for higher flow rates and longer cables. When considering a metric for performance of a lightweight system, a commonly used unit is kg/kA/m. This metric considers both the mass of the system and the current carrying capacity, and smaller is better. Fig. 232 to Fig. 235 show the kg/kA/m for various cable designs. From this comparison, it can be seen that the mass metric demonstrates the greatest sensitivity to variations in flow rates and cable lengths.

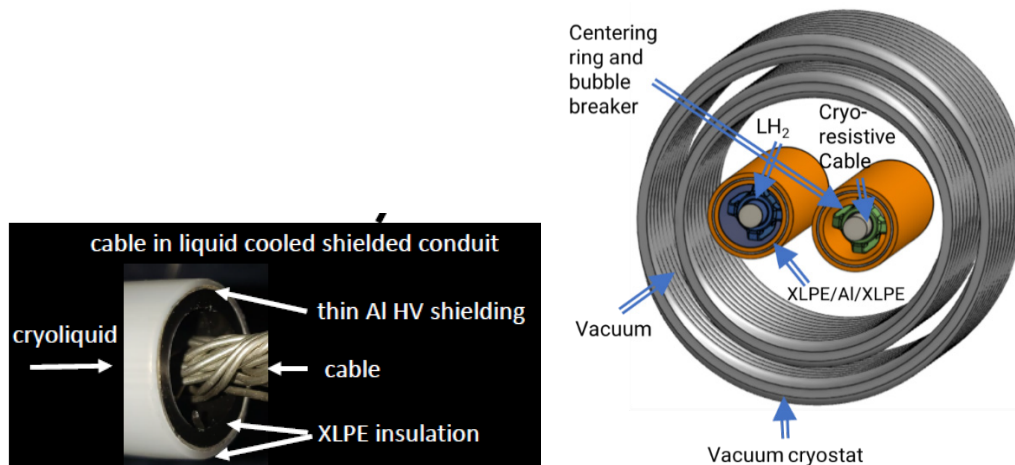


Fig. 230.—Example image of cryoresistive cable in XLPE-AI-XLPE conduit and cryostat with centering rings so electric fields are optimally located in center of conduit.

TABLE 32.—TECHNICAL SIZE AND WEIGHT DATA FOR NEXANS CRYOSTAT

Transfer line size	8/25	10/32	12/32	16/40	16/50	20/50	25/65	32/65	40/80	50/80	50/100
Inner diameter of corrugated tube, mm	8.2	10.1	12.0	16.2	16.2	20.2	25.5	34.2	40.0	50.1	50.1
Outer diameter of outer tube, including single armoring, mm	34.2	43.0	43.0	52.1	62.8	62.8	81.4	81.4	98.2	98.2	120.0
Weight, kg/m	1.5	2.0	2.2	2.5	2.7	3.0	4.0	4.5	7.0	7.5	8.5

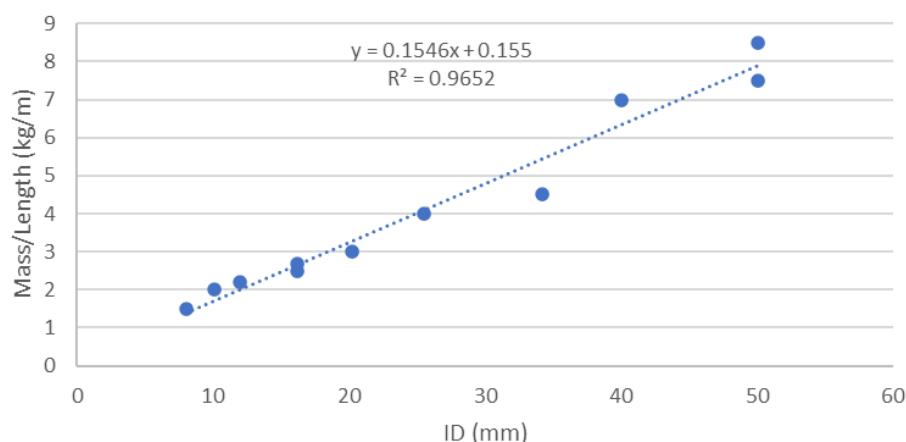


Fig. 231.—Mass per length (kg/m) versus ID (mm) of flexible cryostats from Nexans.

Not included in this system mass is any HEX or circulation pump required for LH₂ as a secondary fluid. The fluid/fluid HEX would be substantially lighter than the cable, but this may not hold true for a high-performance cryogenic circulation pump. Also not included in this system mass is the weight of the terminations, which will have to be tapered exiting the cryogenic environment. Terminations and branch T terminals will be further discussed later in this document. Also shown in Fig. 232 and Fig. 233 are the current densities of the cables, some of which are greater than 100 A/mm². As a proof of concept, a 3-m-long demonstration was created to demonstrate the ability of high surface area stranded cryoresistive cables to carry large current densities in flowing cryogen.

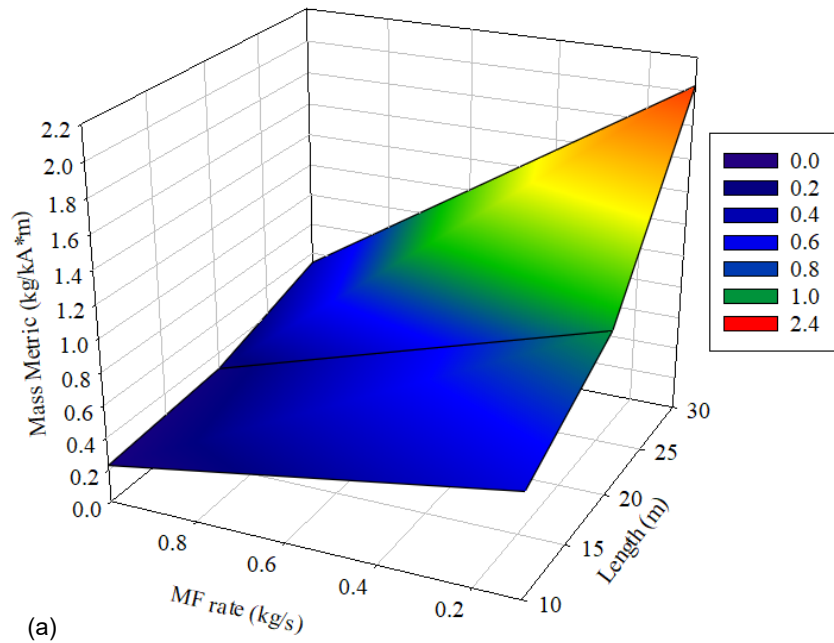
Cable Length 30 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	41.20	0.18	177.52	27.60	28.80	1691.88	9475.16	0.18	5.21
27777 A, 1 kg/s	13.03	3.00	76.12	11.92	2.88	444.10	9475.16	0.05	52.08
83333 A, 0.1 kg/s	123.60	0.06	506.64	78.48	259.17	10129.47	68763.69	0.15	1.74
83333 A, 1 kg/s	39.04	1.10	172.56	26.83	25.86	1580.67	68763.69	0.02	17.40
Cable Length 20 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	33.66	0.19	147.40	22.94	19.22	843.28	9475.16	0.09	7.80
27777 A, 1 kg/s	10.64	3.03	66.68	10.46	1.92	247.69	9475.16	0.03	78.10
83333 A, 0.1 kg/s	100.77	0.06	415.32	64.36	172.27	4732.64	68763.69	0.07	2.61
83333 A, 1 kg/s	31.90	1.14	144.16	22.44	17.26	794.11	68763.69	0.01	26.07
Cable Length 10 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	23.79	0.20	107.96	16.85	9.60	528.94	9475.16	0.06	15.62
27777 A, 1 kg/s	7.52	3.07	54.36	8.56	0.96	190.37	9475.16	0.02	156.35
83333 A, 0.1 kg/s	71.25	0.07	297.28	46.11	86.12	2644.73	68763.69	0.04	5.23
83333 A, 1 kg/s	22.57	1.20	107.08	16.71	8.64	507.03	68763.69	0.01	52.07

Fig. 232.—Parameters of cryoresistive cable system versus COTS Cuponal™ system, assuming single-phase flow.

Cable Length 30 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	41.20	1.45	182.60	28.38	28.80	1715.44	9475.16	0.18	5.21
27777 A, 1 kg/s	13.03	18.29	137.28	21.38	2.88	727.76	9475.16	0.08	52.08
83333 A, 0.1 kg/s	123.60	0.49	508.36	78.75	259.17	10137.45	68763.69	0.15	1.74
83333 A, 1 kg/s	39.04	9.28	205.28	31.89	25.86	1732.42	68763.69	0.03	17.40
Cable Length 20 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	33.66	1.57	152.92	23.80	19.22	860.34	9475.16	0.09	7.80
27777 A, 1 kg/s	10.64	18.23	127.48	19.86	1.92	435.68	9475.16	0.05	78.10
83333 A, 0.1 kg/s	100.77	0.53	417.20	64.65	172.27	4738.45	68763.69	0.07	2.61
83333 A, 1 kg/s	31.90	9.78	178.72	27.79	17.26	900.97	68763.69	0.01	26.07
Cable Length 10 m	30% vapor safe cable radius (mm)	Gap to conduit for < 200 Psi (mm)	ID cryostat needed (mm)	Mass Cryostat, Nexans (kg/m)	Mass Cable (kg/m)	Cryocable Total Mass (kg)	COTS (kg)	Cryo versus COTS Ratio (#)	Je (A/mm ²), high h
27777 A, 0.1 kg/s	23.79	1.79	114.32	17.83	9.60	548.60	9475.16	0.06	15.62
27777 A, 1 kg/s	7.52	17.85	113.48	17.70	0.96	373.17	9475.16	0.04	156.35
83333 A, 0.1 kg/s	71.25	0.62	299.48	46.45	86.12	2651.53	68763.69	0.04	5.23
83333 A, 1 kg/s	22.57	10.55	144.48	22.49	8.64	622.67	68763.69	0.01	52.07

Fig. 233.—Parameters of cryoresistive cable system versus COTS Cuponal™ system, assuming two-phase flow.

500 RRR_{273/20} HPAL Composite 7.5 MW Cryoresistive Cable
Mass Performance Metric (Single Phase Flow)



500 RRR_{273/20} HPAL Composite 7.5 MW Cryoresistive Cable
Mass Performance Metric (2-Phase Flow)

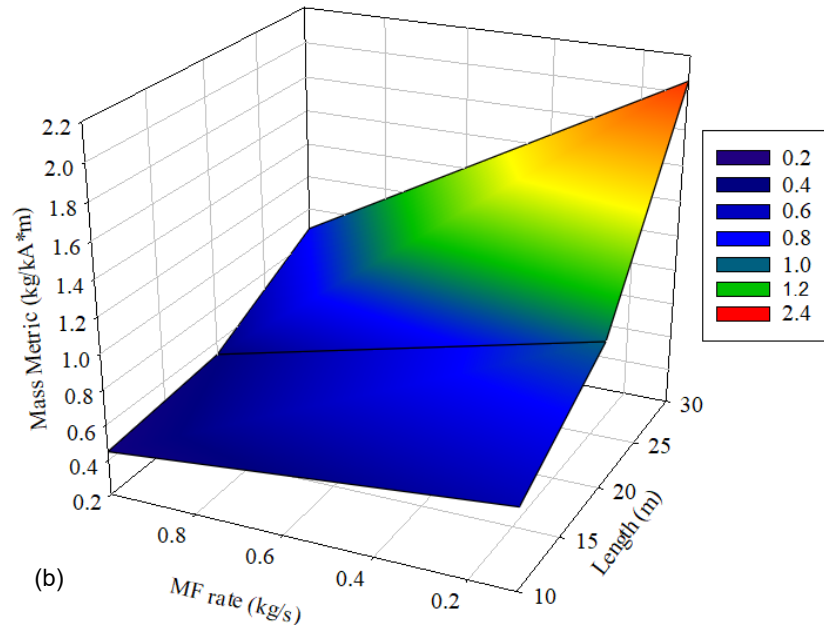
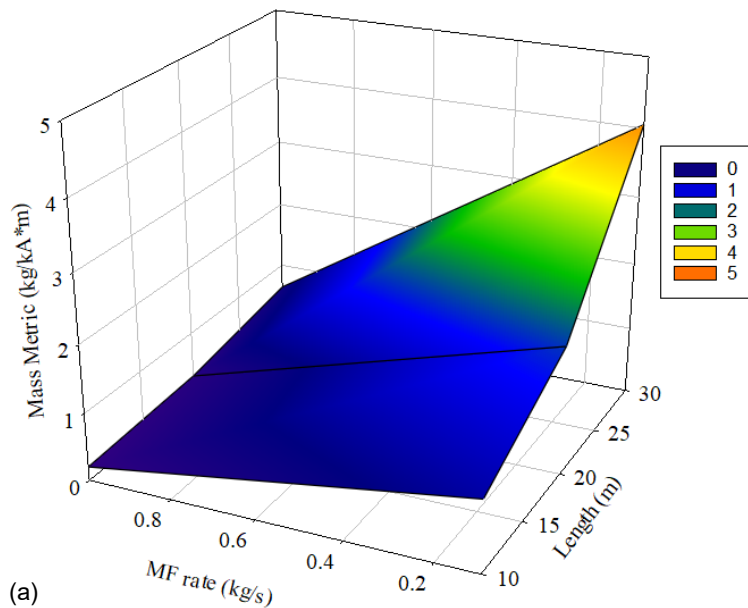


Fig. 234.—The mass performance metric of the 7.5 MW cryoresistive cables systems considering (a) single-phase flow and (b) two-phase flow.

500 RRR_{273/20} HPAL Composite 22.5 MW Cryoresistive Cable
Mass Performance Metric (Single Phase Flow)



500 RRR_{273/20} HPAL Composite 22.5 MW Cryoresistive Cable
Mass Performance Metric (2-Phase Flow)

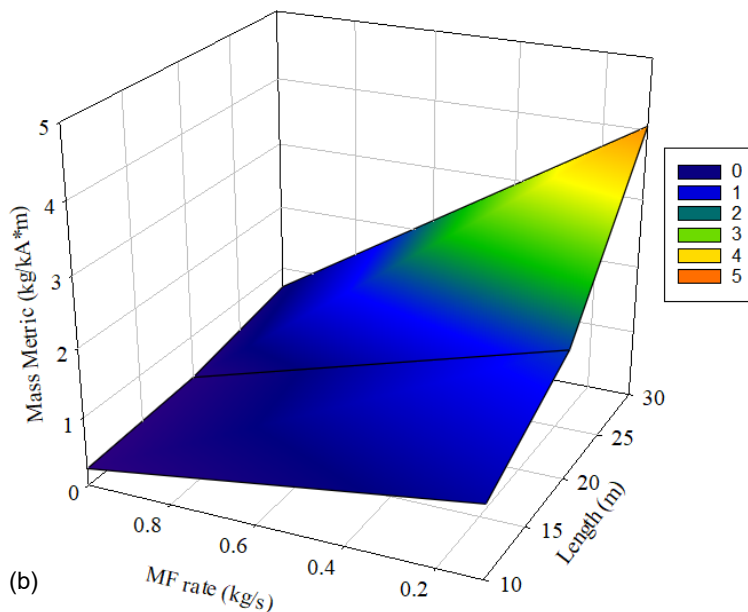


Fig. 235.—The mass performance metric of the 22.5 MW cryoresistive cables systems considering (a) single-phase flow and (b) two-phase flow.

The use of centering rings for the cables in the conduits was to limit fretting of the cryoresistive cable and reduce the risk of dielectric breakdown of the insulating inner XLPE wall of the conduit. Because the centering rings will have a slight difference in electric permittivity from the fluid, the possibility of electric field amplification exists, and this must remain well below the breakdown strength of the XLPE. The use of centering rings in cryoresistive cables, and the important role of their geometry on electric

fields, is well recognized (Refs. 225 and 226). For the design chosen, electric field simulations were performed (Fig. 236) for cast moldable Stycast 1266 with particulate reinforcing silica of differing solid loading percent (permittivity calculated with a rule of mixtures (Refs. 227 and 228)). The solid loading makes epoxy significantly stronger and allows permittivity matching with the LH₂ ($\epsilon_r = 1.2$) and nearby XLPE ($\epsilon_r = 2.3$).

3.3.2.3 Superconducting Cable and Thermal Management

DC SC cables are a clear winner when in steady operation. State-of-the-art high-temperature SC (HTS) cables can carry more than 1,000 A/mm² at 20 K without direct fluid cooling because there is no Joule heating. During unsteady operation, due to the reduced metal stabilizer fraction and large discontinuity in high current carrying capacity, HTS cables may be at higher risk of failure and total inoperability (unacceptable for aerospace).

3.3.2.3.1 Benefits of Superconducting Cables

Superconducting composite cables hold great promise in lightweighting high-power aerospace cables. Already, SC composite cables have been deployed as lighter, more compact, and even cheaper naval degaussing systems (Ref. 229). There are multiple HTS composites, cables, and thermal management architectures. Of the most mature HTS composites purchasable in large orders, there are four options: Bi-2223, Bi-2212, rare-earth barium copper oxide (REBCO), and MgB₂. For DC aerospace power transmission, currently the most rugged option of all HTS composites is REBCO. REBCO-coated conductors and cables made from them are able to be bent to much larger strains than reacted MgB₂ wire, Bi-2212 wire, and Bi-2223 tape. While bismuth strontium calcium copper oxide (BSCCO) superconductors are currently able to keep up with YBCO in terms of engineering current density, if the scalability of the REBCO tape thickness and double-sided substrates successfully advances in long lengths (Ref. 230), REBCO will be a clear best option.

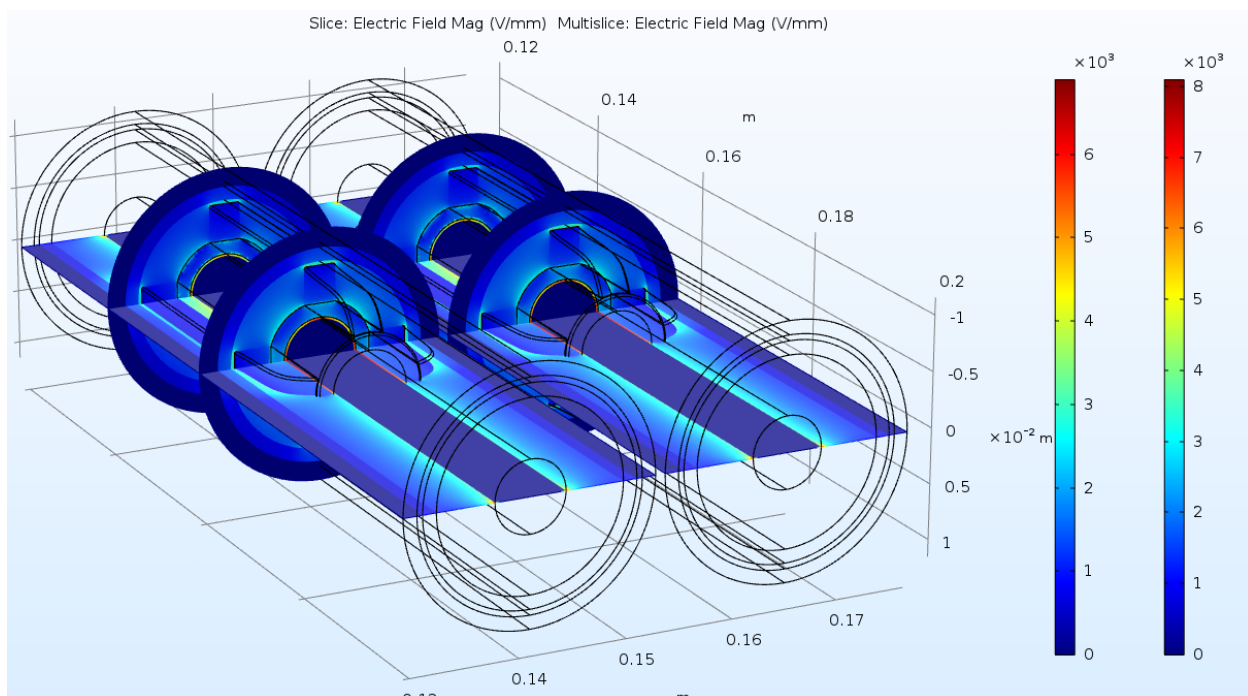


Fig. 236.—Electric field simulation (cable at 10 kV) of cryoresistive cable, centering rings, and XLPE-AI-XLPE conduits. There exists some electric field focusing near cable in the holding positions of the centering rings, but this is lower than the expected flash over limit.

Recent advances have made MgB_2 and Bi-2212 wires more flexible and lightweight by fabricating them into very fine monofilament or few-stranded multifilament wires with a lightweight stabilizer (Refs. 231 to 233). The thermal stability of these fine wires, and cables made from them, will have to be carefully analyzed because of dynamics associated with intrastrand and interstrand current sharing. However, because HTS systems are more stable from near-instantaneous transients (adiabatic) (Ref. 234), a highly increased surface area of ultrafine stable filaments in a nonadiabatic process should greatly decrease recovery time from fault conditions. A large benefit of non-REBCO HTSs is that there may be a severe supply shortage of REBCO HTSs in the coming years (Ref. 235). This is due to a huge spike in demand, which supply will take possibly a decade to match.

To address this shortage, additional studies were conducted in collaboration with Advanced Conductor Technologies to understand the potential for a He-gas cooled, cold dielectric, REBCO CORC[®] cable in SC aviation power transmission applications.

3.3.2.3.2 Superconducting Cable for the CHEETA Aircraft

He-gas cooling (20 K) is superior to LN_2 and other inert cryofluids for three reasons: (1) lower pressure drop and existing circulation pumps for cryogenic He-gas, (2) enhanced power density due to the 6 to 8 times multiplier in J_e , from 63 to 77 K down to 20 K, and (3) if a leak does occur, there is no rapid expansion of liquid cryogen, which can pose an asphyxiation hazard. There are disadvantages to He-gas versus cryofluid cooling: (1) slower return to operation temperature from fault condition, (2) reduced dielectric breakdown voltage of He-gas, (3) reduced cooling on feedthrough terminations, and (4) thermal communication required through dielectric layers (since the He-gas will not function well as a dielectric) (Ref. 236). Conventionally, it is inadvisable to rely solely on the dielectric strength of the cryogenic fluid. While it is true a constantly replenished fluid dielectric can reduce partial discharge and treeing during long term operation, there is a greatly reduced dielectric strength from bubble formation. Many challenges of He-gas cooled cables are being solved through the ongoing U.S. Department of Energy (DOE) Advanced Research Projects Agency – Energy (ARPA-E) Connecting Aviation By Lighter Electrical Systems (CABLES) program (Ref. 237).



Fig. 237.—A turnkey He-gas-cooled CORC[®] cable with CT-to-RT terminations. The terminations are currently a large portion of the system mass and will require weight reduction for aerospace applications (Ref. 237).

While it is true that the REBCO cable itself will have a very low mass performance metric (<0.1 kg/kA/m) and a low cross section, similar to the cryoresistive cable, the cryostat can make up a large portion of the system mass. A direct LN₂-cooled design has experimentally reached 1.44 kg/kA/m for a 1-MW cable utilizing Al corrugated tubing as the cryostat, but hopes are to reduce this to less than 0.4 kg/kA/m by further lightweighting the cryostat (Ref. 238). A He-gas-cooled cable for this program will be substantially lower mass than this due to the lower pressure rating required for the cryostat, the higher J_e at 20 K, the lower insulation mass due to only ± 270 VDC, and the lack of a dense fluid in the conduit.

3.3.2.4 Terminals and Terminations

The EWIS will require branch terminals from main cables and terminations from different thermal gradients and cable environments. Historically, the design of terminations is significantly more complex than that of the cable (Fig. 237). For ground installations, terminations design can be simplified because mass is less of an issue, but for a vibrating aerospace environment, including system mass in the optimization procedures adds significant complexity. As part of this program, Scintillating Solutions LLC has made preliminary designs of terminations and branch terminals.

3.3.2.4.1 Thermal Gradients, Heat Leak, and Joule Heating

A method to reduce the complexity and mass of terminations is to try to minimize the thermal gradient across them. This can be done by keeping many of the system links and power electronics under cryogenic conditions. (Cryogenic power electronics are discussed in Refs. 139 and 239; SC fault current limiters (FCLs) are discussed in Ref. 240.) Termination designs exist that do not physically link the terminals, and instead magnetically couple them through a cryostat transformer (Ref. 241). While these designs currently introduce more heat generation into the cryogenic system than an optimized solid tapered termination, and would introduce substantial EMI, it is worth noting their development. Our approach has been to keep with standard tapered cryoresistive cryogenic terminations and cryoresistive terminals but to optimize their design and material selection. The use of bulk Cu, in general, should be avoided because of its high mass specific electrical resistivity (see Fig. 227).

3.3.2.4.2 T-Terminal and Large Termination for Previous Iteration CHEETA Aircraft

Early in the program, initial designs were performed for a T-terminal (Fig. 238) for REBCO CORC[®] cable with a 20-kA main and 2.5-kA branch and operating at 30, 40, and 50 K. The T-terminal was tapered for current transfer, heating and current sharing within the CORC[®] cable were accounted for, and contact resistivity between the CORC[®] and cryoresistive T-terminal were also accounted for. The CORC[®] insertion into the T-terminal was tapered, as this is required for good current transfer into multilayered REBCO CORC[®] cable (Ref. 242). The conductivity of the CORC[®] cable, required for simulating current sharing within the cable if heating occurs, was described as shown in Eq. (62).

Different-sized T-terminals were investigated, but all of them utilized what is essentially Cuponal[™] (thin Cu exterior, Al core). The Cu and Al in the T-terminals was RRR = 300, and fixed-temperature boundary conditions were used. The T-terminal would also require a T-shaped cryostat, but this is unavoidable. The analysis included magnetoresistivity and mechanical analysis from different coefficients of thermal expansion (CTEs) assuming a zero-strain temperature at 293.15 K (20 °C, room/assembly temperature).

In Eq. (62), T_0 and J_{c0} are a reference temperature and the critical current density at that reference temperature, respectively; α_{Cu} is the temperature coefficient of resistivity; and $\|E\|$ and E_c are the magnitude of the electric field and an electric field criterion at which J_c is described, respectively.

It was found that magnetoresistivity made little difference to the Joule losses due to the low self-fields and domination of contact resistances. Also, it was found that maximum stresses from thermal contraction (90 MPa max) were mostly radial on the REBCO CORC® cable, but lower than the safe limit for REBCO CORC® cable (Ref. 243).

A 37-kA (10 MW with 270 VDC) two-pole terminal was designed early in this program. The cryoresistive terminal did not extend much beyond the initial design phase, but pieces were purchased for constructing it. Shown in Fig. 239 is the general structure of the gas-cooled termination and a 37-kA rated RT feedthrough that was purchased for heavy modification.

$$J_c(T) = \frac{J_{c0} \left(1 - \frac{T}{T_c}\right)^{1.65}}{\left(1 - \frac{T_0}{T_c}\right)^{1.65}}$$

$$\sigma_{Cu}(T) = \left((3 \times 10^{-9}) (1 + \alpha_{Cu}(T - T_c)) \right)^{-1} \quad (62)$$

$$\sigma_{CORC}(T) = \begin{cases} \text{if } T < T_c & \frac{J_c}{\|E\|} \left(\frac{\|E\|}{E_c} \right)^{0.8} + \sigma_{Cu} \\ \text{if } T \geq T_c & \sigma_{Cu} \end{cases}$$

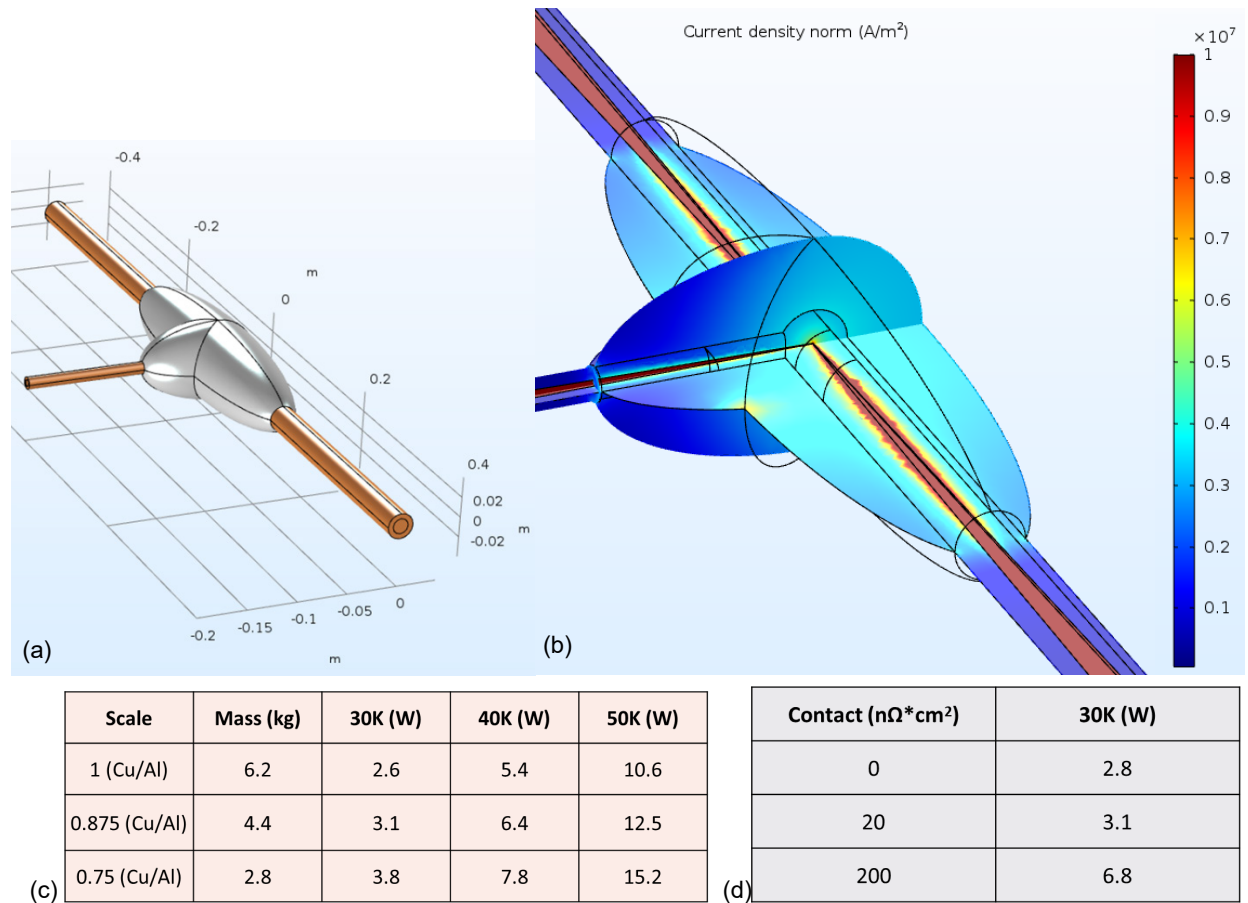


Fig. 238.—T-terminal design for CORC cables. (a) CAD of T-terminal (scale = 1) and (b) current density throughout T-terminal with 20 kA main and 2.5 kA branch. (c) All terminations possessed low Joule heating. (d) Most heating was where current left CORC® cables and traveled through contact resistances = 20 nΩ*cm² (shown for 0.875 scale at 30 K).

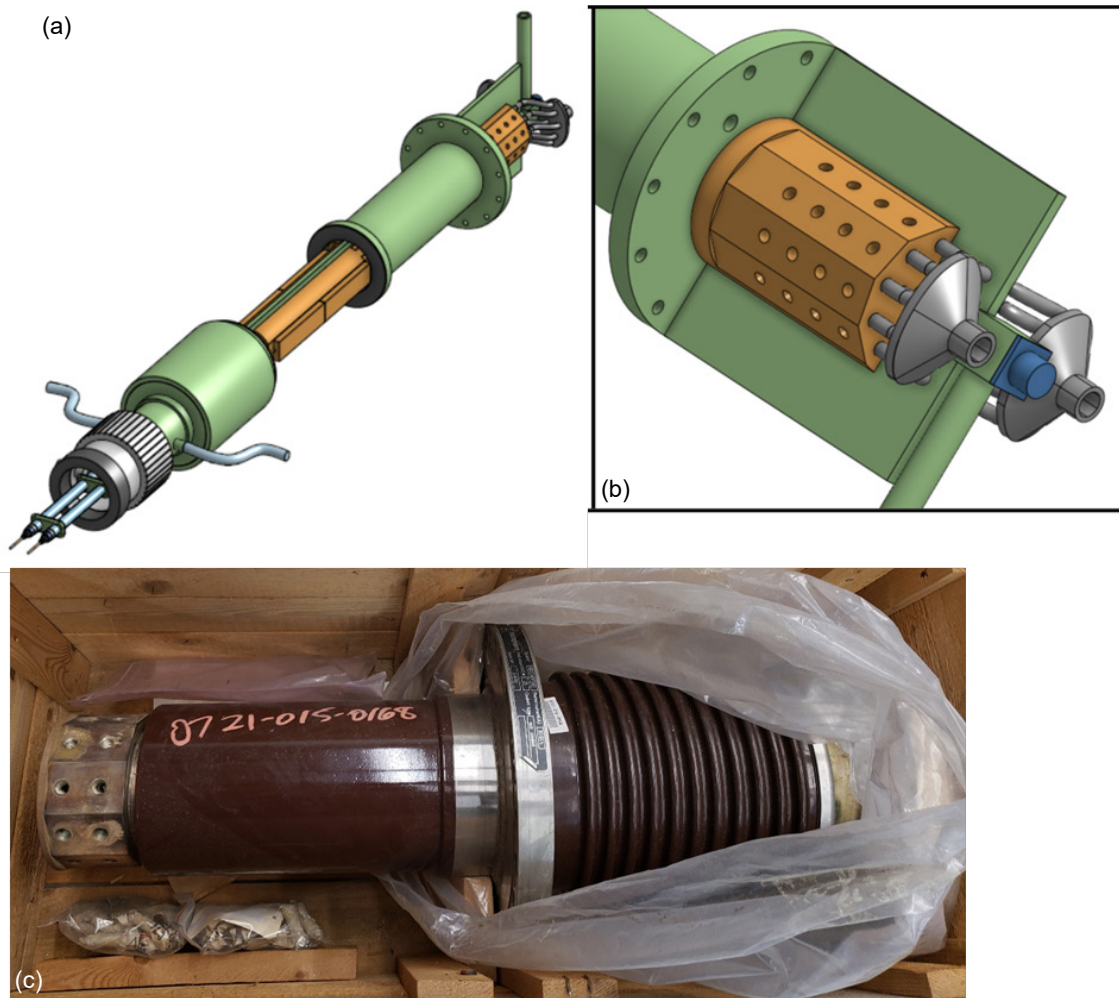


Fig. 239.—Gas-cooled termination and RT feedthrough. (a) 120-kg, 37-kA gas-cooled termination from 20 K to ambient. (b) Details of gas collection ports through termination current lug attachment points. (c) 37-kA termination purchased for purpose of scavenging current lug attachment point at near-ambient temperatures.

This early design did not consider icing and the ice point on the termination; it was simply designed to demonstrate the ability to transfer 10-MW rated power through a two-pole cryoresistive termination to a two-pole CORC[®] cable. Later design iterations replaced the unoptimized Cu termination with an optimized AlBeMet 162H termination; mass specific resistivity of AlBeMet 162H versus HPAL is shown in Fig. 240. It should be noted that Cu is significantly worse than even low-purity Al, which is why this case is not considered. It is also clear from Fig. 240 that AlBeMet 162H would function as a significant lightweighting solution for a termination to the ambient temperature.

Preliminary optimization routines in FEM simulations have been performed with tapered Al terminations as part of another project (ARPA-E cables), but changing the temperatures and material properties in the COMSOL program would be straightforward and would only require time to run the simulations. These optimization FEM simulations will be performed in future work. As part of the demonstration project, 1-kA, 1-kV current terminations were purchased from Cryomagnetics, but again, these were vapor-cooled and not designed to prevent icing. These terminations are shown in the next section.

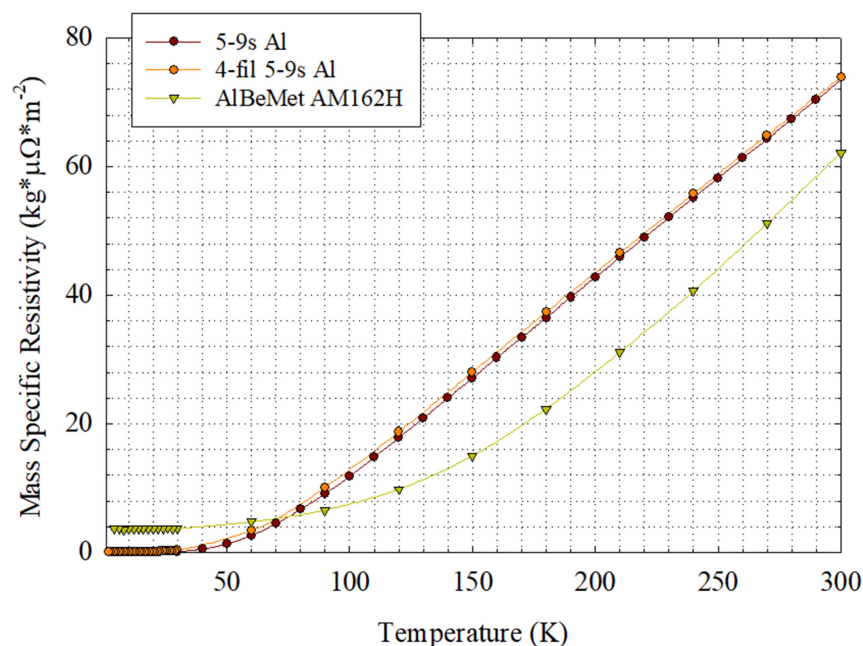


Fig. 240.—Mass specific resistivity of AlBeMet 162H and HPAL versus temperature.

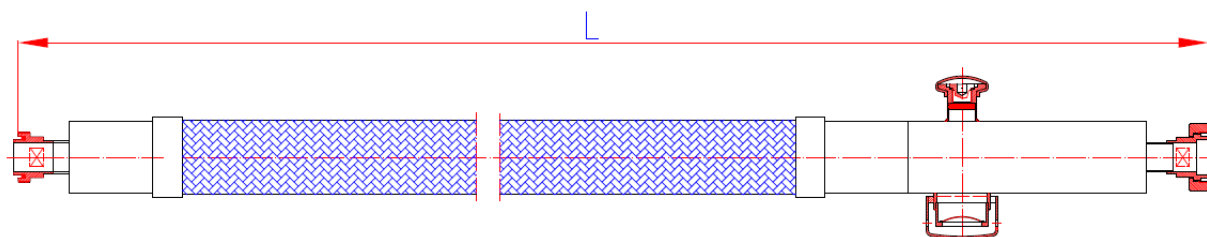


Fig. 241.—Standard Nexans vacuum-jacketed (VJ) flexible cryostat with stainless steel armoring.

3.3.2.5 Prototype Cable Demonstration System

A custom 3-m-long flexible cryostat was purchased from Nexans (Fig. 241). This cryostat has an ID of 40 mm and was terminated with KF-40 flanges on both sides. (Although KF-40 flanges were an early design choice, later design choices made the use of the KF-40, a vacuum flange, nonideal.)

3.3.2.5.1 System Layout

Early designs existed of a two-pole demonstration system, where it would be cooled with LHe and have an internal phase separator. Fig. 242 shows the schematic of an early design that would have integrated power electronics supplied by UIUC.

Due to cost constraints, an LN₂ demonstration was pursued. Because HPAL wire RRR does not take off until below 40 K, and the conductor is currently in low supply (although this is changing with recent HPAL wire research by Hyper Tech Research Inc.), the conductor chosen for commissioning the demonstration system was multifilamentary Cu-Litz cable. The use of cryoresistive cables in flowing cryogen to achieve high power transmission has been proven before in LN₂ and LHe (Refs. 244 to 246), but for the purpose of this project it needed to be reverified because of the bubble dynamics associated with ampacity and the importance of nonthermal runaway (Ref. 247).

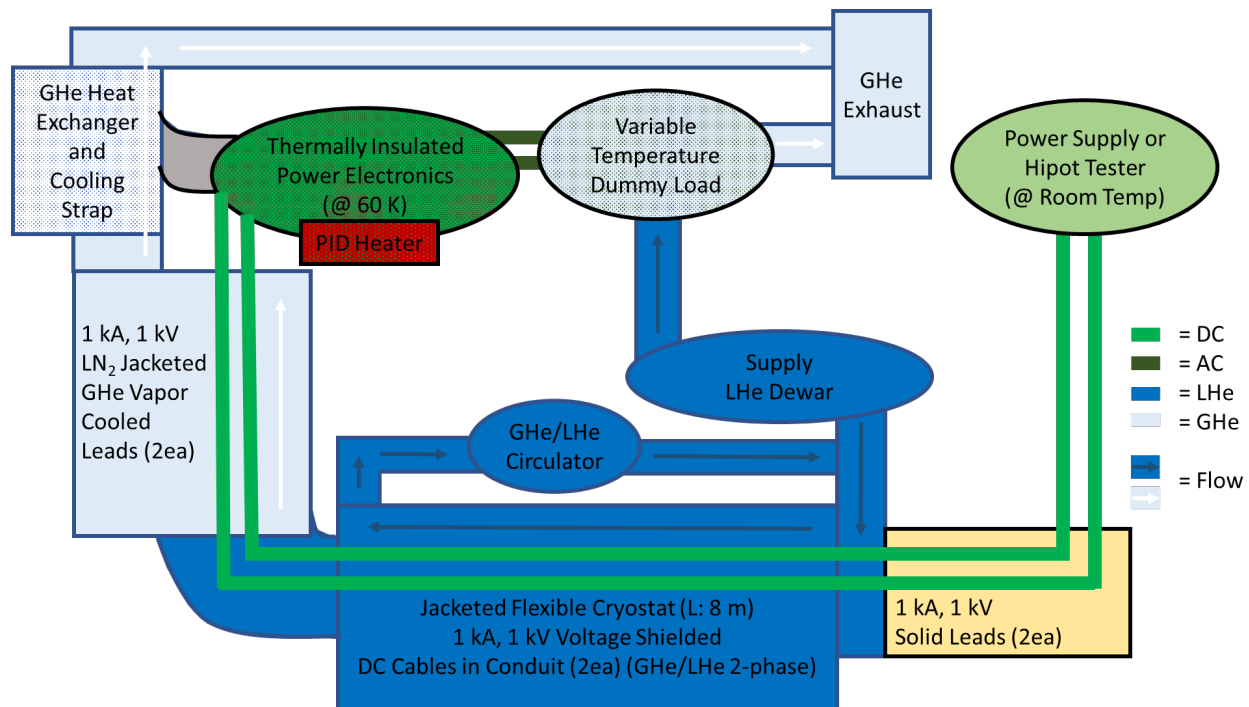


Fig. 242.—Early design iteration of LHe/GHe-cooled cryoresistive DC cable demonstration system.

A two-pole hairpin configuration was chosen for the LN₂-cooled prototype cable demonstration system commissioning. Power electronics were not integrated into this test, and the commissioning runs were performed to initially prove out the ampacity of cryoresistive cables at 77 K.

3.3.2.5.2 Commissioning and Experimental Results of Cryoresistive Cu-Litz Cable

Custom 1-kA, 1-kV terminations were purchased from Cryomagnetics (see Fig. 243). These terminations had an exhaust that could be collected, but like other designs, they were not designed to be ice-proof. Custom feedthroughs and manifolds were created in-house using vacuum flanges and chambers with cryogenically rated gaskets and epoxies. To commission the demonstration system, a two-pole hairpin Cu-Litz sample was created. The Cu-Litz cable had 810 filaments, each filament a 80- μ m Cu core with polyurethane insulation measuring a few micrometers thick. A cryogenic liquid flow meter (Sierra Instruments Innova 240S) was put in line with the LN₂ flow, but it was found that our flow was too low to be picked up by the flow meter. Therefore, a reduced-size LN₂ delivery dewar on a Mettler Toledo digital scale rated to 250 kg was used, and the delivery mass flow rate was determined by the change in mass on the scale.

The end of the demonstration system with terminations also had a window to view into the chamber (not used due to positioning), a phase separator exhaust, and a 2-psi pressure relief valve. The LN₂ in this demonstration flowed parallel down both cable conduits, starting at the hairpin, and exhausted on the termination end through the LN₂ phase separator. While it is true the conduits have been hipot-tested to greater than 50-kV breakdown strength (determined as part of the ARPA-E CABLES program), hipot testing was not performed during this set of testing because of the likelihood of the breakdown occurring at the nonoptimized exhaust manifold.

First, a solder pot was used to pre-tin two Cu-Litz cable ends into current lugs. Then the two current lugs were connected to form a hairpin sample. The free ends of the Cu-Litz cable were attached to string, which was attached to a metal beaded chain. Using a strong rare-earth magnet, the string and Cu-Litz

could be pulled down an XLPE-Al-XLPE conduit measuring 4 m long with a 6-mm ID (Fig. 244). The hairpin was placed in a larger ID XLPE-Al-XLPE conduit, and the small conduits were sealed in the larger conduit using Stycast 2850FT epoxy with Catalyst 23LV epoxy hardener after plasma-treating the XLPE bonding surfaces. The larger conduit was then sealed into a flange to fit onto one end of the Nexans cryostat (Fig. 245).

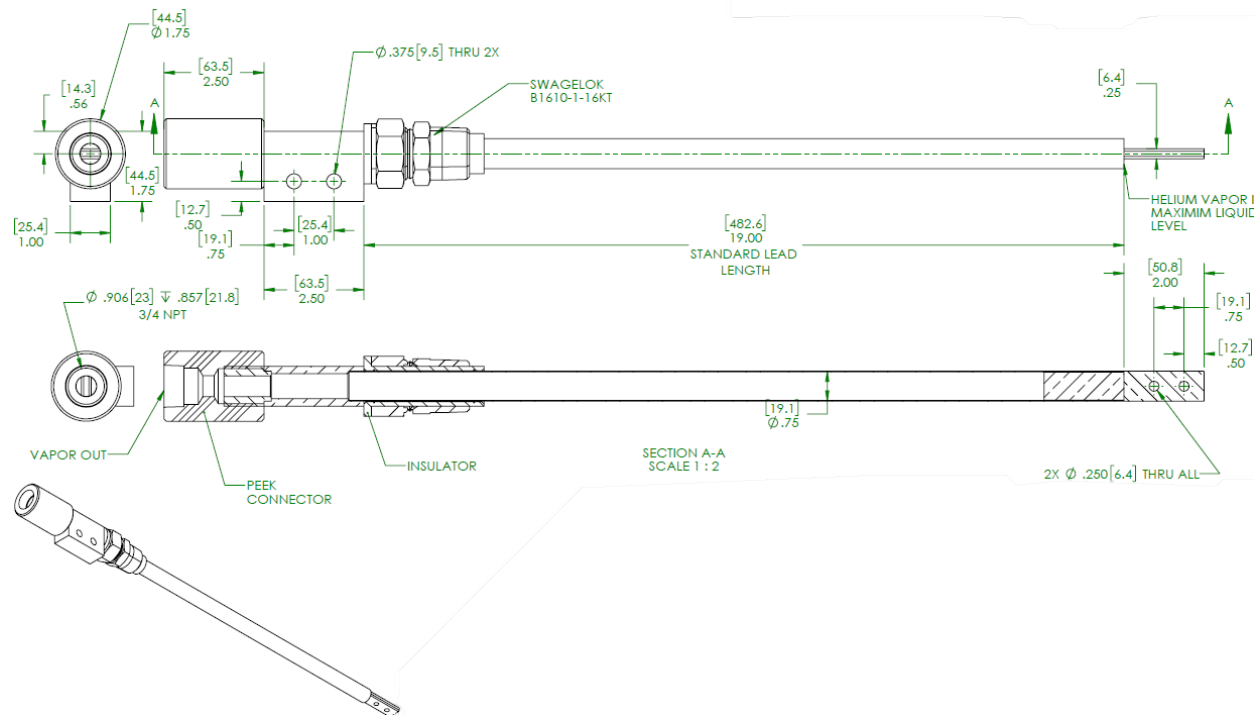


Fig. 243.—Cryomagnetics vapor-cooled leads, rated for 1-kA and 1-kV steady operation.

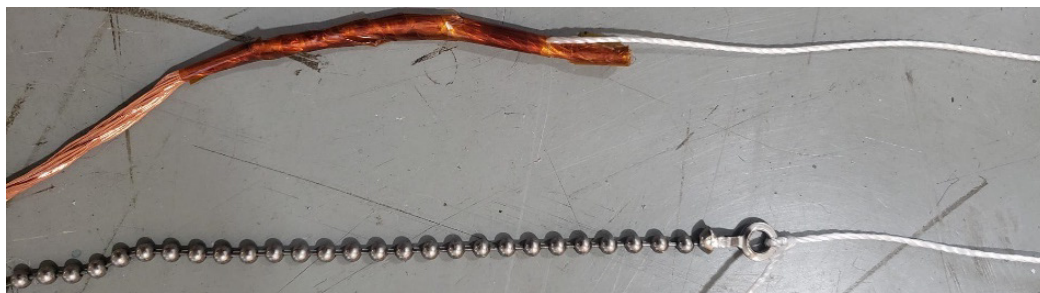


Fig. 244.—Metal beaded chain and string used to pull 5 m of Cu-Litz down 4-m-long XLPE-Al-XLPE conduits.



Fig. 245.—Assembly of the two-pole Cu-Litz hairpin.

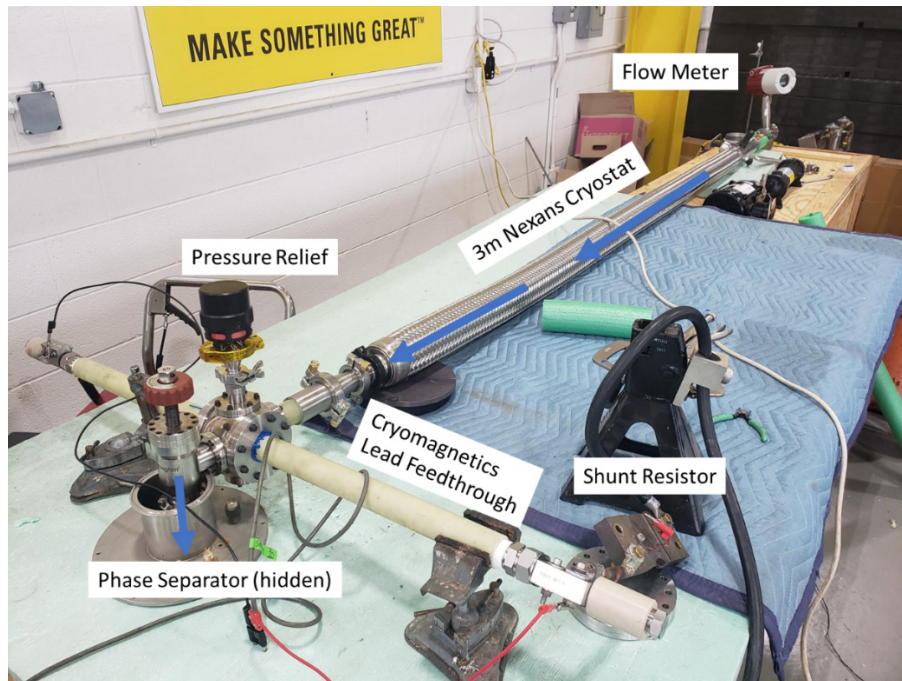


Fig. 246.—Assembled prototype cable demonstration system (image 1).

After the conduits were fed into the cryostat, the exhaust manifold, flanges, and terminations were assembled (Fig. 246, Fig. 247). Next, the Cu-Litz cables were attached to the Cryomagnetics current leads, and voltage taps were placed on the outside ambient current lugs of the current leads to monitor voltage drop and communicate with the burnout protection system (typically used for SC composites). Two HP6681A power supplies were used to supply up to 1160 A and 8 VDC into the hairpin sample. The power supplies were run in voltage control mode; this mode will self-protect the cable because once thermal runaway occurs, the current level will drop. Using voltage control mode, a pseudo-steady state can be reached that is the maximum ampacity of the cable. DAQ and controls were performed using LabVIEW.

After an initial test run with low flow rates, which did not reach peak ampacity, a final test was performed to reach peak ampacity (Fig. 248). This occurred at ~ 320 A, which correlates to a ~ 80 A/mm² ampacity in 0.025 kg/s flowing LN₂. This is similar to previous tests on Cu-Litz cable performed at AFRL as part of the U.S. DOE ARPA-E Aviation-class Synergistically Cooled Electric-motors with iNtegrated Drives (ASCEND) program. A constant voltage ramp rate was chosen, and therefore when the current

started to flatten out (i.e., precursor to excess heating and thermal runaway) is when the test was ended. Shown in Fig. 249 is the demonstration system operating during a high-amperage run. It is clear the 2-psi pressure relief valve was functioning, given that the flow impedance at the phase separator was high.

Procedures taken during the prototype cable demonstration system commissioning were largely successful for a quick proof-of-concept of cable ampacity. Future experiments will test lightweighted and low heat leak terminations, lower CTs (single-phase subcooled LN₂, LHe, He-gas), hipot and partial discharge testing, and REBCO CORC[®] cable and HPAL conductors. Future iterations of the demonstration system will focus on the utilization of technologies to create a lighter flexible VJ cryostat; Ti or Al construction without armoring are possibilities.

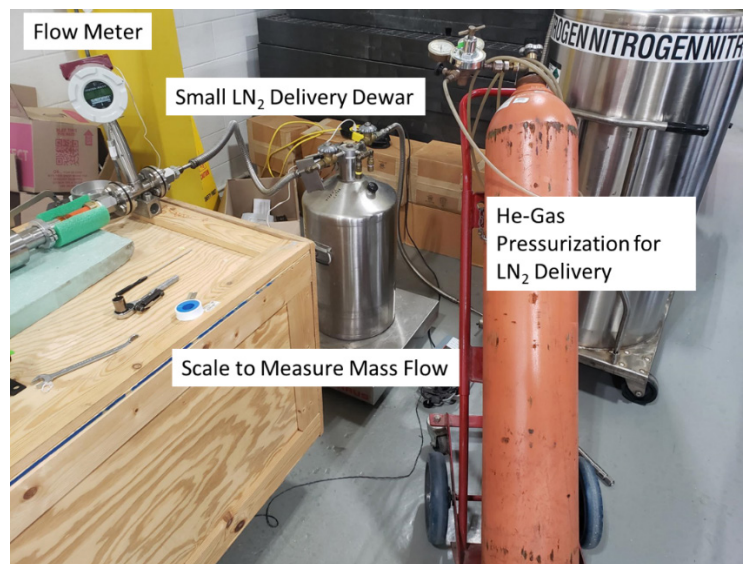


Fig. 247.—Assembled MMI demonstration system (image 2).

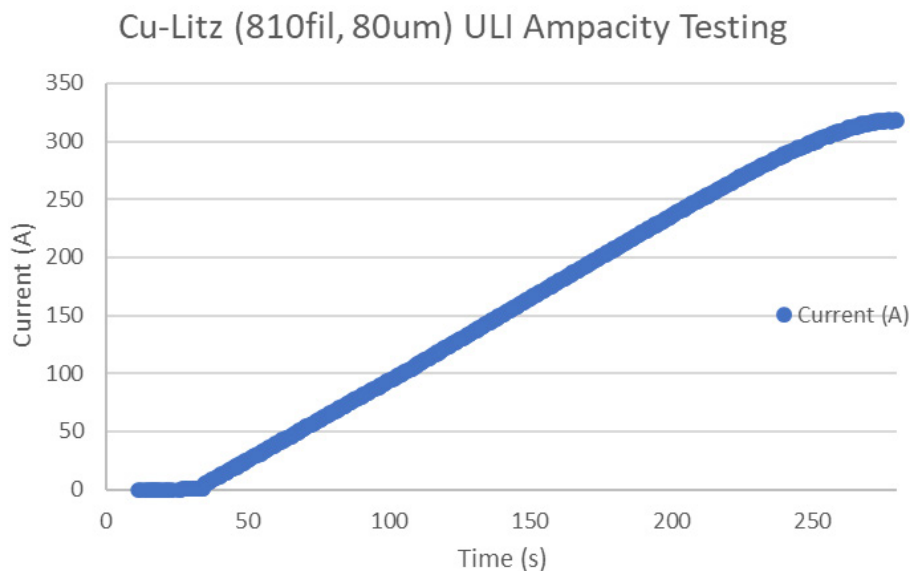


Fig. 248.—Final prototype cable demonstration system commissioning run, ampacity testing of Cu-Litz.



Fig. 249.—Prototype cable demonstration system commissioning during ampacity testing.

3.3.2.6 Cryogenic Electrical Wiring Interconnection System: Conclusions

The design of a cryogenic aerospace EWIS system was undertaken by Scintillating Solutions LLC as part of involvement in the NASA ULI program. Both cryoresistive and SC cables were investigated, and design studies were performed for cryoresistive cables. Cryoresistive branch terminals and terminations underwent design studies, and terminations will be further optimized through FEM simulations and novel material selections. A new Al-Be nanocomposite cryoresistive material was analyzed electrically (and magnetically; results not shown). The electric fields possible during fault conditions were prepared for in the design using a cold dielectric XLPE-Al-XLPE conduit around the cryoresistive cable. The Al in the conduit behaves as a ground shield, which is important: using the cryogenic Al within the cryostat as a ground shield is superior to using the inner cryostat wall because, when in operation, electrical arc damage may partially harm the ground shield.

A prototype aerospace cryogenic cable demonstration system has been designed, fabricated, and commissioned using cryoresistive Cu-Litz cable. In flowing LN₂, the Cu-Litz reached nearly 80 A/mm² ampacity. In flowing LH₂, or a suitable alternate primary coolant at lower than 77 K, this ampacity should increase due to reduced resistivity of the cryoresistive cable. Experiments to view dynamics of vapor boiling and localization during peak current density operation have yet to be successfully performed.

3.3.3 Circuit Protection Strategies

Owing to the unique high-current operating condition of superconductors and hyperconductors, protecting the electrical circuit under fault conditions can become difficult. This section provides an analysis of anticipated fault conditions as well as several strategies for design of electrical circuits and circuit protection at the MW class for cryogenic-electric aircraft configurations.

3.3.3.1 Insulation and Electric Circuit Protection

Insulation is one component in the safety design of the cables. and insulation failure occurs due to electrical, thermal, and vibrational stresses. Electrical insulation is considered a core technology for electric aircraft, with operating parameters of high voltage, high frequency from wide band-gap power convertors, and high altitude complicating the insulation design. Design parameters for insulation are dielectric strength, thermal conductivity, and mass/volume. There is a need for standards and environmental test systems to simulate high voltage, frequency, and altitude conditions.

Insulation options of aromatic polyimide (PI) (Kapton[®], DuPont Electronics, Inc.), ethylene tetrafluoroethylene (ETFE) (TEFZEL[®], The Chemours Company FC, LLC), composite (Teflon[®]-Kapton[®]-Teflon[®] (TKT), and polytetrafluoroethylene (PTFE) (Teflon[®], The Chemours Company FC, LLC), and their respective properties are ranked as most desirable to least desirable for weight, flammability, smoke generation, arc propagation resistance, and so forth, and can be found in Fig. 250. Table 33 covers more detailed electrical, material, and mechanical properties for many insulation material choices. The latest research area looks at combining the most desirable properties of several materials in a composite form. An example of this is the micro-multilayer multifunctional electrical insulation (MMEI) system, where different layers in the composite are chosen for corona resistance, moisture barrier, EMI shielding, thermal management/heat dissipation, and mechanical durability (Fig. 251).

Four types of insulation combinations are utilized for HTS cables operating at low temperatures. These include vacuum insulation and solid insulators, liquid insulation and solid insulators, tape insulation impregnated with LN₂ or LH₂, and LHe and solid insulation. The choice between these four options is determined by material compatibility, thermal expansion properties, and cost. The weight of insulation on the Nexans cable is proprietary. Calculating the weight of insulation for CORC[®] design cables requires the angle utilized and the degree of overlap when wrapping the HTS tapes around the core. Both the angle and degree of overlap determine the length of insulation used. This length and the thickness of the insulation can then be utilized to determine the weight of just the insulation, based on the density of the insulation choice. The weight of insulation is minimal compared to the weight of the other components of the electrical distribution system.

- **PI [Aromatic Polyimide (KAPTON)]** - (mil spec 81381)

- *Desirable properties:* abrasion/cut-through, low-smoke/non-flame, weight/space
- *Limitations:* arc-track resistance, flexibility

- **ETFE (TEFZEL)** - (mil spec 22759/16)

- *Desirable properties:* chemical resistance, abrasion resistance, ease of use
- *Limitations:* high temperature, cut-through, thermal rating (150.C)

- **Composite (TKT)** - (mil spec 22759/80-92)

- *Desirable properties:* high temperature rating (260.C), cut-through resistance, arc-track resistance
- *Limitations:* outer layer scuffing

- **PTFE (TEFLON)** - (mil spec 22759/12)

- *Desirable properties:* 260.C thermal rating, low-smoke/non-flame, high flexibility
- *Limitations:* Cut-through resistance, "creep" at temperature

Comparative Properties of Wire Insulation Systems				
Relative Ranking	Most desirable $\rightarrow$ Least			
	1	2	3	4
Weight	PI	ETFE	COMP	PTFE
Temperature	PTFE	COMP	PI	ETFE
Abrasion resistance	PI	ETFE	COMP	PTFE
Cut-through resistance	PI	COMP	ETFE	PTFE
Chemical resistance	PTFE	ETFE	COMP	PI
Flammability	PTFE	COMP	PI	ETFE
Smoke generation	PI	COMP	PTFE	ETFE
Flexibility	PTFE	ETFE	COMP	PI
Creep (at temperature)	PI	COMP	PTFE	ETFE
Arc propagation resistance	PTFE	ETFE	COMP	PI

FAA Aircraft Electrical Wiring Interconnect System (EWS) Best Practices

Fig. 250.—Comparison and property rankings of wire insulation options.

TABLE 33.—INSULATION PROPERTIES

Insulation material	Abbrevi- ation	VDE symbol	Dielectric constant (K) at 1 MHz	Break- down voltage (kV/mm)	Radiation resist (cJ/kg) (+rad)	Density (g/cm ³)	Flex	Strength (N/mm)	Break elong. (%)	Absorp. (%)	Vapor perm.	Weather resist	Flammability
Polyvinylchloride	PVC	Y	4	10	8·107	1.35	Good	15	250	0.4	Avg	Good	Self-extinguish
Polyvinylchloride heat resistant	PVC	Y	3.5	20	1·108	1.35	Good	18	200	0.4	Avg	Good	Self-extinguish
Polyethylene, low density	LDPE	2Y	2.3	75	1·107	0.9	Avg	10	400	0.03	Low	Good	Flammable
Polyethylene, high density	HDPE	2Y	2.3	100	7·106	0.9	Low	20	500	0.03	Low	Good	Flammable
Polystyrene	PS	3Y	2.5	30	5·106	1.06	Avg	55	30	1	Low	Good	Flammable
Polyamide	PA	4Y	4	25	1·107	1.1	Avg	50	20	2	Avg	Good	Self-extinguish
Polytetrafluoroethylene	PTFE	5Y	2.1	25	1·105	2.17	Low	40	350	0.01	Very low	Very good	Nonflammable
Fluorene-ethylene- propylene	FEP	6Y	2.1	20	3·105	2.15		20	250	0.01	Very low	Very good	Nonflammable
Ethylene tetrafluoroethylene	ETFE	7Y	2.6	30	1·107	1.7	Avg	45	200	0.01	Very low	Very good	Nonflammable
Polypropylene	PP	9Y	2.3	35	1·105	0.91	Avg	30	500	0.1	Low	Good	Flammable
Polyurethane	PUR	11Y	6	20	5·107	1.2	Good	30	400	5	Low	Very Good	Self-extinguish
Polyethylene-terephthalate	PET	12Y	3.3	30	1·107	1.4	Low	100	50	0.3	Low	Good	Nonflammable
Natural rubber	NR	G	4	20	8·107	1.6	Very good	100	350	0.1	Low	Good	Flammable
Styrene-butadiene-rubber	SBR	G	4	20	2·107	1.6	Very good	5	250	0.1	Low	Good	Flammable
Silicone-rubber	SIR	2G	3.2	20	2·107	1.25	Very good	5	200	1	Very low	Good	Flame-retardant
Ethylene-propylene-rubber	EPR/ EPDM	3G	3.5	20	1·108	1.45	Very good	5	200	0.2	Low	Good	Flammable
Ethylene-vinyl-acetate	EVA	4G	5	30	1·107	1.4	Very good	6	300	0.1	Avg	Good	Flammable
Polychloroprene	CR	5G	8	20	2·107	1.55	Good	15	300	1	Avg	Very Good	Self-extinguish
Cross-linked polyethylene	XLPE	2X	2.4	50	1·107	0.92	Avg	22	300	0.03	Low	Good	Flammable

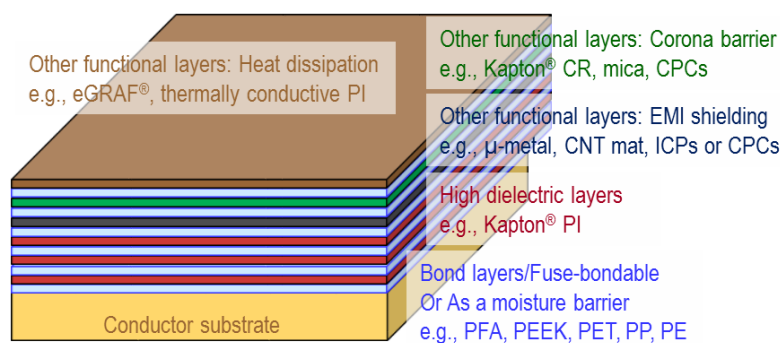


Fig. 251.—Composite insulation: Micro-multilayer multifunctional electrical insulation (MMEI) system (Ref. 248).

TABLE 34.—MEA PROTECTION SYSTEM COMPONENT PLACEMENT AND DELAY TIME, AFTER REVIEW OF BAKHSHI ET AL. (REF. 249)

Configuration	Classification	Delay, ms	Functional placement in MEA
Silicide polysilicon fuse (SPF)	Current breaker	0.5	Low power load
SS-AC fast circuit breaker	Current breaker	3	AC lines and busbars
SS-DC fast circuit breaker	Current breaker	3.7	DC lines and busbars
Electro-mechanical circuit breaker (EMCB)	Current breaker	5	Rectifiers
ABB fast circuit breaker	Limit current and breaker	5	Power distribution units
SSCB	Limit current and breaker	3	Starter generator
Resistive DC SFCL	Current limiter	10	DC low power loads
Inductive DC FCL	Current limiter	3.2	DC loads
Inductive AC FCL	Current limiter	7	Converters and motors

3.3.3.2 Electric Circuit Protection

The topic of circuit protection was investigated, which encompassed the following areas: insulation choices; oversize capability with a factor of safety; and FCLs and circuit breakers, which can be mechanical, solid-state, or a hybrid configuration. One can look at circuit protection and resiliency for more-electric aircraft (MEA) as a starting point for protection and resiliency design for all electric aircraft. Faults can occur due to component malfunction stemming from either device defects, sensor failures, faulty connections, weak communication hardware, or software malfunctions. MEA are protected with fuses, ultrafast programming of silicide polysilicon fuses, circuit breakers, solid fast breakers, solid-state AC and DC fast breakers, and FCLs (resistive, inductive, AC, and DC types). Placement of the circuit breakers and FCLs and their delay times for MEA are shown in Table 34. With regard to all electric aircraft, power system failures are inevitable and must be designed for. Short faults can result from in-flight vibration and weather. Design contingencies for resilience include insulation choices, oversize current capability with a factor of safety, and utilizing FCLs and solid-state circuit breakers (SSCBs).

Oversizing current capability with a factor of safety harnesses the capability of SC cables' critical current density increasing as operating temperature decreases. The temperature-dependent critical current and ampacity margin (I/I_c) can be used so that when a cable fails, power operations can continue, with other cables providing additional current by operating at a lower temperature. This also reduces the weight and size of the system as a whole.

FCLs are another important component in circuit protection and resiliency. FCLs reduce fault currents and prevent equipment damage and maintain power system stability. They operate by providing a series of resistances when a fault current arises to decrease the magnitude of the fault current. This allows circuit breakers to operate at the rated capacity level and protects the system until the circuit breaker interrupts. FCLs are quicker in response, as circuit breakers take 2 to 3 cycles to operate. There are two types of FCLs, superconductive (SFCLs) and nonsuperconductive. SFCLs, being superconductive, have zero resistance during normal conditions. During a fault, with the high current, the temperature rises above the critical temperature and the SFCL changes from the SC to the normal state, which is referred to as “quenching,” and increases resistance to limit the fault current. The SC state returns when the fault is removed. During the quench period, inductive shunt impedance (Z_{SH}) or parallel resistance avoids overvoltages. Resistive SFCLs have the advantage of being smaller and lighter than inductive SFCLs. An inductive SFCL is very similar to a transformer, where the secondary winding is SC. During normal operation, the resistance of the FCL is approximately zero, so its impedance results from the resistance of the primary winding. During a fault current, a quench occurs, and the corresponding resistance of the FCL increases the effective inductance in the primary line (Fig. 252). Nonsuperconductive FCLs’ resistance is bypassed during normal operation. Examples are saturated core FCLs and solid-state FCLs (SSFCLs) (Fig. 253).

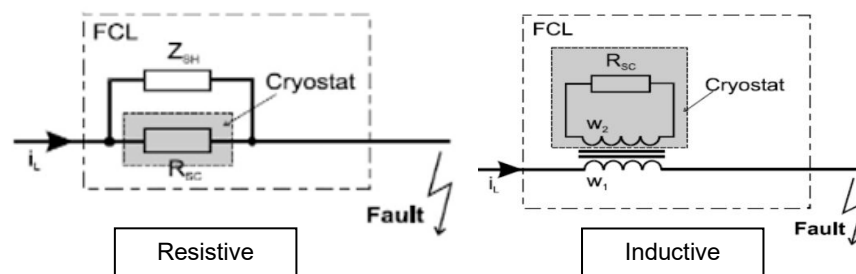


Fig. 252.—Resistive and inductive SFCLs.

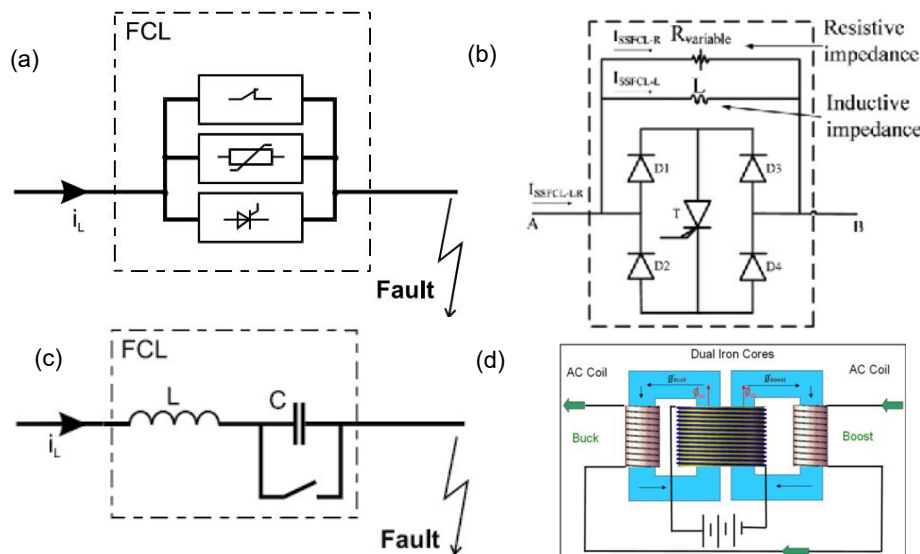


Fig. 253.—Nonsuperconductive FCL examples. (a) Serial SSFCL. (b) Bridge-type SSFCL. (c) Resonance-type SSFCL. (d) Saturated core FCL.

An interesting application is the combination of a SMES (SC magnetic energy storage) device and a SFCL. The SFCL provides effective current limitation in a few milliseconds. A SMES device has high power density, fast response time, and unlimited charge/discharge cycles. A combination of SMES-FCL designed for the NASA N3X by Alafnan et al. (Ref. 250) utilizes 65 pancake coils for the SMES device, and 2 pancake coils for the SFCL. The coils are composed of YBCO tapes with a Cu stabilizer. The SMES device has an inductance of 1.5 H, a current rating of 2 kA, and a capacity of 2MJ. The FCL has a $2\ \Omega$ resistance. When Fault 1 occurs on the propulsion side, the two pancakes act as an FCL, and the rest of the coils are isolated and protected. When Fault 2 occurs on the generation side, the current and voltage on the DC bus drops and the SMES's 65 coils work in discharge mode. Results from this architecture show the current decreases from 7 to ~2 times the rated current in a few milliseconds, and voltage is maintained at 0.86 per-unit nominal voltage versus 0.23 without the SMES-FCL.

3.3.3.3 Circuit Breakers, MW-Class

Circuit breakers are another important component in circuit protection and resiliency. The three main types are mechanical, such as the AC mechanical switch; solid-state, such as the power electronic switch; and hybrid, which combines the mechanical and solid-state switch in order to balance loss and speed. Each type has its own advantages and disadvantages, as seen in Table 35. Mechanical circuit breakers have low conduction loss, can withstand high voltage, are simple in structure and design, and cost the least; however, they arc while switching, have a long breaking time, and are hard to miniaturize. Solid state circuit breakers have high switching speeds and short response times; however, they have high conduction loss and are expensive. Hybrid circuit breakers have high breaking and overload capacity, with loss and cost in between those of mechanical and solid-state circuit breakers. All-electric aircraft require a circuit breaker that has a long life, high reliability, and arc-extinguishing capability, while minimizing volume and weight. For these reasons, a solid-state DC circuit breaker is used. Examples of DC circuit breakers developed at various research and development (R&D) institutions are listed in Table 36 with their voltage and current parameters. Recently, Fuji (Japan) has developed a three-port, nonpolar, plastic-case circuit breaker with a width of 54 mm and a rated operating voltage of 500 V, breaking fault current of 10 kA, and a response time of 5 ms (Ref. 251). In regard to the weight of circuit breakers for electric aircraft applications, Anderson et al. (Ref. 252) estimates an AC circuit breaker to be ~260 kVA/kg and a DC circuit breaker to be ~93 kW/kg. For the mass estimate for the DC circuit breaker of 93 kW/kg, one can calculate the total mass of the DC CB for the motors, invertors, fuel cells, and batteries associated with the CHEETA configuration, which amounts to 1,018 kg (Table 37).

TABLE 35.—ADVANTAGES AND DISADVANTAGES OF VARIOUS TYPES OF CIRCUIT BREAKERS

CB type	Switch type	Advantages	Disadvantages
Mechanical	Mechanical Oscillation zero crossing	Low conduction loss High withstanding voltage Simple structure and few control steps Low cost	Arc in switching process Long breaking time Hard to miniaturize Limited reliability
Solid state	Power electronic switch Turn off zero crossing	Fast switching speed and response time Can improve capacity in series or parallel	High conduction loss in normal operation High cost
Hybrid	Communication through zero	High breaking and overload capacity Cost greater than mechanical, but less than solid state	Losses are between mechanical and solid state

Prior research has also included the combination of a mechanical circuit breaker and FCL, which is referred to as a SC momentary circuit interrupter (SMCI). The SMCI provided fault protection with ultralow power loss $<1\text{ W}$ with an ultrafast response time of $<10\text{ }\mu\text{s}$. The current state of the art of hybrid circuit breakers (HCBs) has the mechanical switch/breaker, commutation path, and arrester in parallel design. In contrast to HCBs and SSCBs, the SMCI and isolation switch employs a series design. The SMCI injects a high transient voltage utilizing a pulse transformer and power electronics. The transformer is subject to cryogenic conditions, and the power electronics are galvanically isolated. This drives the fault current to zero in $10\text{ }\mu\text{s}$. The fault current is held at zero for 100 to $300\text{ }\mu\text{s}$ with AC ripples. The isolation switch opens to isolate the fault. Future research in this area will look at combining mechanical circuit breakers with SFCLs, using algorithms to control the circuit breaker and combining digitalization, industrialization, and 5G. More needs to be done to standardize test methods and specifications of circuit breakers. Future research will also look at compound control of various functions, such as FCL, DC power flow controller, converter with DC fault blocking ability, and controller assembly.

TABLE 36.—DC CIRCUIT BREAKERS DEVELOPED AT VARIOUS RESEARCH INSTITUTIONS, AFTER REVIEW BY HUO (REF. 251)

Year	R&D institution	Type	Rated capability
2011	ABB	Hybrid	80 kV/5 ms/7.5 kA
2014	Alstom	Hybrid	120 kV/5 ms/7.5 kA
2015	State Grid Smart Grid Research Institute	Hybrid	200 kV/3 ms/15 kA
2019	ABB	Solid state	1.5 kV/25 μs /5 kA
2020	Huazhong University of Science and Technology	Mechanical	535 kV/3 ms/25 kA
2020	Tsinghua University, Nari Company, and Global Energy Internet Research Institute	Hybrid	535 kV/3 ms/25 kA

TABLE 37.—CALCULATION OF DC CIRCUIT BREAKER MASS ASSOCIATED WITH MOTORS, INVERTERS, FUEL CELLS, CONVERTERS, AND BATTERIES OF CHEETA CONFIGURATION
[Based on power system architecture of Fig. 86]

Component	Number of components	Power, MW	Circuit breakers per component, kg	Total circuit breaker mass, kg
Motors	9	2.5	26.9	241.9
Motor inverters	9	2.5	26.9	241.9
Fuel cells	12	1.64	17.6	211.6
Battery converters	3	5	53.8	161.3
Batteries	3	5	53.8	161.3
Total				1,018.1

4.0 H₂ Power and Energy Systems Technology

A novel contribution of the CHEETA research program was the development of various H₂-electric technologies that enabled lightweight cryogenic energy storage and distribution, as well as integrated fuel cell/battery hybrid electric power system architectures that offer advanced efficiency for future aircraft. The advancement of LH₂ tank design, fuel cell system configuration, thermal management strategies, and high specific energy battery systems is discussed in this section.

4.1 H₂ Fuel System

The H₂ fuel system utilized in the CHEETA aircraft configuration represents a dramatic shift in energy storage as compared to incumbent aircraft configurations. In order for H₂ to be suitably energy dense that it can be integrated into a transport-class aircraft, it is preferable to store as a liquid. However, the 20-K boiling temperature requires the use of ultra-insulated storage tanks and consideration of challenges associated with pump and H₂ transfer systems, as well as additional considerations for aircraft integration.

4.1.1 Lightweight H₂ Tank Design

The design of cryogenic LH₂ tanks represents a key enabling technology for future H₂-electric aircraft configurations. While LH₂ tanks exist for a range of transportation and stationary power systems, these tanks must be significantly lighter than those utilized in ground applications. For this reason, the use of modern materials and manufacturing strategies will allow for a new class of lightweight H₂ tanks to be developed.

As part of the CHEETA program, research was conducted on the H₂ flow path schematically depicted in Fig. 254.

Fig. 254 shows the principal flow regime for the component cooling approach and its serial as well as parallel cooling arteries with its supply branches. Following the blue arrows in the figure, this simplified flowchart shows the liquid fuel distribution across the aircraft, from the storage tank to the propulsors. Most of the liquid is used for cooling the fully SC motor, the high-temperature superconductor (HTS) lines, busbars, and current leads. The SC engine (motor) with its field winding and armature needs to be maintained SC at an optimum temperature. This is achieved by making use of the high heat-sinking capability of H₂ using forced flow boiling through the helical motor windings to fully exploit the latent heat of vaporization. Superconducting power transmission lines must be maintained below their critical temperature as well. LH₂ also resides in cryogenic transfer lines and is directed to flow over cold plates for heat-sinking cryogenic power electronics and for cooling batteries.

From the motor, we drive the liquid flow to the cryogenic inverter and various other cryogenic power electronic components. Those are efficiently heat-sinked to run at their nominal optimal operating temperature before entering a further HEX. The current design requires a liquid cooling flow through the SC motor of only approximately 0.1 kg/s, which would account for an approximately 30% drain of the LH₂ tank. Liquid tank withdrawal is achieved with the use of specific LH₂ pumps that can provide the pumping flow at light pump weight and durability for decades and can be maintained and replaced, if required. For aircraft applications, those pumps do not yet exist but are being developed (see Section 4.1.2.5). One major problem with pumps is the susceptibility to cavitation the pump experiences and the immediate destruction of piston rings. Furthermore, one pump stack is designed for supporting three to four tanks. Attached to the pump outlet are transfer lines, called cable cryostats (CCs), that can be either flexible or rigid. Fortunately, those CCs have been available since the 1960s and are now at a very high TRL, making them ready to be integrated in an airframe structure. The same applies to CCs with integrated SC power lines.

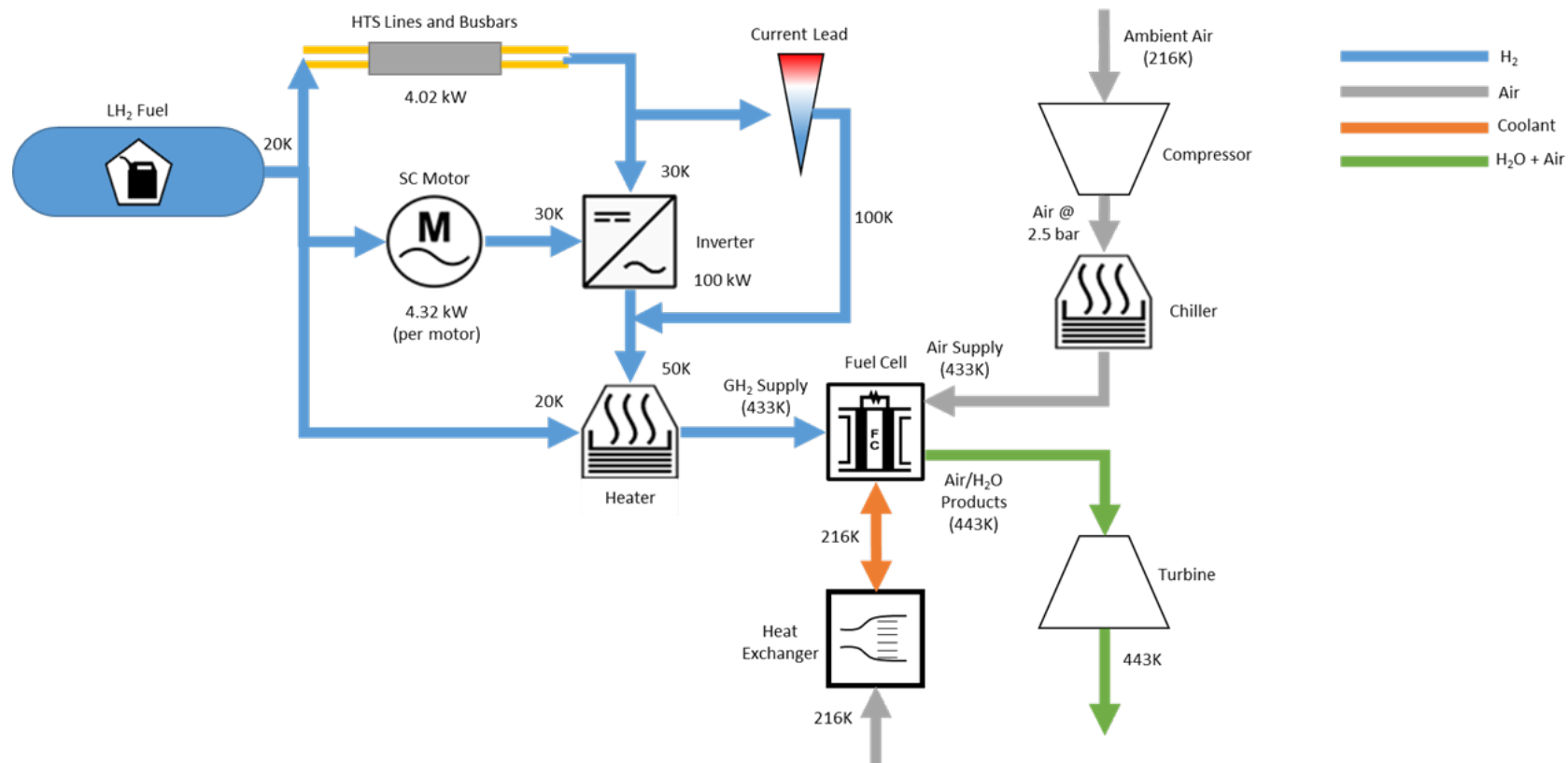


Fig. 254.—Cryogenic flow circuit of an all-electric aircraft.

A bypass line also leads the LH₂ flow directly to a HEX to generate additional gaseous flow for maintaining the fuel cells. As fuel cell technology advances, the H₂ gas flow that accounts for 70% of the LH₂ tank drain will be reduced. There may be additional components in an aircraft that can benefit from operating at a CT that we are not aware of yet. Many components tend to work and respond much faster due to the low electrical resistance of their metallic components.

Depending on the aircraft range (mid- to long-range) the fuel cell requirement determines the LH₂ tank volume and size. As shown in Fig. 254, the cryogenic mass flow through motor and HTS lines is also returned to the fuel cell after component cooling has been accomplished.

4.1.1.1 LH₂ Storage Tanks

Fig. 255 shows possible tank locations in a narrow-body all-electric aircraft. Designing cryogenic storage tanks of a size required for a mid-range aircraft is challenging for a number of mechanical, thermal, and assembly reasons. Fig. 256 displays a cross-section of a cylindrical tank visualizing various cryogenic design aspects that need to be understood, accounted for, and designed for when developing an aircraft lightweight LH₂ tank for enduring many thousands of fueling and liquid withdrawal cycles. The cross-sectional view is key for understanding thermal as well as mechanical tank aspects. One of the key design parameters when creating an efficient LH₂ tank is the insulation scheme and the tank build, which safely seals the cryogenic environment from RT.

A possible promising approach is a LH₂ tank enclosed in a vacuum environment, which acts to reduce the main heat losses summarized in the annotations shown in Fig. 256. Steps to consider when minimizing the heat ingress into the liquid volume include minimizing thermal radiation into the cryo environment; maintaining a vacuum environment; optimizing any tank suspensions and openings, feedthroughs, or sleeves; ensuring any outgassing from composites or metal surfaces and other components is well controlled and tolerable; estimating the sloshing effect on thermal and mechanical loads; understanding the thermal load on the tank induced by vibrations that are being transmitted to the tank by the airframe structure; and understanding the tank temperature as the aircraft undergoes altitude changes, to name but a few.

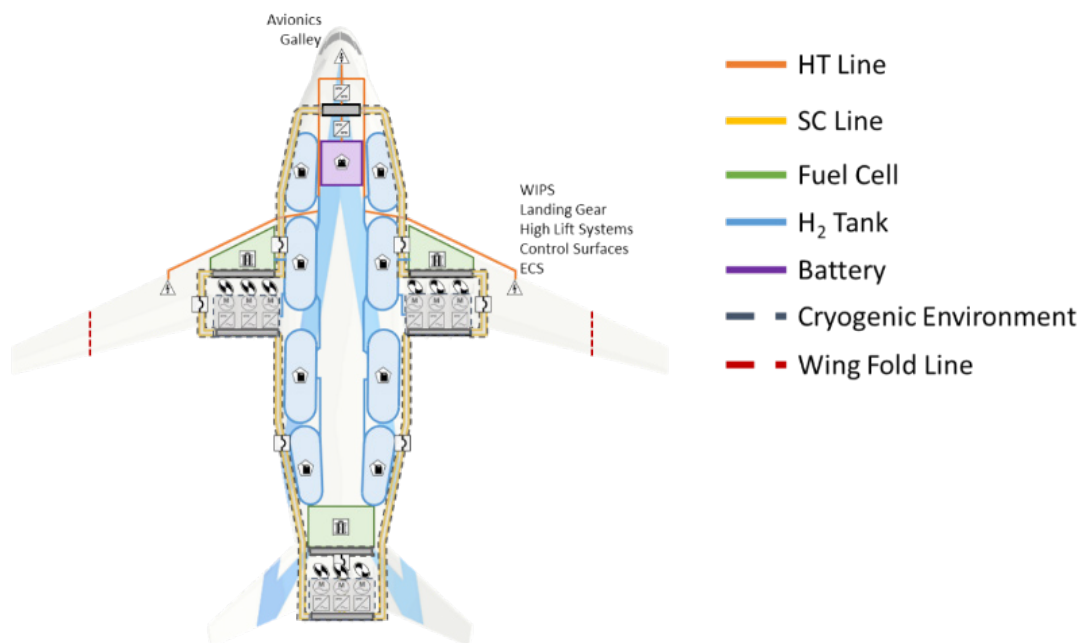


Fig. 255.—CHEETA tank locations.

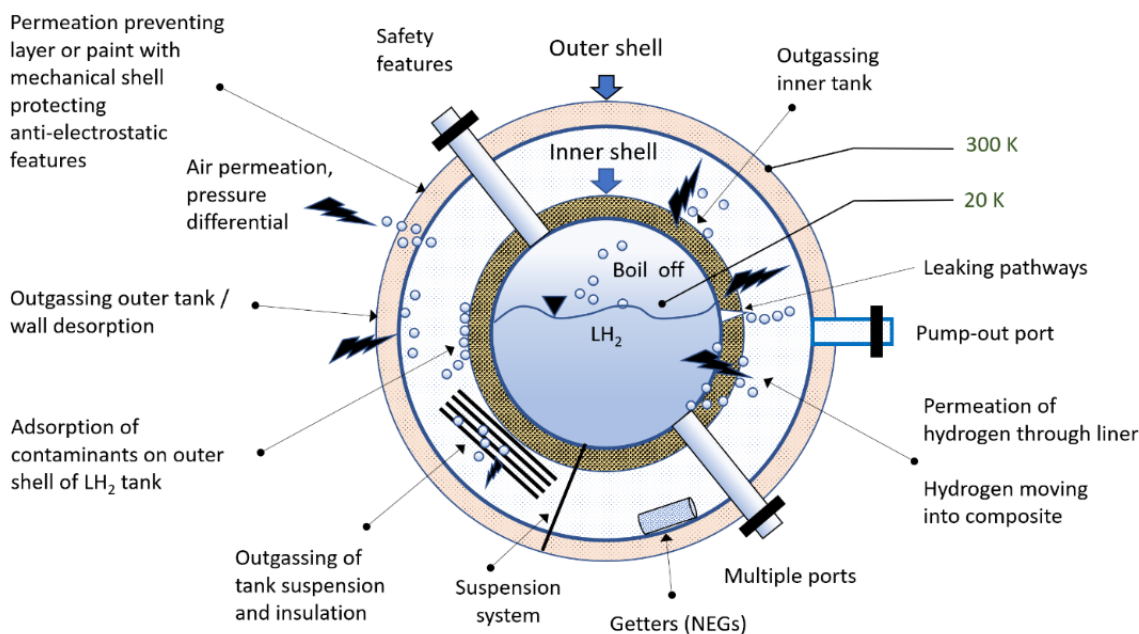


Fig. 256.—Assembly of storage tank walls.

Within the CHEETA program, excellent progress was made in designing a composite LH₂ tank with low mass fraction. We basically need to include a liner material to ensure LH₂ does not creep into micro-pathways of any composite structure or diffuse through any composite matrix (permeates), as shown in Fig. 256.

4.1.1.2 Static Shell Loads

Three different tank designs made of Al (2219-T87/6061-T6) and composites (high-strength carbon-fiber-reinforced polymer (CFRP) and glass-reinforced plastic (GRP) have been considered as shown in Fig. 255 and Fig. 257. Each of the tank designs consists of an outer tank, a vacuum space, and an inner LH₂ tank. The lightest weight design that provides adequate factor of safety against two failure modes is being considered and evaluated for practical use. The inner shell is subject to internal pressure, where the cryogenic tensile strength serves as the critical failure mode, while the vacuum shell is sized to prevent buckling, where the elastic modulus serves as the critical failure mode.

Most of the systems evaluated were designed for a composite matrix only, which allows the pressure vessel to be wound into different shapes and create the maximum liquid storage conditions. This needs to be done with caution, since it does not safeguard against any H₂ creep into a composite matrix. Other designs assume a stainless-steel foil and other foils applied on the internal surface of the tank. Another route is to combine all the aforementioned characteristics and choose a high-strength material that at the same time acts as an efficient H₂ barrier but also gives structural ability. Many calculations were run to obtain the minimum weight fraction for the inner and outer tank while ensuring adequate pressure-bearing capability to safeguard the tank against buckling. Based on Fig. 256, several tank structures were modeled. As an example, Fig. 258 shows the weight percentage of each component for tank 3. Note the small percentage of the tank insulation and liners, indicating a very small proportional contribution to the mass weight fraction.

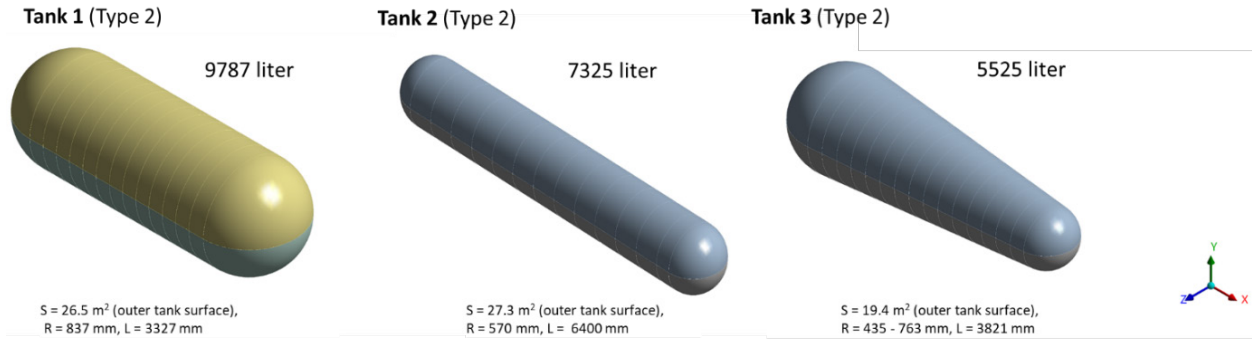


Fig. 257.—CHEETA concept vehicle tank types.

Components in % based on kg/m tank length,
with tank weight fraction = 0.685 (kg/kg LH₂)

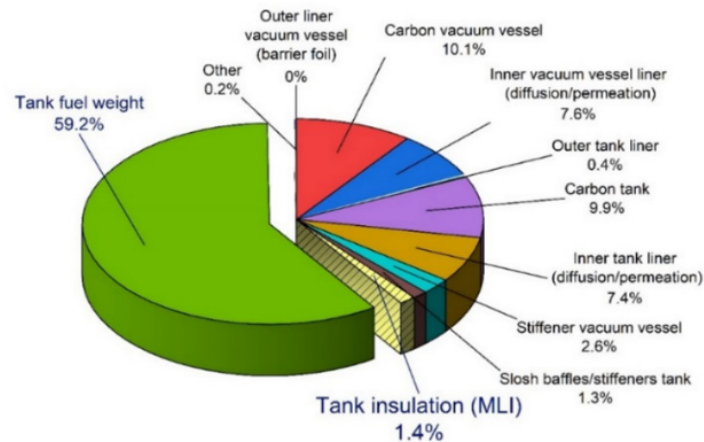


Fig. 258.—Design example: distribution of tank weight.

4.1.1.2.1 Minimum Tank Mass Weight Fraction

A very important parameter for defining low-weight tanks is the tank weight fraction (*TWF*) that compares various LH₂ tank designs. The most common definition we would recommend is as follows:

4.1.1.2.2 Minimum Tank Mass Weight Fraction (Conventional)

$$TWF = \frac{W_t}{W_f} \quad (63)$$

where W_f is the mass of the fuel in the tank and W_t is the total tank mass, including the outer shell and inner liquid container, with components.

4.1.1.2.3 Gravimetric Storage Density (After Verstraete (Ref. 253))

$$\eta_{grav} = \frac{W_f}{W_f + W_t} \quad (64)$$

This term represents the fraction of LH₂ stored in the tank for the overall liquid tank system mass. In that case, we can call this the gravimetric storage density (GSD).

4.1.1.2.4 Results Summary

Table 38 shows the achieved values for three tank types and three wall thickness types (T1 to T3) for Al-lined carbon fiber tanks comparing lowest weight fraction with maximum achievable storage density.

One finding was that low-pressure cryogenic composite tanks do not benefit from high-strength carbon fiber, and glass fiber may be more than adequate if fatigue loads are favorable (needs further investigation). Fig. 256 also shows the need for implementing a dedicated suspension system that allows for ease of assembly, bears the dynamic load, and does not contribute much to boiloff. Fig. 259 shows the conventional suspension elements for cryogenic use (e.g., magnetic resonance imaging (MRI) systems), and Fig. 260 shows how those conventional suspension elements can be used for a composite LH₂ tank.

Given the length of the pressure vessel, which can vary from 3 to 7 m, assembly is difficult. As an important finding, a novel suspension system was developed that allows for rolling of the inner tank while the outer shell is being supported by the inner tank wall during evacuation of the intermediate space. Fig. 261 shows the principal design for bearing static and dynamic loads. The inner vessel can be rotated and locked in place by a locking mechanism.

TABLE 38.—MINIMUM TWF VERSUS GSD FOR CHEETA AIRCRAFT

Tank	Type	Aluminum layer, mm	Carbon fiber layer, mm	Weight fraction	Storage density
Tank 1 (9787 L)	T1	1	2	0.4587	0.6855
	T2	2	2	0.6718	0.5982
	T3	1	4	0.7122	0.5840
Tank 2 (7325 L)	T1	1	2	0.6338	0.6121
	T2	2	2	0.9296	0.5182
	T3	1	4	0.9871	0.5033
Tank 3 (5525 L)	T1	1	2	0.5984	0.6256
	T2	2	2	0.8775	0.5326
	T3	1	4	0.9312	0.5178



Fig. 259.—Composite suspension straps and rods.

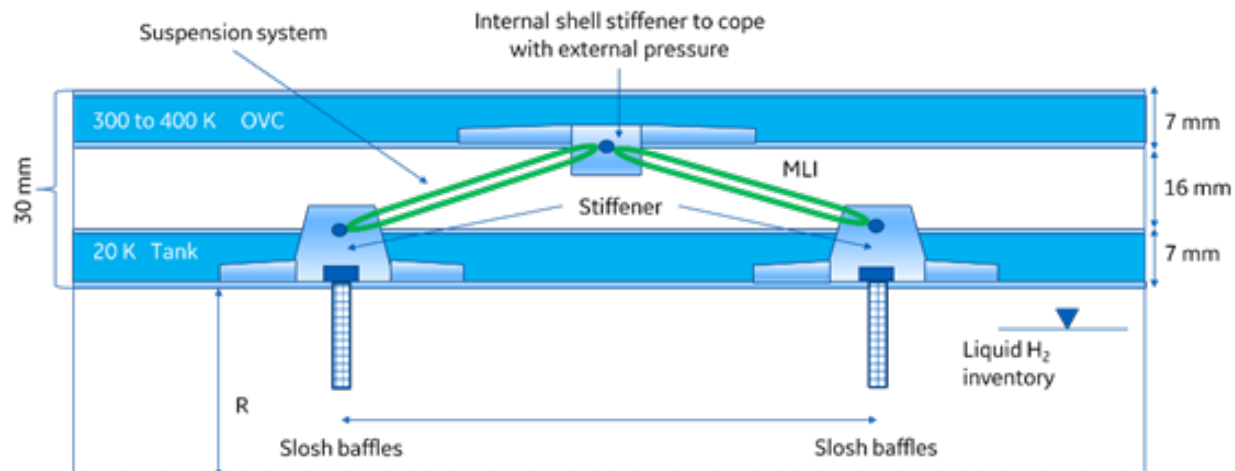


Fig. 260.—Conventional suspension elements for a composite LH₂ tank.

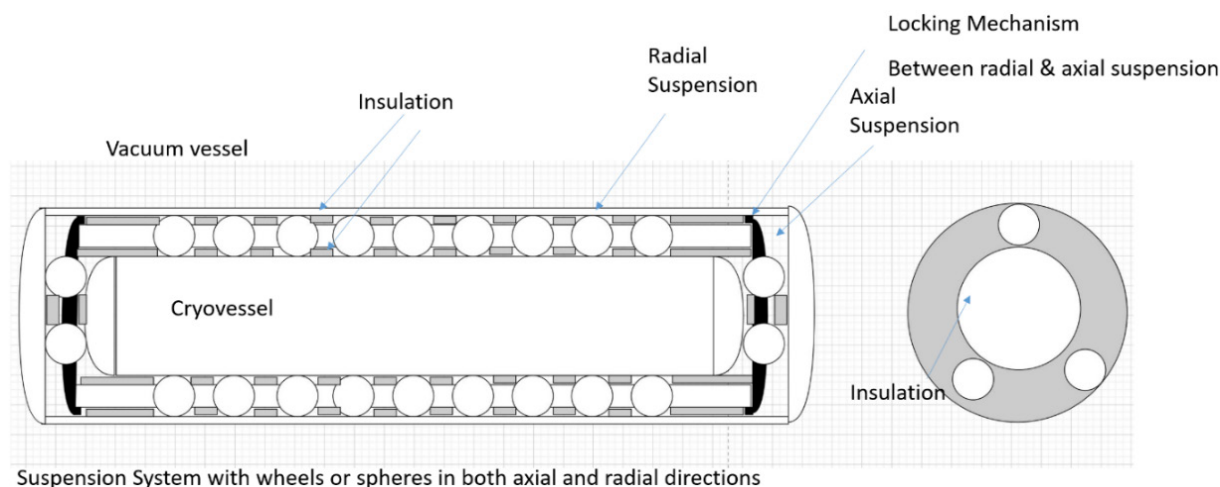


Fig. 261.—Suspension system based on ceramic spheres.

4.1.1.3 Dynamic Shell Loads

Dynamic loads that in general need to be minimized also act mechanically and thermally on the chosen suspension system. We started a program within the CHEETA aircraft framework to understand the dynamic liquid movement of LH₂ in CHEETA aircraft-type thin-walled composite tanks. Dynamic shell loads are a result of liquid movement in the storage tanks, for example, as shown in Fig. 257. For conventional aircraft designs, this is not a new topic, but it is entirely new for all-electric aircraft with a large tank storage volume of 8,000 to 10,000 L of LH₂. We found that sloshing indeed has several effects on the tank due to consequences of the liquid movement (moving around <1 ton of liquid/tank):

- Sloshing imposes stress concentrations on the liquid tank shell as well as on the suspension on the outer vacuum chamber carrying the weight of the liquid, which is especially challenging in the presence of multiple penetrations (see Fig. 256).
- The tank pressure rises due to that movement, caused by a corresponding temperature rise due to liquid/vapor mix (composite tank temperature profile circumferentially varies largely when not in touch with liquid surface).

- Increased bubble formation due to liquid vapor mix increases the cavitation risk for the attached LH₂ pump and puts further strain on the downstream phase separator when withdrawing liquid from the tank.
- Sloshing induces a dynamic imbalance force/moment on the aircraft (affects flight stability, lateral tanks as well as left and right tanks) that can be severed.

During tank defueling, additional load parameters need to be considered during takeoff, landing, and cruising. Sloshing does not occur if a tank is completely filled with liquid; however, in aircraft operation, a cryogenic tank needs sufficient ullage space of at least 10 to 20%, which allows sloshing to occur. After receiving typical g-loads on an aircraft from GE Aviation, we imposed those g-loads laterally and longitudinally on a large-sized liquid tank to conclude that sloshing effects indeed need to be considered. A solution strategy was therefore needed.

Sloshing software screening, benchmarking, and validation concluded that ABAQUS software would be best suited for this application (also used for LH₂ tanks for ground transportation and recently for LH₂-powered trucks). We simulated cryogenic sloshing (ABAQUS provides several options) and concluded that our solver benchmarking gave us the best method, especially for CHEETA aircraft configuration with large-sized, composite LH₂ storage tanks. The downside is that for large tank sizes (9,787 L; size $\phi = 1.674$ m, $L = 3.327$ m), the mesh gets excessively large and computing times tend to get long, even with high-performance computers. To address this, a coupled Eulerian-Lagrangian formulation available in the ABAQUS FEA package is employed to simulate LH₂ sloshing. To dampen the sloshing of LH₂, various baffle designs were considered. As an added complication, baffle performance (efficiency) varies with the LH₂ tank fill level (some baffles work with high fill level, some only at low fill level). We chose 30% and 70% fill levels for analysis, for data reduction, and for narrowing down the set of solutions.

We concluded the following:

- An optimized cross-winged baffle design can be found that is effective in dampening the fuel sloshing. Variables are baffle location within and on the tank, size, material, and construction (baffle with slots or porous, made of lightweight materials).
- Unlike other standard baffle designs, the cross-winged baffle was found effective in both longitudinal and lateral acceleration scenarios for the aircraft.
- Cross-winged baffles are an effective means for suppressing fuel sloshing independent of fill level. This was a surprising result.
- The chosen baffle design efficiently eliminates the need to compartmentalize the tank as required for Jet-A tanks.

In the next step, we planned to couple the mechanical movement of the liquid with the thermal solver to understand the sloshing effect on the tank temperature profile and boiloff. For this, we assumed temperature profiles in the composite tank based on actual tank thermal performance (insulation quality and so forth).

Fig. 262 and Fig. 263 show that semicircular and ring baffle designs (e.g., three-ring baffle) are ineffective in minimizing turbulence in the fluid with less than 50% fill level, whereas a four-cross-winged baffle quickly reduces the liquid movement.

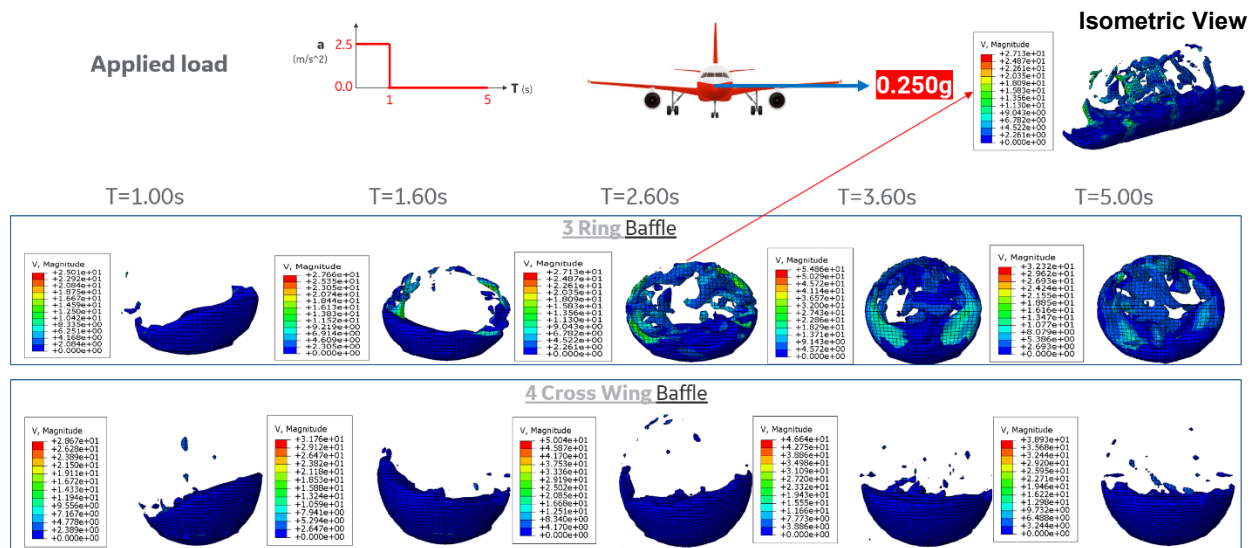


Fig. 262.—Baffle performance under lateral loading. Top: Applied load. Inset, top right: Isometric view of sloshing. Middle: Three-ring baffle. Bottom: Four-cross-winged baffle.

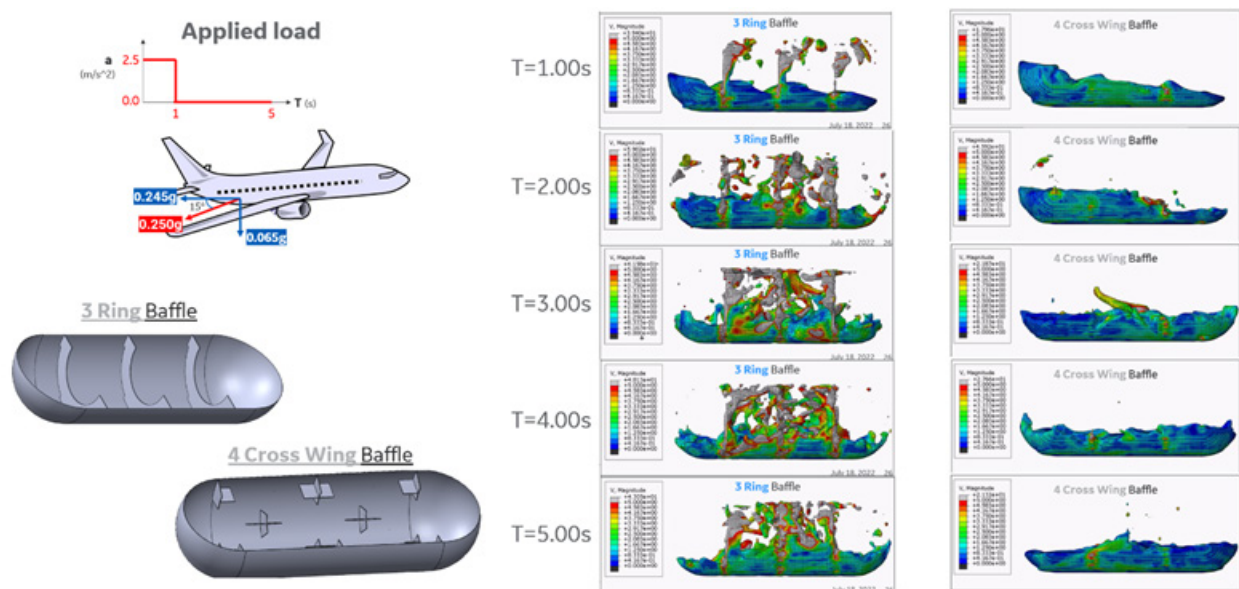


Fig. 263.—Baffle performance under longitudinal loading. Top left: Applied load. Bottom left: Three-ring and four-cross-winged baffle geometries. Right: Baffle performance for three-ring and four-cross-winged baffles.

4.1.1.3.1 Tank Vibrations

Mechanical tank vibrations translate to a thermal load increase, resulting in a higher boiloff than expected. This was first pointed out by Rotenberg (Ref. 254) when analyzing a BMW LH₂ tank. However, at this research level, it is still unclear in which way sloshing and vibration during cruising would affect the boiloff, or whether sloshing is the dominant factor of the two. This requires further analysis.

4.1.2 LH₂ Transfer Process

The transfer of LH₂ between tanks and to power systems features several complex considerations, from the systems necessary to ensure liquid volume is transferred to the logistics of maintaining insulation across transfer and fueling lines. Here, the capabilities and limits of bearings, pumps, transfer lines, valves and seals, and associated configurations are discussed.

4.1.2.1 Single-Tank Operation

Fig. 264 shows the schematics of a single-tank liquid withdrawal using a drainpipe connector to a phase separator with the vapor return siphoned back to the LH₂ tank. Ascent and descent angles for the LH₂ supply line need to be considered in the defueling process as indicated in Table 39 and Fig. 265. At no time should there be vapor in the liquid pump fueling line.

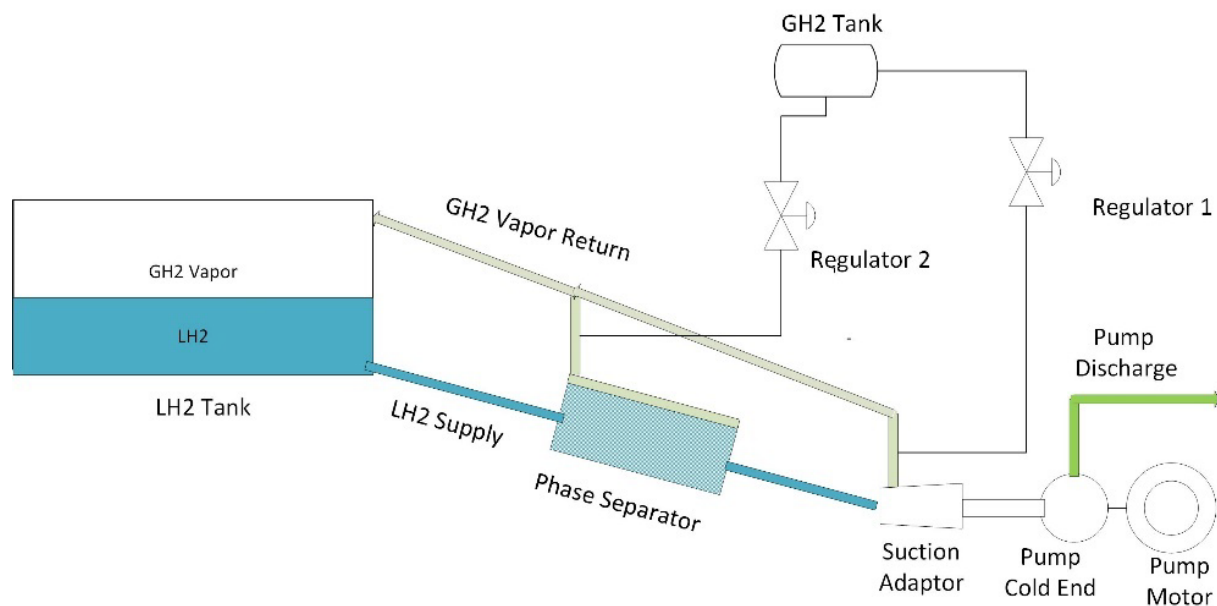


Fig. 264.—LH₂ withdrawal from tank (schematics).

TABLE 39.—PITCH AND EXTRACTION ANGLES

Mission phase	Aircraft pitch angle, deg	LH ₂ extraction pipe angle, deg
Ground idle	0°	25°
Takeoff	15°	10°
Climb	20°	5°
Cruise	0°	25°
Descent	–5°	30°

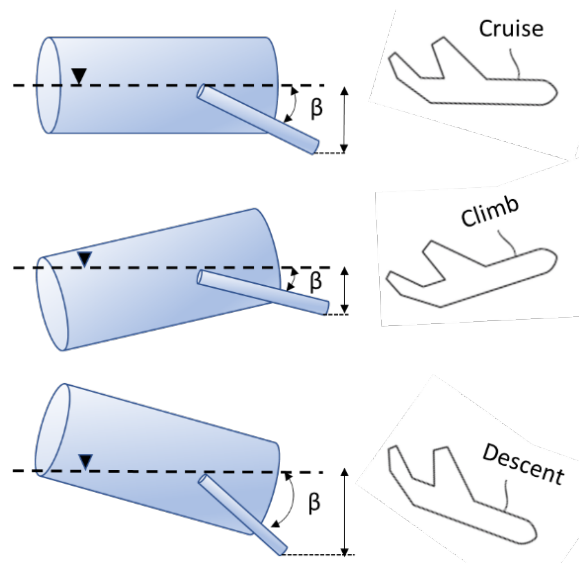


Fig. 265.—Pitch and extraction angles for flight profiles.

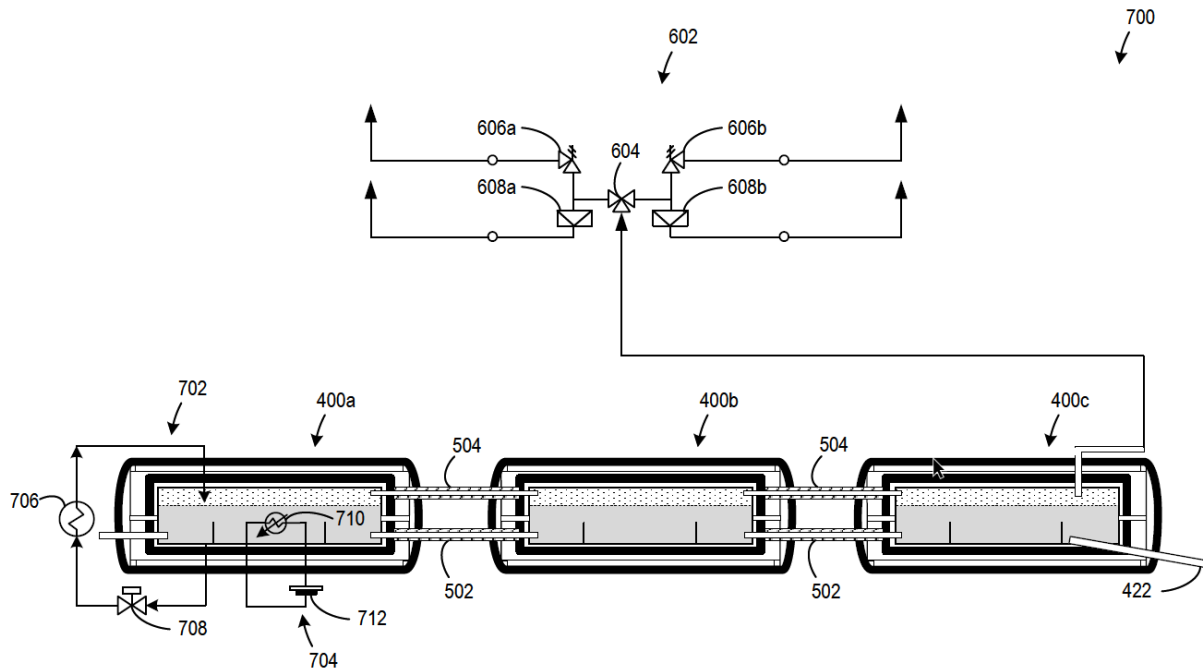


Fig. 266.—All tanks connected together with single withdrawal line.

4.1.2.2 Multitank Operation

Starting from a single tank, we also analyzed any multitank operation, as shown in Fig. 266. Understanding how the process of multitank operation works has been one of the key findings of the CHEETA program, with various options being covered broadly in the patent application. We concluded the following:

- The tanks across each semispan are served by different pumps, giving a fuel redundancy in case of pump failure.
- Novel strategies can be leveraged in cryogenic tank design that operates all or a subset of eight tanks as a single cryogenic vessel.

- All or a subset of eight tanks can be filled simultaneously from a single inlet port.
- Multiple tanks can be protected by a single set of pressure relief valves and burst disks.
- The extraction of LH₂ from all tanks can be performed simultaneously.
- All tanks with same LH₂ level can be configured to maintain the same saturated pressure.
- The tank internal pressure can be monitored and managed by a single thermosiphoning system and a single heater system.
- Multitank LH₂ fuel systems can be integrated to provide an isolation capability, redundancy, and redistribution capability to manage center of gravity placement.

Fig. 267 shows a further detailed process flow control system capable of running six to eight tanks in an aircraft with built-in redundancy.

4.1.2.3 Tank Diagnostics

GE Research studied the latest sensor technology for tank diagnostics and prediction of early tank failure in the composite tank structure (e.g., for indicating composite cracks). Some sensors (e.g., flat polymer piezoelectric sensors) can even be embedded in the laminate. Several ultrasensitive sensors for H₂ detection in vacuum have been identified at GE Research. This is important, since during a total breach of vacuum, a tank would empty within 5 to 10 min, depending on tank size and fill level.

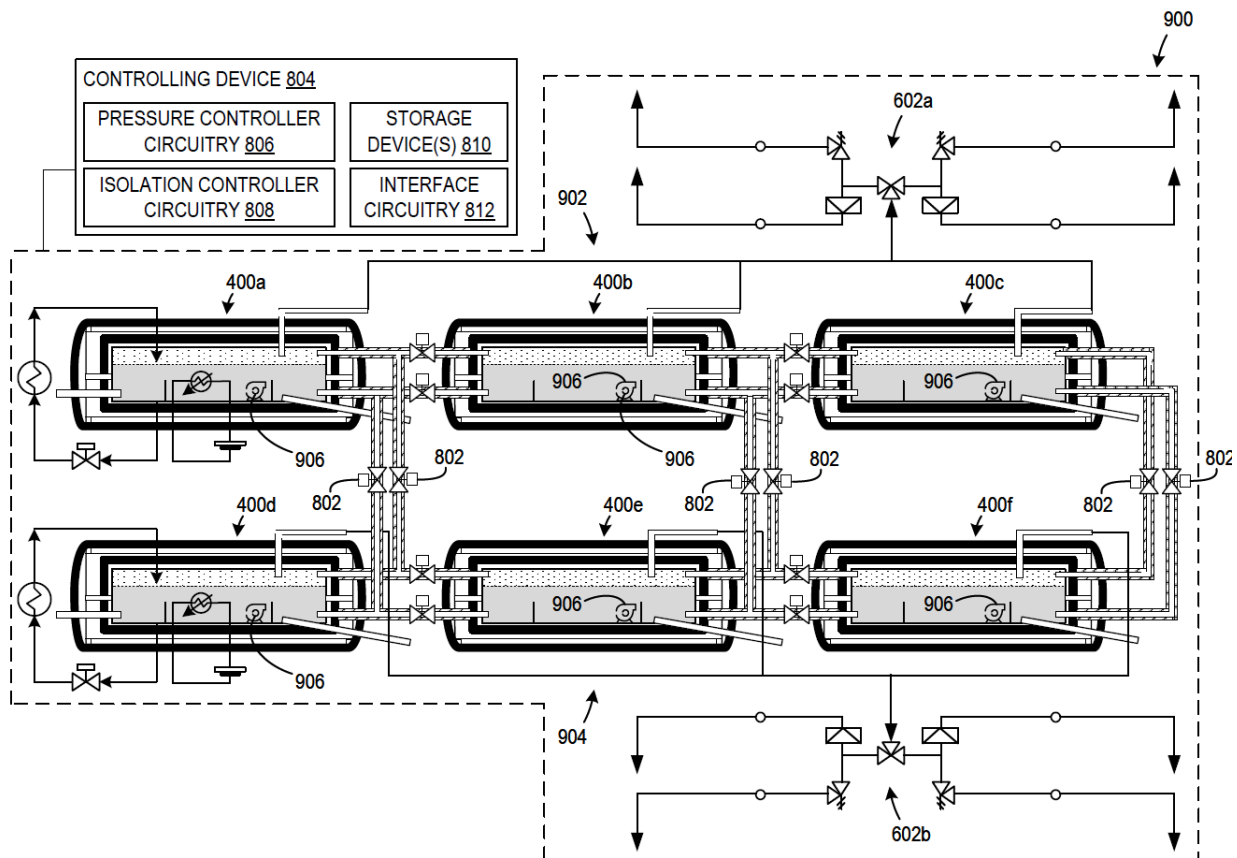


Fig. 267.—Multitank liquid withdrawal with complete process elements.

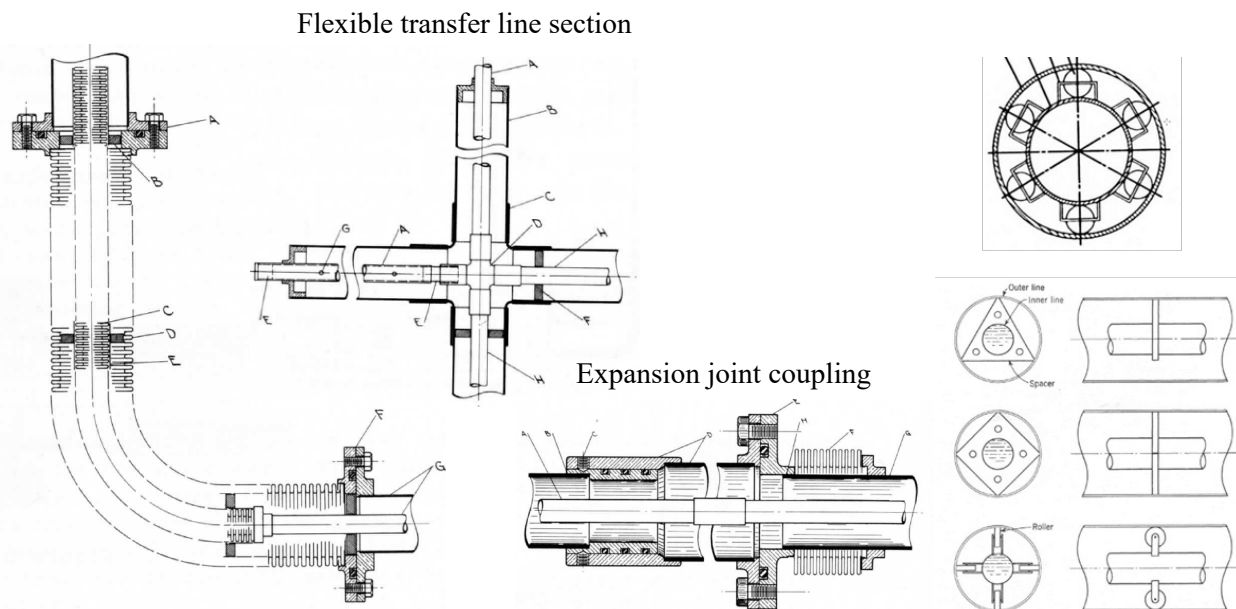


Fig. 268.—Rigid and flexible lines with ceramic spheres support (top right).

4.1.2.4 Cryogenic Transfer Lines

Referring to Fig. 254 and following Fig. 266 and Fig. 267, any LH_2 -powered aircraft needs cryogenic transfer lines as well as current-carrying transfer lines. We differentiate between cryogenic VJ liquid or gaseous transfer lines, and so-called cable cryostats (CCs). Although those cryogenic transfer lines have been in use since the early 60s, the long length in an aircraft requires consideration with respect to line shrinkage, mechanical stability, repeated coupling, and thermal loads on those lines.

GE Research showed that transfer lines with an internal rolling mechanism (see Fig. 268, top right) used with flexible line bellows are advantageous in bends, at crossovers, and with expansion couplings. In operation, these lines develop vapor bubbles that need to be transferred back to a reservoir or to a storage tank. Care needs to be taken so the liquid/gas fraction is high enough for current-carrying lines and for liquid transfer lines. GE Research developed a unique debubbling device on the transfer lines that ensures only liquid is transferred through the line. Several VJ lines with different spacers for managing longitudinal thermal expansion are shown in Fig. 268.

4.1.2.5 Pumping LH_2

Referring to Fig. 264 to Fig. 267, we found that present-day LH_2 pumps are inadequate for use on an aircraft. Therefore, specific pumps need to be developed with respect to durability, weight, and reliability and need to be safe against cavitation. The dry piston (no lubrication) operation of the cold end results in piston ring wear, for example, and needs to be replaced periodically (every 200 to 3,000 h). Typical piston rings are shown in Fig. 269 and are made of bronze-filled PTFE or polyether-ether-ketone (PEEK).

Pumping specifications need to be developed and determined for an aircraft (e.g., required flow rates, power rating, and discharge pressure) and defined for any designed flow circuit so that liquid withdrawal from H_2 tanks at inflight pitch angles, as shown in Table 39, can be achieved with minimum liquid depletion level. We ran a thorough market search on state-of-the-art cryogenic pumps. Some of those findings are shown in Fig. 270.

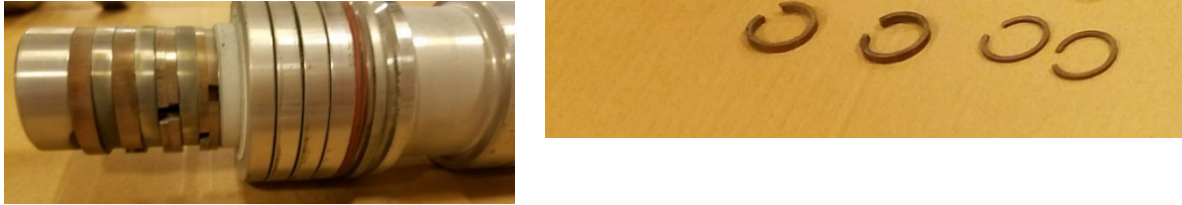


Fig. 269.—LH₂ pump cold end and disassembled piston rings (right).



Fig. 270.—Survey of cryogenic pumps for natural gas and LN₂.

There is currently no pump available with specifications that would be suitable on an aircraft. A dedicated pump design needs to align with the flow circuit, that is to say, all losses within any cryogenic flow circuit need to be calculated (Refs. 255 and 256) as summarized here.

4.1.2.5.1 Loss at the Pump

The loss at the pump is the difference between the mass of liquid that enters the pump container and that which enters the transfer line.

4.1.2.5.2 Flashing Loss

Energy is introduced into the liquid as it passes through the pump and through the transfer line; the former is the pump work, and the latter is heat leak. A part of the pump work and all of the heat leak cause liquid to vaporize (flash) as it is enthalpically throttled.

4.1.2.5.3 Cool-Down Loss

The cool-down loss is caused by the need to cool a transfer system down to its steady-state operating condition.

4.1.2.5.4 Trapped-Liquid Loss

The actual mass of liquid trapped in the system depends upon the detailed layout, including elevations. Based on those losses, the actual pump efficiency within a cryogenic flow circuit can then be calculated.

4.1.2.5.5 Cryogenic Bearings

Hydrogen-electric aircraft technologies require electric propulsors to achieve the goal for zero emission. Those electric propulsors are preferably SC with high current density, resulting in an increased power density. Propulsors can either be partially or fully SC. Our research shows a cryogenic cooling concept feasible for indirect cold mass cooling in the >20 K or higher temperature range, depending on conductor choice. For a number of reasons, we would not bath-cool the propulsor (direct cooling) but prefer an indirect cooling approach where field and armature windings are not directly exposed to H_2 .

The stator of this motor is exposed to the rotating magnetic field of the field coils, which rotate at 4,500 rpm in the CHEETA aircraft design. This rotation initiates eddy currents in the armature structure. Those AC losses need to be transferred to a cooling medium. In the proposed configuration, a helical cooling coil is mounted on the inner surface of the stator. The cooling coil is configured such that LH_2 can pass through the stator. The armature winding is cooled with high efficiency with LH_2 forced-flow boiling. The heat load generated from the armature due to those AC losses is substantial and may be around 2.3 kW.

We analyzed a novel rotating thermal intercept between a stationary armature and a rotating field coil not based on thermal slip rings, rotating cryocoolers, rotating conventional thermosiphons, or cryogenic immersion techniques, that is efficient and can be implemented and also adapted for different thermal loads. Instead, we proposed to develop a rotating thermal intercept modify a bearing design using high-strength copper alloy balls combined with gaseous-helium-filled heat-exchanging elements and rotating heat pipes. This allows us to transfer heat from the field coils to the heat sink (armature) as shown in Fig. 271. Our first analysis showed that those bearings can efficiently transfer heat between rotating surfaces. Further details are given in the patent disclosure and in Reference 257.

4.1.2.6 Hydrogen Fuel System: Conclusions

The effort to bring cryogenics and superconductivity “on board” an aircraft, while progressing, still faces several considerable gaps in key technology areas. In this CHEETA final report, we found that lightweight LH_2 tanks and pumps as key components require further development to achieve future maturation. As indicated, the TRL of a composite tank is still very low, despite some recent encouraging progress. To close the gap, the following considerations must be addressed:

- First is the selection of suitable composite fibers, or the development of specific high-strength carbon/glass fibers for specific reinforced structural tanks.
- Second, an enhanced epoxy matrix mixture with properties that work well at CTs needs to be found. Most of the resins we use derive from compositions and mixes created 40 years ago. There is considerable research going on with regard to a specific increase in certain material properties (e.g., thermal resin conductivity and resin strength), but those need to be validated until found suitable for future aircraft tanks.

To conclude statistically significant reliability of tank designs, feasibility studies need to be run on a large number of composite designs with a low-pressure rating (e.g., tanks made out of PEEK, fatigue tests on liners, etc.) to achieve minimum tank mass weight fraction that delivers a high payload. This will be best advanced by building a series of test tanks.

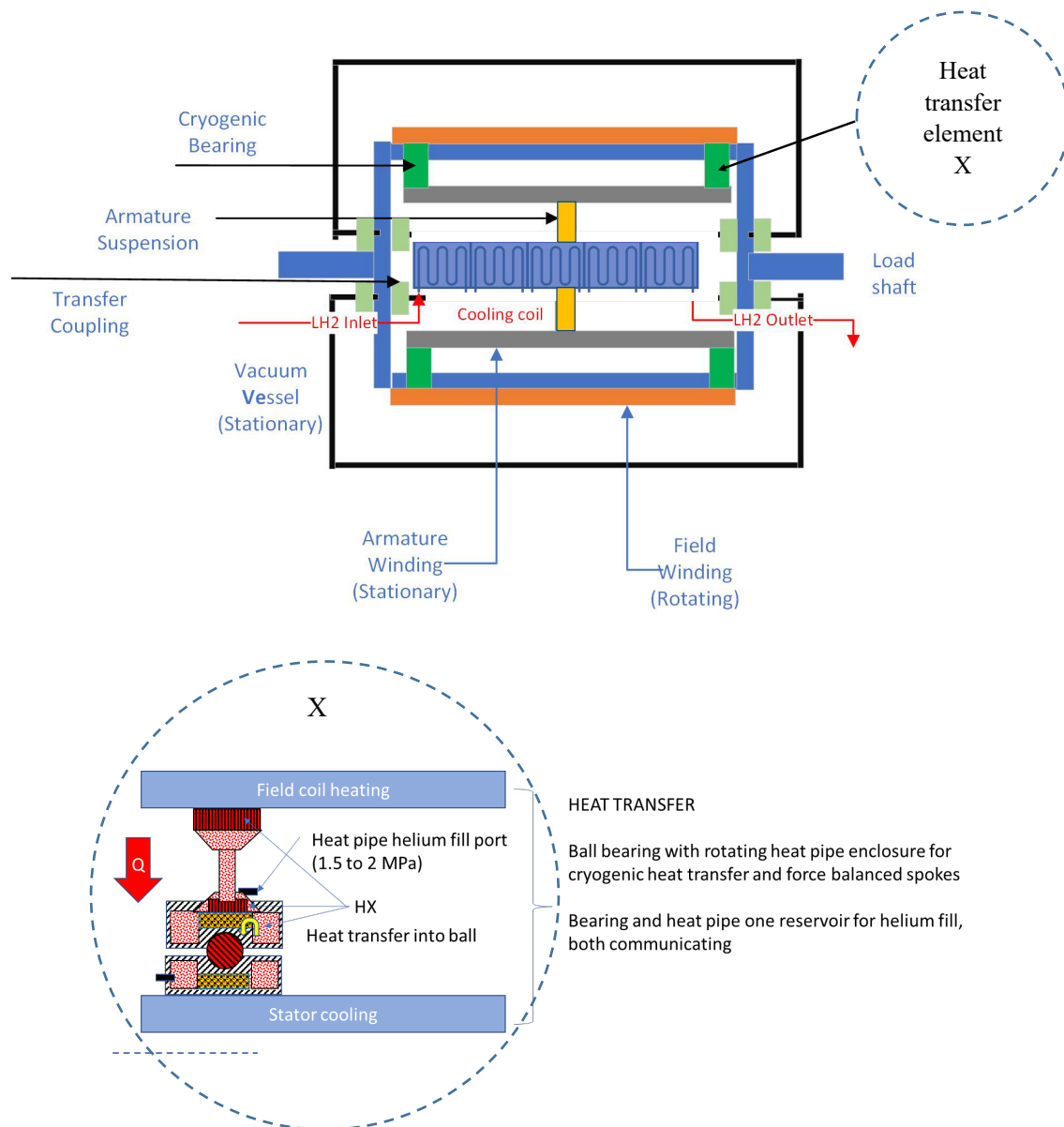


Fig. 271.—Superconducting motor with bearing cooling.

The cryogenic airframe infrastructure for the cryo aircraft has yet to be optimized. It needs to be refined step by step with respect to liquid and gaseous H_2 transfer lines (sizing and routing) to provide dedicated cooling temperatures and mass flows, with appropriate heat sinking capability for all components, including built-in contingencies, and for complying with all H_2 safety features.

The CHEETA project has laid the groundwork for the required cryogenic cooling infrastructure.

4.2 Power System

There are several possibilities for the use of H_2 as a fuel for aviation, with the simplest being a fuel-burning gas turbine propulsion system. H_2 -fueled reciprocating engines have been successfully used in aircraft, such as Boeing's Phantom Eye. Phantom Eye is a high-altitude, long-endurance experimental aircraft that flew nine times. Challenges facing H_2 -fueled aircraft include cyclic thermal shock, H_2 embrittlement of metals, leakage, leak detection, tank mass fraction, boil off, lack of durable aviation LH_2 pumps, immature infrastructure for distribution and storage of LH_2 , and impacts on terrestrial power grids when "green" H_2 is produced in quantities required for commercial aviation.

A natural extension of H_2 -fueled gas turbines is to incorporate SC cryogenically cooled generators to convert shaft power into electrical power. This approach is viable, and the overall thermal efficiencies are expected to be similar to simple-cycle power plants. These approaches can be adopted, but they are not truly "zero" emission, as the combustion process produces NO_x emissions that cannot be easily abated.

Another alternative is H_2 -fueled fuel cells, which produce electricity and water by electrochemically combining the H_2 with O_2 from the air. Various types of fuel cells are available; the most pertinent technologies of choice for use with H_2 are low-temperature proton exchange membrane (LTPEM), high-temperature proton exchange membrane (HTPEM), and solid oxide fuel cell (SOFC).

4.2.1 Fuel Cell System Design

While conventional H_2 fuel cells are electrochemical devices that generate DC electrical power by way of a redox reaction of H_2 and O_2 , the systems are constructed through the integration of several components. These configurations involve stacks of multiple cells that, together, provide the desired electrical power supply. Additional components are used within the fuel cell system to ensure effective operation of the fuel cell stacks, with conventional balance of plant components for proton-exchange membrane fuel cell (PEMFC) architectures including a compressor, expander, humidifier, intercooler, TMS, and various pumps, valves, and seals. The design of the fuel cell systems for the CHEETA aircraft are discussed here.

4.2.1.1 Fuel Cell Options

4.2.1.1.1 Low-Temperature Proton-Exchange Membrane Fuel Cells

LT-PEMFCs typically operate with a steady-state temperature between 60 and 80 °C. These have been successfully used in the automotive industry and range in power from 80 to 200 kW. Most available membranes are variations of perfluorosulfonic acids (PFSAs), such as Nafion[®] (The Chemours Company FC, LLC), which allows an operational temperature up to 120 °C. Operating temperatures above 100 °C require that the cell runs at a pressure above 1 bar. The membrane needs to be hydrated for performance and requires humidification of the gases. The control of humidification and the heat transportation are critical to the optimal function of this type of fuel cell. The need for a high water content practically limits the temperature of operation to 80 °C at 1 atm. The lower operating temperature also poses a significant thermal management challenge due to the reduced potential for heat rejection to ambient. Operation at higher temperatures has not been fully exploited due to the tradeoff between high temperature and component life. Efforts are underway to improve the life, performance, operating temperature, power density (kW/L), and specific power (kW/kg). There are additional issues from gas crossover, notably N_2 and H_2O from the cathode side migrating through the membrane to the anode side of the cell. This migration results in dilution of the fuel over time, and the H_2O/N_2 has to be purged periodically along with some H_2 , leading to a 3% loss of H_2 (relative to total H_2 consumed).

4.2.1.1.2 High-Temperature Proton-Exchange Membrane Fuel Cells

HT-PEMFCs are typically designed to operate between 160 and 200 °C. Membranes are generally nonfluorinated polymers, such as phosphoric acid doped polybenzimidazole (PBI). The primary issues with this membrane are poor performance relative to the LTPEM and acid leaching leading to corrosion of the components in the stack. The corrosion has been addressed by material choices and coatings, while performance improvements are still being made today. The primary advantages of HT-PEMFC technology are that there is no need for humidity management and thermal management is far easier. Research is still underway, and an alternate ion pair membrane developed by Sandia National Laboratory (Ref. 258) is currently under evaluation for use in HTPEMs. It may be several years before the full potential of this membrane is unlocked. Significant progress is expected in the area of HTPEM technology, with continued research funded by the DOE and other national and international agencies.

4.2.1.1.3 Solid Oxide Fuel Cells

SOFCs, operating between 600 and 1,000 °C, have garnered interest as the high operating temperature enables the addition of a bottoming cycle to recover energy from the exhaust stream. The typical electrolyte is yttria-stabilized zirconia (YSZ), which can be doped with other elements to improve ion conductivity. There are multiple configurations that are commonly identified by the support for the ceramic cells. They can be electrolyte-supported, electrode-supported, or metal-supported cells. SOFCs have been successfully deployed in the field using different fuel sources, including natural gas, renewable biogas, and H₂ (e.g., Bloom Energy).

4.2.1.1.4 Summary of Fuel Cell Types

The summary provided here is not a complete description of the choices of fuel cell types and does not include all of the materials that can be employed. A comprehensive compilation of fuel cell types is provided in the U.S. DOE Fuel Cell Handbook (Ref. 259). A summary of the status of the three technologies is compared in Table 40.

4.2.1.2 Potential Architecture and Loads Bookkeeping

Based on the choice of fuel cells, two primary configurations are possible, one for the PEM and the other for the SOFC. The basic configuration is given in Fig. 272. The SOFC, with its high-temperature operation, enables a bottoming cycle for waste heat recovery (WHR), with the generator producing additional power. The SOFC does not require any cooling and hence has a benefit of no specific thermal management requirement. The PEM system does require both a TMS and an electric motor to drive the compressor. The overall effect of the two different architectures leads to different loads bookkeeping requirements, as shown in Fig. 273.

Consequently, the PEMFC net power produced is lower than the power produced by the stack, whereas the SOFC net power produced is higher than the power produced by the stack. With the latest metal-supported SOFC technology, this system has a potential for much higher efficiencies than the PEM system.

TABLE 40.—SUMMARY OF EVALUATED FUEL CELL OPTIONS

Parameter	LT-PEMFC (Refs. 260 to 262)	HT-PEMFC (Refs. 263 and 264)	SOFC (Ref. 265)
Operating temperature	60 to 80 °C	160 to 200 °C	800 °C/650 °C
Stack power density	5.4 kW/L	TBD	TBD
System power density	0.64 kW/L	TBD	TBD
Stack specific power	5.4 kW/kg	TBD	TBD
System specific power	0.86 kW/kg	TBD	TBD
Fuel	H ₂	H ₂ , methanol	H ₂
Oxidant	Air	Air	Air
Efficiency at partial power	64%	57%	80%/77%
Efficiency at peak power	48% to 53%	40%	52%/50%
Cell power density	1.3 W/cm ²	0.85 W/cm ²	3.2/2.2 W/cm ²
Peak stack power	134 kW	NA	NA
Air stoic	1.5	2	Varies
Heat rejection	2.4 kW/°C	<1 kW/°C	NA
TRL	6	3	3

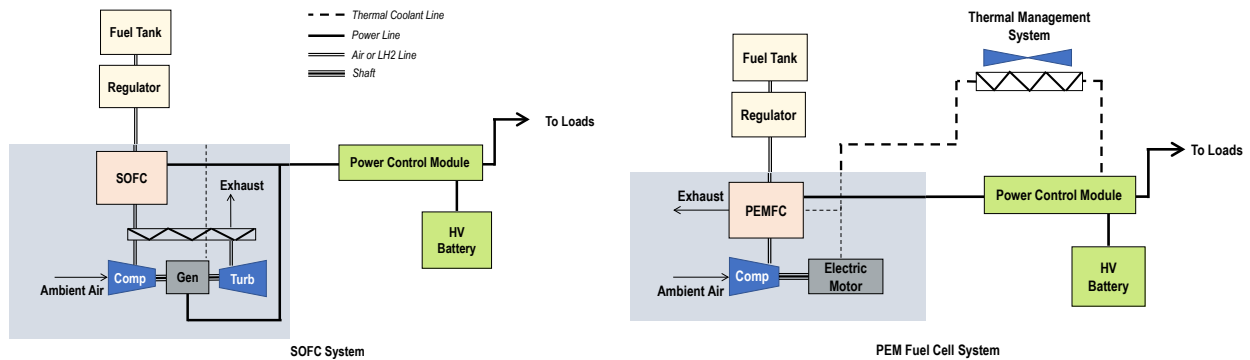


Fig. 272.—Fuel cell power system configurations.

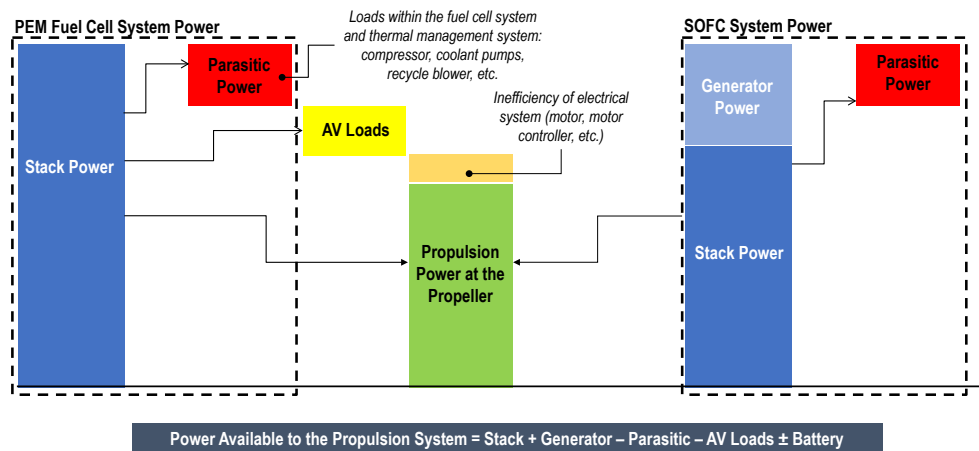


Fig. 273.—Bookkeeping of overall system-level loads.

4.2.1.3 Technology Projections

The CHEETA program requires projection of technologies to the year 2040 and beyond for an aircraft with entry into service around 2050. There are a few key metrics based on the ultimate DOE goals for automotive fuel cells that were assumed feasible in this time frame. The key metric was peak efficiency of 70% at or below 25% power. The stack efficiency target is >50% at full load. Combining these two metrics, the peak power density of the cell at full load was estimated to be 2 W/cm² based on the observed trends shown in Fig. 274.

Fuel cells have been used in the aerospace sector since the Gemini spacecraft program of the 1960s. Significant development work has been performed since the Apollo era, but limited data are available in the public domain. Over the last 20 years, increased work has been performed on kilowatt-scale PEMFC stacks, due in part to growing interest in a H₂ economy (Ref. 266). Performance evaluation of first-generation automotive fuel cells in 2018 by Argonne National Labs (Ref. 267) is a benchmark. The next-generation stack performance, published in 2022 (Ref. 268), shows a significant improvement in both performance and system-level weight and power density metrics. It is anticipated that the general acceptance of fuel cells in the transportation industry is expected to lead to a significant improvement in stack and fuel cell systems. The performance of PEMFC stacks has improved by over 50% in two decades. It is anticipated that it could reach 2 W/cm² over the next decade. Factoring in this trend leads to the CHEETA HT-PEMFC performance projections shown in Fig. 275.

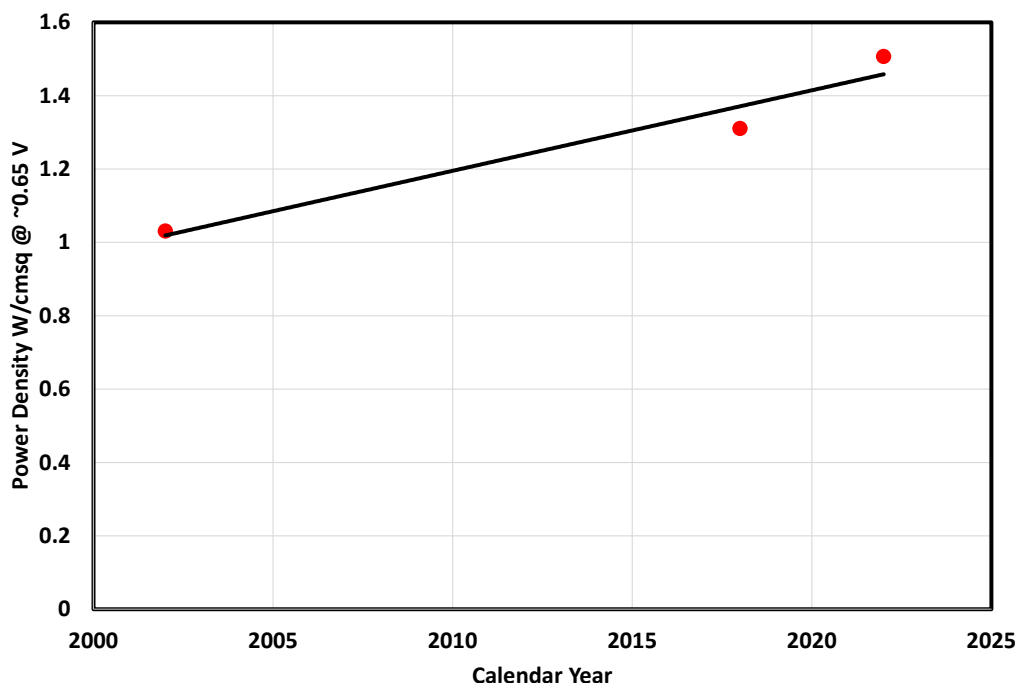


Fig. 274.—Cell power density at near constant voltage – industry data.

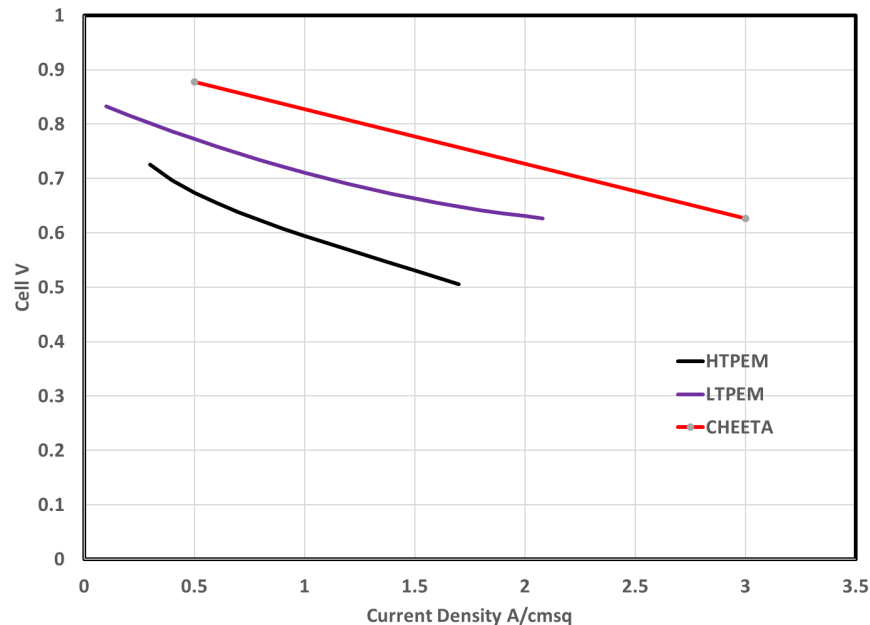


Fig. 275.—Comparison of current PEM technologies to CHEETA aircraft 2050 targets.

A comparison is made to the GEN1 automotive fuel cell performance and estimated current status of HTPEM data (Ref. 263). In the projected characteristics, the stack pressure varies from slightly above ambient to 2.5 to 3 bars. In addition, there is a metric for heat rejection established by the DOE where the $Q/\Delta T$ is 1.45 kW/°C or lower, where Q is the heat rejected to ambient and ΔT is the thermal potential as given by $(T_{coolant} - T_{ambient})$ and $T_{ambient}$ is assumed to be 40 °C. The ability to meet this target is a function of stack efficiency and stack operating temperature. Increasing the stack efficiency at rated power requires oversizing the stack, and increasing the heat rejection potential requires increasing the stack operating temperature. A detailed evaluation of thermal management indicates a preference for HTPEM over LTPEM. However, as can be seen in Fig. 275, there is a significant challenge in performance relative to the CHEETA aircraft performance target.

4.2.1.4 Fuel Cell Selection

All three fuel cells listed in the previous section were analyzed. The LTPEM suffers from a significant cooling requirement and was considered impractical. The HTPEM was considered feasible under the assumption that the cells can provide the performance defined in Fig. 275. This implies significant advancement in cell technology and requires corresponding investment in the development of HT-PEMFCs and stacks. Metal-supported SOFC has demonstrated the ability to achieve the performance assumptions of Table 40 and is a promising technology. However, SOFC was not considered practical by the CHEETA team due to the need to develop large-area cells and compatible sealing materials. Additional problems related to the containment of the high-temperature stacks, as well as the plumbing, were considered to have a detrimental effect on the overall system weight for SOFC. Based on these considerations, the final technology selected for the CHEETA aircraft concept is HTPEM.

4.2.1.5 Power Plant Configuration

The HTPEM membranes for this analysis are assumed to be liquid-cooled. The primary difference between the HTPEM and LTPEM stacks is the choice of materials. It is possible that the interconnects for the HTPEM could be Al or carbon composites. It is expected that species permeation through the membrane will be negligible, and hence the H₂ can be dead-ended. These assumptions lead to the overall configuration shown in Fig. 276.

The use of a WHR turbine is made feasible for HTPEM due to high stack air exhaust temperature (approximately 180 °C). This WHR offsets the power required to drive the air compressor. The exhaust temperature is maintained by the stack cooling and is essentially constant during most operating conditions. The HEX between the stack inlet and outlet serves the purpose of ensuring that the inlet to the stack is not too hot under conditions where the compressor exit temperature is elevated. A bypass line is required to ensure compressor stall margin is maintained under varying operating conditions. There may also be a need for an inlet particle separator and a filter to ensure the inlet air to the stack does not adversely affect stack performance. The exhaust area is sized to recover sufficient energy in the turbine and to also ensure that moisture is condensed. The overall system impact of the air inlet and exhaust configurations is a net positive, albeit insignificant, thrust.

4.2.1.5.1 Compression System

In the early stages of the program, the nominal required system power was estimated to be 26 MW, and it was assumed that all of the fuel cell stacks could be co-located. This assumption yielded the maximum corrected flow rate and pressure ratio of the compressor. The maximum stack pressure was also limited to 3 bars. Fig. 277 illustrates the compressor flow and pressure ratio requirement over the flight envelope.

These data and assumptions employed to develop Fig. 277 were used as a guideline to evaluate the performance potential and efficiency of the compression system across the flight profile. However, this is not the final flight profile, and subsequent performance optimization was generated using a standard 1-MW stack. There is a potential to consider standalone modules of 1 MW, each with its own compression and cooling systems. The corrected air flow rates will be nearly 26 times lower than the flows shown in Fig. 277, and the design would likely be a two-stage radial machine or a combination of axial and radial machines. As all of the compressor exit flow needs to be routed to the stacks and then back to the turbine, a combination of axial stages followed by a radial stage may be appropriate. This is particularly important if the stacks are distributed rather than co-located. In the final analysis, the stacks are to be located in clusters at three different locations in the aircraft that are coincident with the propulsors. This gives credence to the idea of axial stages followed by a radial stage for the compressor.

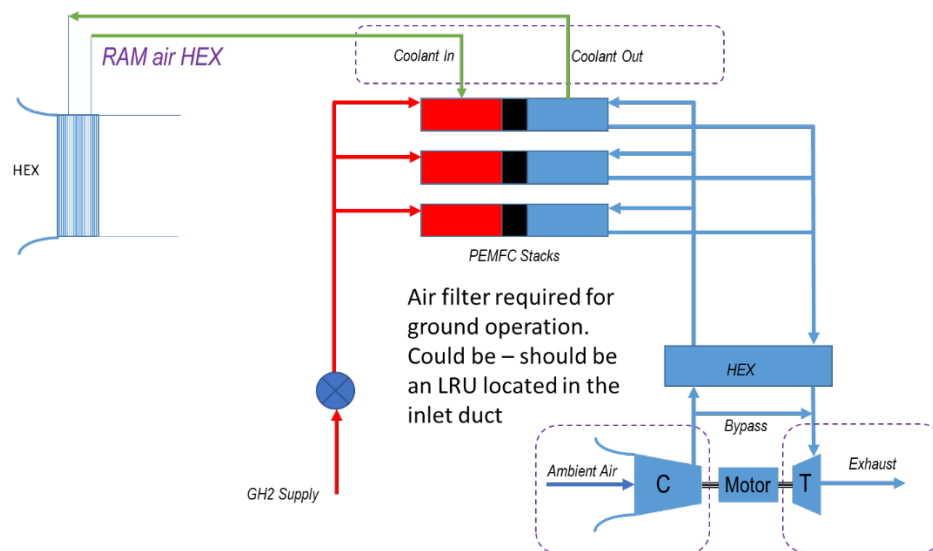


Fig. 276.—Schematic of overall fuel cell system configuration.

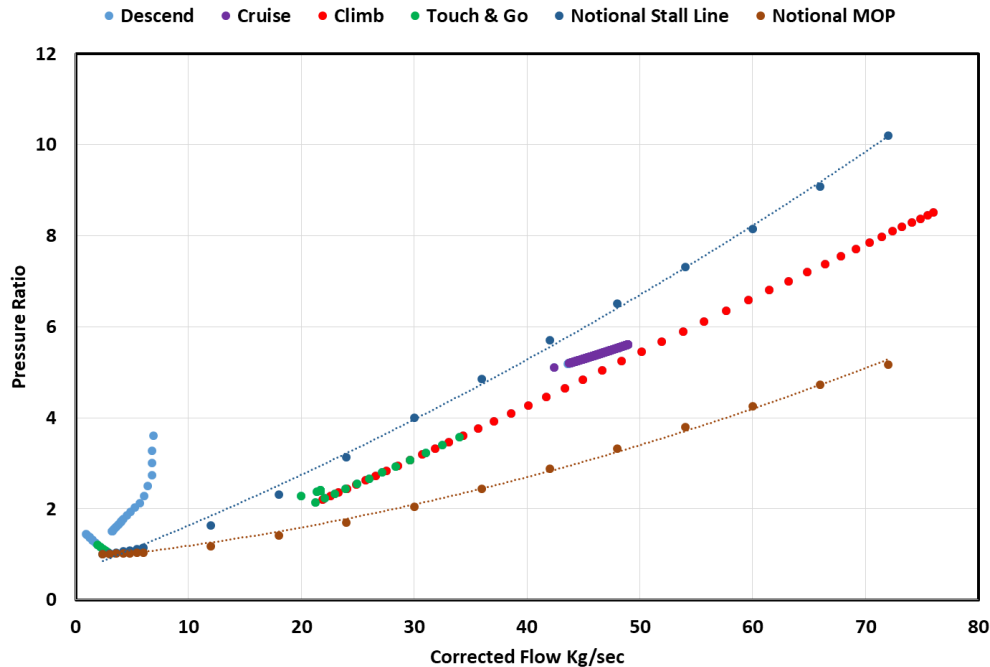


Fig. 277.—Typical compressor requirement over operating envelope.

4.2.1.5.2 Power Output and Conditioning

In general, the power from each of the stacks is conditioned by a DC-DC converter and the regulated power is sent to the main bus. This has the advantage of ensuring the bus is at a constant voltage and is not adversely impacted by stack-to-stack variation as well as changes in bus voltage with load. This adds significant overhead to the system in both weight and efficiency. The efficiency penalty could be managed if a cryo-cooled DC-DC converter is developed and employed. For this study, it was assumed this problem could be overcome in the future by running each stack to a nominal bus voltage as a function of flight condition to ensure stack-to-stack variations are accounted for and minimized. This leads to a nominal bus voltage that sags with load, and the motor controller/inverter drive needs to be able to perform under the varying bus voltage. In the absence of this ability, the system would need a DC-DC controller for each stack module.

4.2.1.5.3 Thermal Management

Fuel cell cooling is provided by a TMS comprising a ram air HEX, coolant pump, ancillary hardware (e.g., pipes, fittings, filters, valves), and individual stack regulators to ensure adequate coolant flow is provided to each stack. The basic stack module was considered to be 1 MW, and all the requirements were developed for this stack module. This enables the aircraft configurator to evaluate aircraft performance by scaling the total power as required. Several HEXs were evaluated, and their performance was accessed to enable an informed tradeoff between weight, pressure drop, and coolant flow required. At the time of this work, a compact plate-fin HEX was chosen to provide optimum design for this application. More recently (2023), Boeing performed further evaluation to arrive at better thermal management solutions, such as microtube ram air HEXs and pumped two-phase (P2P) cooling systems to improve thermal system size, weight, and (parasitic) power (SWaP) metrics. These more advanced approaches were not included in the power system configuration, sizing, and analysis presented previously.

Both LTPEM and HTPEM systems were analyzed. Fig. 278 shows the effect of fuel cell stack temperature on the ram air HEX weight and drag. The drag here is defined by the face area of the HEX

and the pressure drop. In an actual installation, the net drag (or thrust) produced will be a function of the final integrated configuration. A comparison of the LTPEM and HTPEM thermal management is provided in Fig. 278. The data presented are for a hot day takeoff condition where the heat load is maximized and the heat rejection potential is minimized due to high ambient temperatures. The evaluation was performed at an assumed hot day temperature of 35 °C. In practice, higher hot day temperatures may be expected, but these can be mitigated using water spray in the ram air inlet duct during takeoff and low-altitude operations. Although the LTPEM suffers significantly at higher ambient temperatures, it is not a problem for the HTPEM.

A key conclusion that can be drawn from Fig. 278 is that the LTPEM, with its relatively low operating temperature of 80 °C, would have a TMS with approximately 4 times the weight and 2 times the drag of an HTPEM. The data also shows that two different HEX designs were evaluated for the HTPEM power plant (i.e., PEM-HT1 and PEM-HT2). The intent of these designs was to illustrate the tradeoff between weight and drag. Put simply, a larger frontal area can be utilized in ram air HEXs to reduce drag penalties, but this comes at the expense of increased component size and weight. These size and weight penalties also translate to larger and heavier nacelles to house the thermal system. HEX integration into the aircraft was not part of the Boeing scope of work, and further evaluation is required to arrive at an integrated solution that optimizes aircraft performance.

4.2.1.5.4 Air Management System

In addition to the compressor, the air supply system includes a motor to drive the compressor and an exhaust system with a turbine to extract useful work and reduce the system power demands from the compressor drive motor. After accounting for component pressure drops, the typical pressure ratio available from the compressor exit to the turbine inlet is a function of both altitude and flight Mach number.

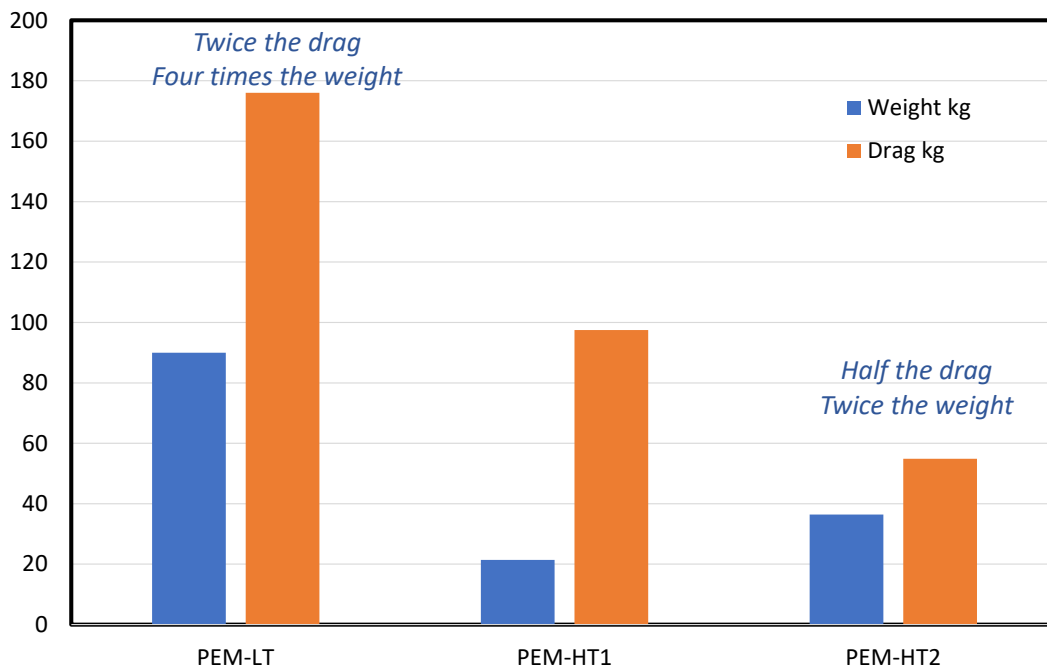


Fig. 278.—Thermal management impacts for hot day takeoff condition.

The design assumed fixed inlet and exhaust areas, where the exhaust area was sized to maximize turbine power contribution and minimize thrust contribution. An additional constraint was placed on the sizing of the exhaust nozzle to ensure the water vapor in the exhaust air stream will not freeze. The overall net thrust produced by the exhaust is shown in Fig. 279, for takeoff and cruise flight conditions. The power contribution of the turbine as a fraction of the compressor power is shown in Fig. 280. The net thrust/drag contribution from the system was considered negligible relative to the overall requirements and was not included in the final performance map.

At cruise, significant expansion ratio is available due to the inlet ram contribution. The HTPEM air exit temperature is also significantly higher than the LTPM. These enable significant turbine power contribution, where the system recovers nearly all of the power required at lower power settings. This is reflected in the projected system performance.

4.2.1.6 Performance Summary

In the early stages of the program, it was realized that design iterations were needed to define the fuel cell performance over the flight envelope and the fuel consumption. Both of these parameters significantly impact the overall weight of the aircraft and justify design iterations. To enable this optimization process, it was decided to generate a fuel cell map that defines fuel cell net efficiency as a function of altitude and flight Mach numbers for a 1-MW stack. In addition to the efficiency, the pressure drops through the ram air HEX and the ram air flow required were also provided to inform airplane design. Typical performance of the fuel cell power system at selected Mach numbers and altitudes is provided in Fig. 281.

Data in Fig. 281 show that system efficiency decreases with increasing altitude at the highest net power output. However, at lower power settings, there is crossover at the highest altitude of 37 kft and little difference between altitudes of 20 and 30 kft. This was evaluated further, and referring to Fig. 280, the cruise power contribution from the turbine essentially offsets all of the power required by the compressor. This was determined to be the primary cause, along with the efficiency assumptions of the compression system. This trend is believed to be realistic and would be affected by factors such as actual compressor efficiency, turbine efficiency, and nozzle exit area. This behavior was not observed in the LTPM evaluation because the turbine does not contribute enough power to offset the required compressor power.

The corresponding ram air flow required by the TMS is shown in Fig. 282, and the ram air HEX air-side pressure drop is shown in Fig. 283. The air flow increases with higher power setting and decreases with higher altitudes. This behavior is expected, as the ambient air temperature decreases with altitude. The required air flow difference between altitudes of 30 and 37 kft is very small. This is primarily due to the much smaller difference in the total temperature between the two flight conditions chosen.

The HEX pressure drop behavior is significantly different than the air flow behavior. In all cases, higher air flow leads to higher pressure loss through the HEX. The pressure drop decreases up to 20-kft altitude and then increases again with increasing altitude. The primary reason for this behavior is the reduction in ambient air density being faster than the reduction in air flow required and the corresponding increase in the air flow velocities through the HEX.

A performance map was provided to the airframe designer to scale the performance when changing the airframe configuration during its design iterations.

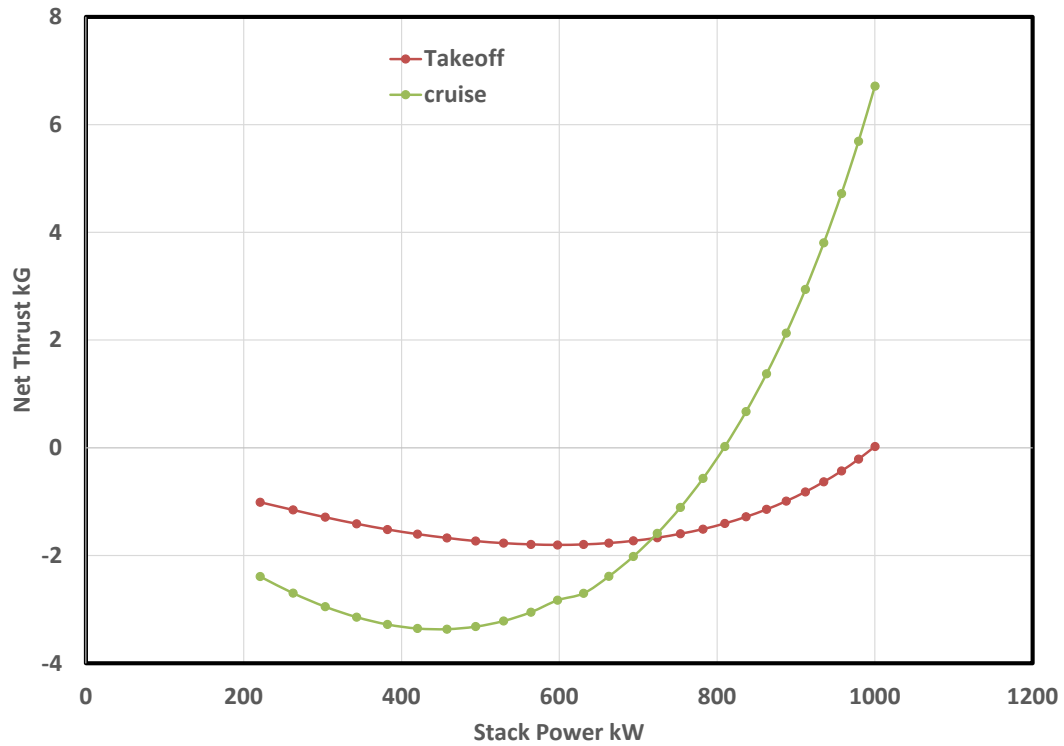


Fig. 279.—Net thrust from power system during takeoff and cruise.

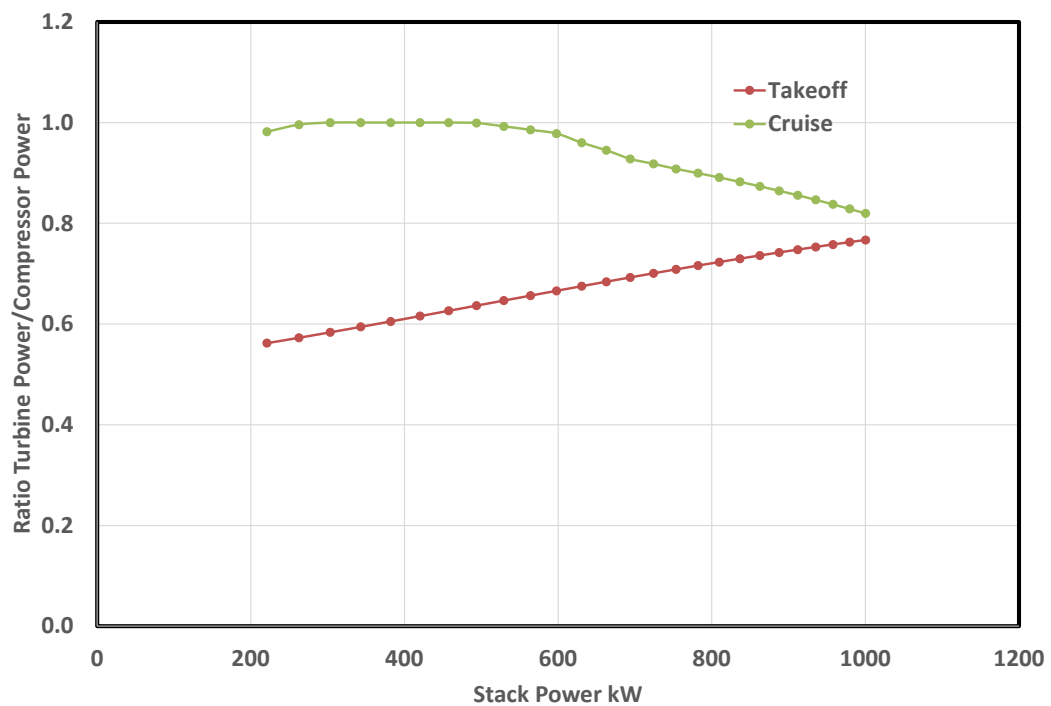


Fig. 280.—Turbine power relative to compressor power demands.

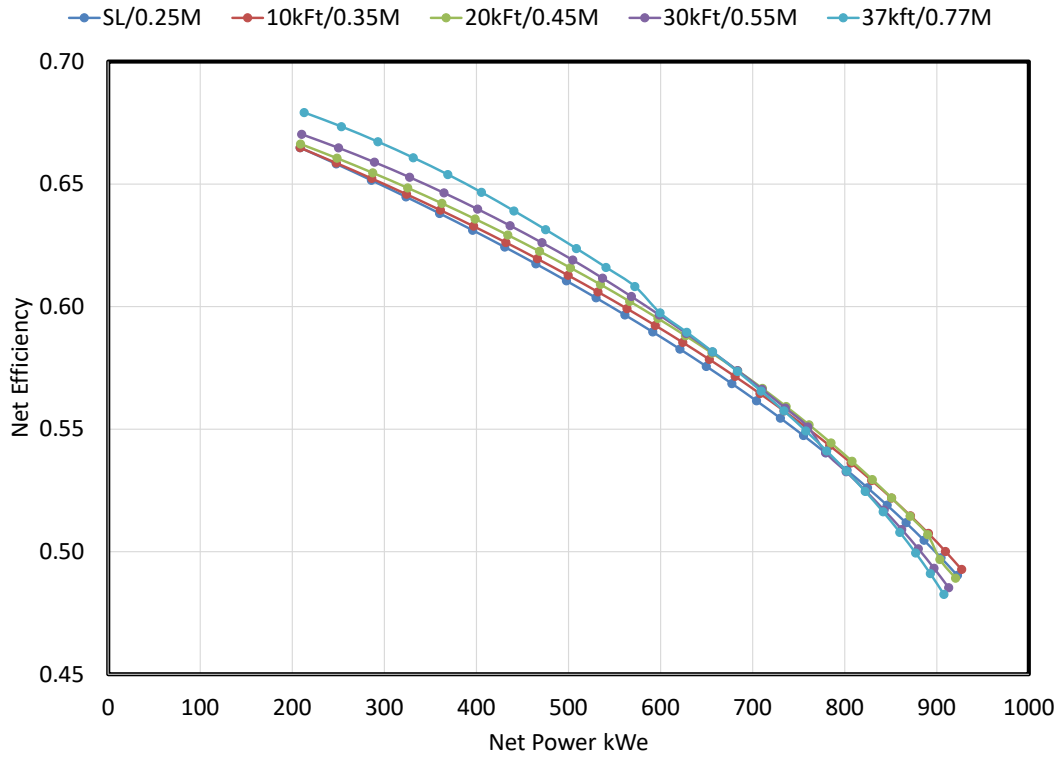


Fig. 281.—Power system performance for select flight conditions.

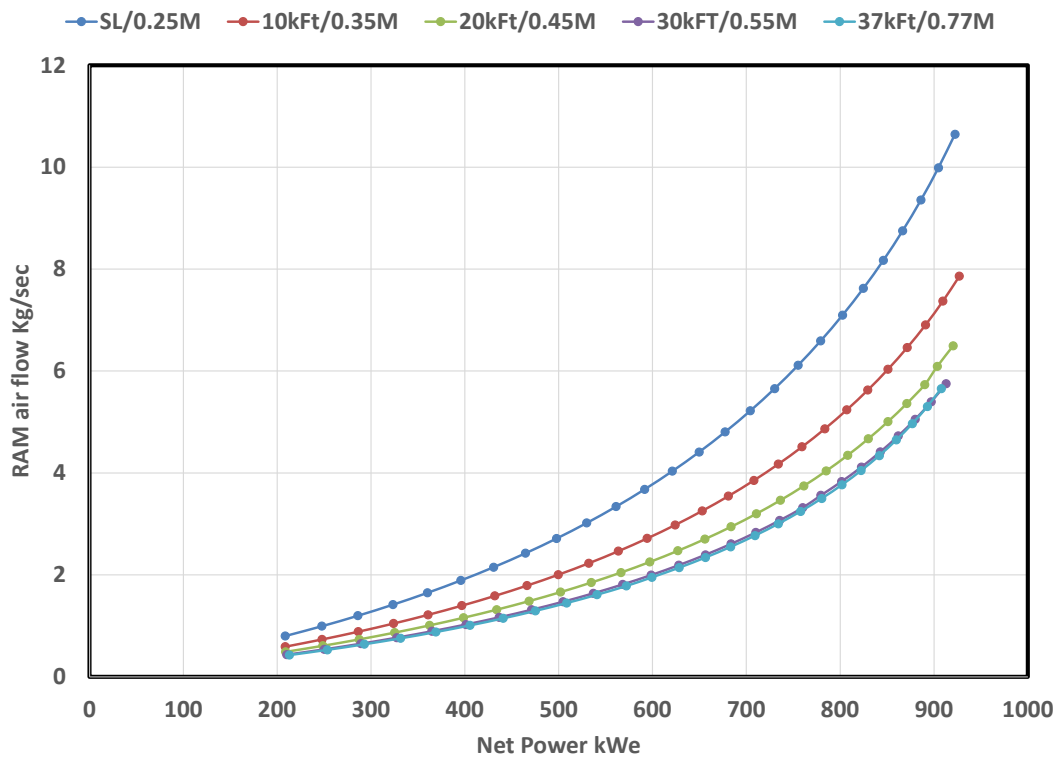


Fig. 282.—HEX air flow requirement.

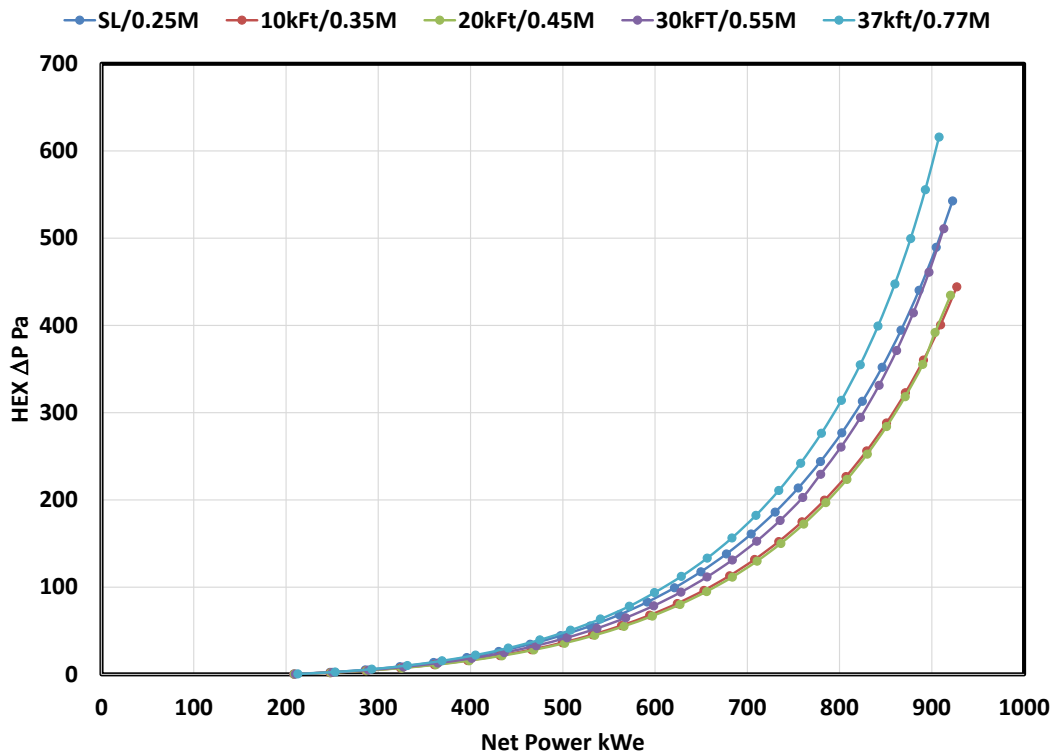


Fig. 283.—HEX air-side pressure drop.

4.2.1.6.1 Effect of Fuel Cell System Degradation

All fuel cells degrade over time, and the rate is generally in $\mu\text{V/h}$ at a given cell power density. The primary mechanism of this degradation is thermal and power cycling, as well as changes to the microstructure of the catalyst layer (e.g., agglomeration of the catalyst leading to reduction in active sites and contamination). From a system point of view, the effect of the anticipated degradation rate can be modeled as a change in the I - V curve, which impacts the final system performance. At the present time, a cell meeting all of the DOE goals and the CHEETA program targets does not exist. A more rigorous analysis can be found in studies from an Argonne National Laboratory project report (Ref. 268).

The best available data on HTPEM degradation (Ref. 269) indicate a rate of $\sim 0.5 \mu\text{V/h}$. This translates to 5 mV per 1,000 h. For the purpose of this effort, the degradation rates were assumed to be 1, 2, and 5 $\mu\text{V/h}$ at 0.5 A/cm^2 current density, and the effect of degradation rates was evaluated at the EOL performance of the fuel cell system over 20,000 h. It is assumed that the stacks could be oversized to enable meeting the mission targets at the EOL. In addition, the degradation was modeled as an offset and as a function of current density, as the effect of degradation could be larger at higher current densities. The degradation at 0.5 A/cm^2 was modeled as 80%, 75%, and 66% fixed for the 1, 2 and 5 mV/h and the remaining as a function of current density. This approach was considered reasonable for lack of better data. The resulting changes to the slope of the I - V curve were implemented in the system performance model to arrive at the net efficiency. Fig. 284 compares the effect of degradation at takeoff, and Fig. 285 compares the performance at cruise.

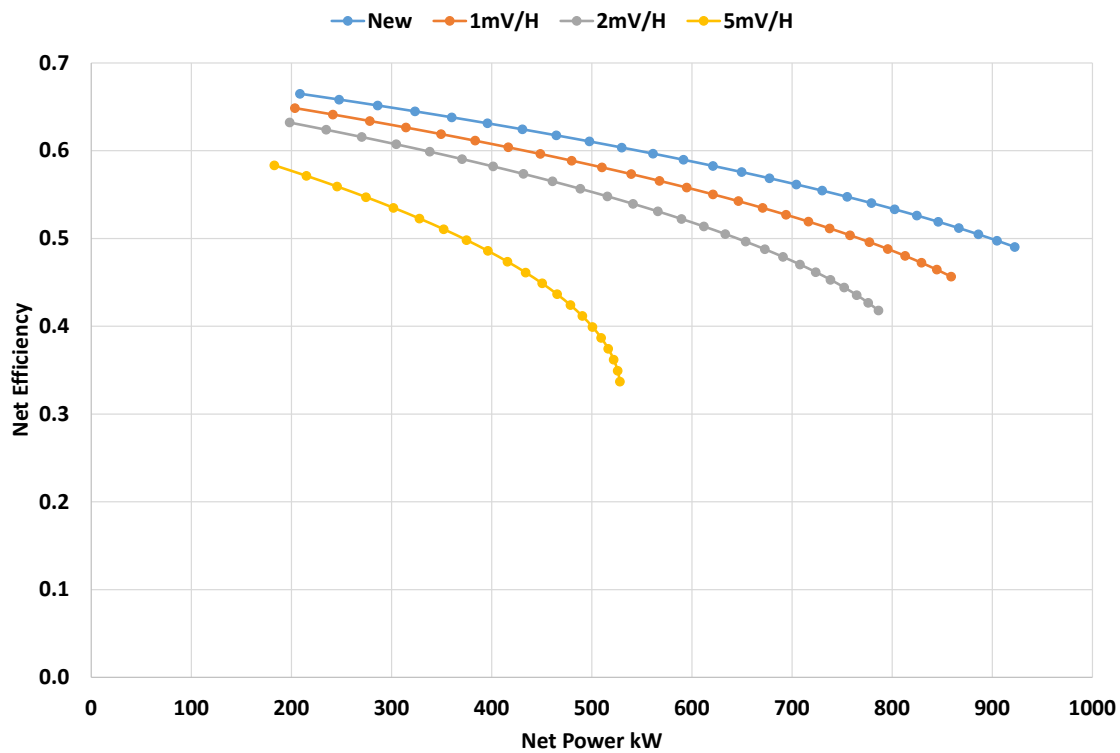


Fig. 284.—Effect of degradation on takeoff performance.

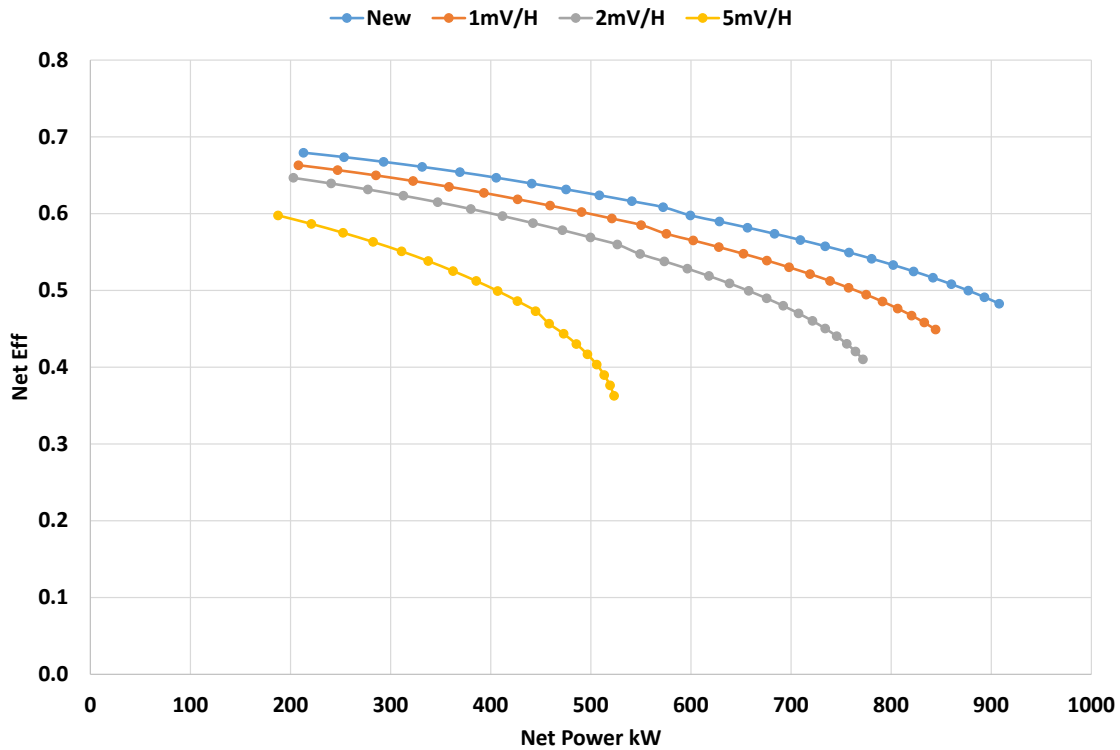


Fig. 285.—Effect of degradation on cruise performance.

It can be seen from this data that the 5- $\mu\text{V}/1,000\text{-h}$ degradation has a very high impact on system performance and could not be used for practical applications. The performance impact is similar for both takeoff and cruise. The impact on peak power is about 6.7% and 14.7% for 1 and 2 $\mu\text{V}/\text{h}$, respectively. The impact on efficiency at peak power is about 7% and 15%, respectively. However, the actual impact in cruise may be far less, as cruise power is significantly less than takeoff power. This work is not a complete analysis on the effects of fuel cell degradation, but it provides insight into the level of degradation that may be required for a viable system.

4.2.1.6.2 Overall System Weight

System weights were estimated based on the status of automotive LTPEM technology with assumptions about future technology advancement. The best-in-class systems have a gravimetric stack power density of about 4 to 5 kW/kg and a system-level gravimetric power density of about 0.5 kW/kg.

The following assumptions were made to estimate the potential weight reduction:

- Increase the active area from the current 45% of the total cell area, including flow passage manifolds, to the DOE goal of 65%.
- Improve the power density of cells from 1.35 to 2 W/cm².
- Change the interconnect material used in the automotive industry (e.g., stainless steel/titanium to Al).
 - This has been demonstrated to work with HTPEM (Ref. 263).
- Change cooling strategy from every cell to
 - Every other cell,
 - Every third cell (Ref. 270), or
 - Peripheral cooling to avoid direct cell cooling entirely.

These strategies result in a stack gravimetric power density increasing from the current level of 2.6 to 5.2 kW/kg. Under these assumptions, the weight breakdown and total power system weight for a 1-MW system are as shown in Table 41.

The weight breakdown of Table 41 yields a system-level power-to-weight ratio of 2.74 kW/kg, which is much higher than current technology and DOE goals but lower than other published estimates that imply a future potential of 8 kW/kg (Ref. 271). The current estimate of 2.74 kW/kg is considered to be reasonable and based on technology projections that exist. However, significant use of composites for interconnects may enable further weight savings and drive the specific power metric beyond 3 kW/kg.

TABLE 41.—POWER SYSTEM WEIGHT BREAKDOWN

Component	Weight, kg
Stack with end plates	193
Axial flow compressor (Ref. 272)	37
Ram air cooling system	37
Electric motors and controllers (4 kW/kg)	25
Miscellaneous	73
Total for 1-MW HTPEM	365

4.2.2 Advanced Thermal Management System

Managing the MW-class heat of a H₂-electric power system that would be used on a transport-class aircraft represents a key challenge to the feasibility of these aircraft concepts. For this reason, the purposeful and efficient design of the TMS represents a consideration of significant importance when configuring the CHEETA vision vehicle concept.

4.2.2.1 Plate-Fin Heat Exchanger Solution

The baseline ram air radiator employed for the CHEETA TMS is an Al plate-fin HEX in a cross-flow configuration, as depicted in Fig. 286. This HEX leverages current state-of-the-art heat transfer surfaces (i.e., fins) that are commonly found in aircraft environmental control systems (ECSs) and propulsion systems (Refs. 273 and 274). Coolant selection for this HEX study is based on the Mouromtseff number as described in the literature (Refs. 275 to 277). A potassium formate and water mixture was selected owing to its high thermal conductivity, but a glycol-water mixture would work nearly as well.

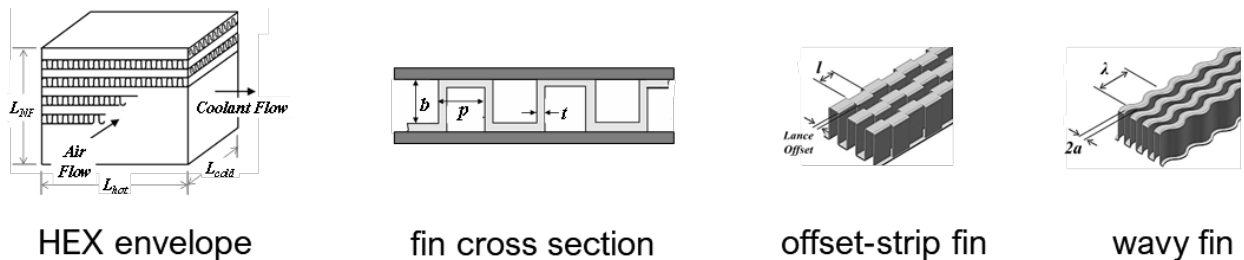


Fig. 286—Geometry definition for baseline ram air HEX.

TABLE 42.—TMS METRICS FOR BASELINE PLATE-FIN DESIGN

Inlet conditions	Flow side	Medium	Temperature, °C	Pressure, kPa	Volume flow, L/s	Mass flow, kg/s
	Hot	PF/W	180	413.7	4.80	6.1
	Cold	air	51.5	104.2	9,761	9.4
Fin geometry	Flow side	Type	b , mm	p , mm	t , mm	l or λ , mm
	Hot	Offset	2.540	2.540	0.1016	8.89
	Cold	Wavy	8.763	1.270	0.0762	19.05
HEX and pump geometry	Flow side	L_{flow} , cm	L_{NF} , mm	No. layers	HEX + pump volume, L	HEX + pump mass, kg
	Hot	172.72	83.01	69	171.16	58.86
	Cold	11.43		70		
TMS SWAP metrics	Flow side	$\dot{Q}$, kW	ΔP , kPa	Mass/ $\dot{Q}$, kg/kW	Volume/ $\dot{Q}$, L/kW	Pump power, kW
	Hot	1,003.9	22.45	0.059	0.170	1.40
	Cold		0.4047			

Details of the plate-fin HEX design, including inlet conditions, fin geometry, HEX envelope, pump size and weight, and SWaP metrics are provided in Table 42. The HEX is sized for 1 MW of heat dissipation during takeoff conditions, where heat loads are maximized and heat sink is minimized (i.e., high ambient air temperature and low forward velocity). The SWaP metrics for 1 MW of heat dissipation include volumetric and gravimetric heat densities (i.e., 0.059 kg/kW and 0.170 L/kW) and pump power (1.4 kW). An aircraft designer can scale these metrics to higher or lower heat loads simply by multiplying the SWaP metrics by the ratio of Q_{new}/Q_{old} , where Q_{old} is 1,003.9 kW (approximately 1 MW). In the case of the CHEETA aircraft concept, the total takeoff heat load is 12 MW, so the SWaP metrics translate into 708 kg, 2,040 L, and 16.8 kW of pump power. Employing a LT-PEMFC would result in the SWaP growing by a factor that is proportional to the HEX inlet temperature difference (i.e., by a factor of 4.33).

Designing the ram air radiator for cruise conditions will improve SWaP metrics, but the airplane would require battery power for takeoff and climb. Even a notional battery capable of providing 1,000 Wh/kg would result in a battery weight of nearly 3,000 kg to meet the CHEETA concept's energy storage requirement of 3 MW-h to reach cruise conditions. Today's state-of-the-art batteries would drive up the battery weight to nearly 12,000 kg.

4.2.2.2 Microtube HEX Solution

An Al microtube HEX design was developed using the same requirements and operating conditions as the plate-fin unit defined Table 43. The design was analyzed using commercial HEX software (Ref. 278) and published correlations for inline tubes in crossflow (Ref. 279). A comparison of plate-fin and microtube solutions in Table 43 shows significant improvements in all SWaP metrics. For the 12-MW CHEETA concept heat load, the TMS mass drops from 708 to 443 kg, volume is reduced from 2,040 to 1,880 L, and pump power is reduced from 16.8 to 15.0 kW. The microtube solution is 37% lighter and 8% more compact and achieves the same cooling with 11% less pumping power. The TRL of microtube HEXs for ECS applications is 4, but the use of Al limits the manufacturing readiness level (MRL) to 3.

TABLE 43.—COMPARISON OF TMS METRICS FOR PLATE-FIN AND MICROTUBE DESIGNS

TMS SWaP metrics using plate-fin HEX	Flow side	Q , kW	ΔP , kPa	Mass/ Q , kg/kW	Volume/ Q , L/kW	Pump power, kW
	Hot Cold	1,003.9	22.45 0.4047	0.059	1.70	1.40
TMS SWaP metrics using microtube	Flow side	Q , kW	ΔP , kPa	Mass/ Q , kg/kW	Volume/ Q , L/kW	Pump power, kW
	Hot Cold	1,002.1	3.06 0.3478	0.037	0.157	1.25

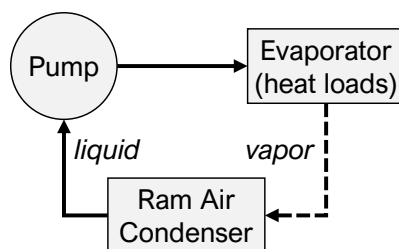


Fig. 287.—Simplified pumped two-phase loop.

The team also studied a P2P cooling solution that would exploit the high HTC and boiling phenomena to maintain the electronics (e.g., fuel cells) at uniform temperature and drastically reduce pumping power. A simplified TMS architecture, as illustrated in Fig. 287, includes a liquid pump, evaporator section, and ram air condenser. In an effort to minimize SWaP, the P2P configuration employs water spray in the inlet duct for the ram air HEX to reduce the air temperature from approximately 50 °C (see Table 42) to 35.5 °C. Water spray is a common practice for both commercial and military aircraft to alleviate hot day conditions. The working fluid for this study is cyclohexane (C_6H_{12}), which was selected based on several factors, including its boiling point and required pressure (i.e., 1 MPa). Fig. 288 provides a T-s diagram to illustrate how the TMS cycle could operate using cyclohexane as a working fluid.

```

graph TD
    Pump((Pump)) -- solid --> Evaporator[Evaporator<br/>(heat loads)]
    Evaporator -.->|vapor| Turbine[Turbine]
    Turbine ==> Gen((Gen))
    Turbine -.->|power| Gen
    Turbine -.->|vapor| RamAir[Ram Air<br/>Condenser]
    RamAir -- solid -->|liquid| Pump
  
```

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4.2.4 Pumped Two-Phase Cooling With Waste Heat Recovery (WHR)

The P2P cooling approach developed here provides dramatic SWaP benefits over conventional pumped liquid cooling systems. However, there is room to achieve further improvements in SWaP by extracting work from the high-enthalpy vapor exiting the evaporator. This WHR is accomplished by incorporating a power turbine downstream of the evaporator. A WHR system provides two significant benefits: (1) it can produce net positive power output, and (2) it reduces the ram air condenser load. A simplified P2P cooling system with WHR is illustrated in Fig. 289. This system utilizes the same working fluid and inlet water spray and is designed to the same 1-MW evaporator (i.e., fuel cell) heat load. Comparison of Fig. 288 and Fig. 290 shows the impact of incorporating WHR into the P2P cooling approach.

As summarized in Table 44, the WHR system generates 158 kW of electrical power based on an evaporator load of 1 MW, which corresponds to an energy conversion efficiency of nearly 16%. Other Boeing studies have shown that this approach can achieve conversion efficiencies exceeding 20%, which can have a profound influence on fuel cell electric aircraft. The total CHEETA heat load at takeoff is 12 MW, which would correspond to 1.92 MW of power generation from the TMS. Expressed differently, a P2P cooling system with WHR would be capable of producing nearly 10% of the total vehicle power. This can enable the fuel cell power system to shrink in size, weight, and cost. The pump power in this configuration is only 0.51 kW, which is more than 30 times less than the conventional pumped liquid cooling solution. The ram air condenser in the P2P with WHR configuration weighs 6.93 kg and occupies a volume of only 28.3 L. The condenser is 34% less voluminous than the simple P2P condenser.

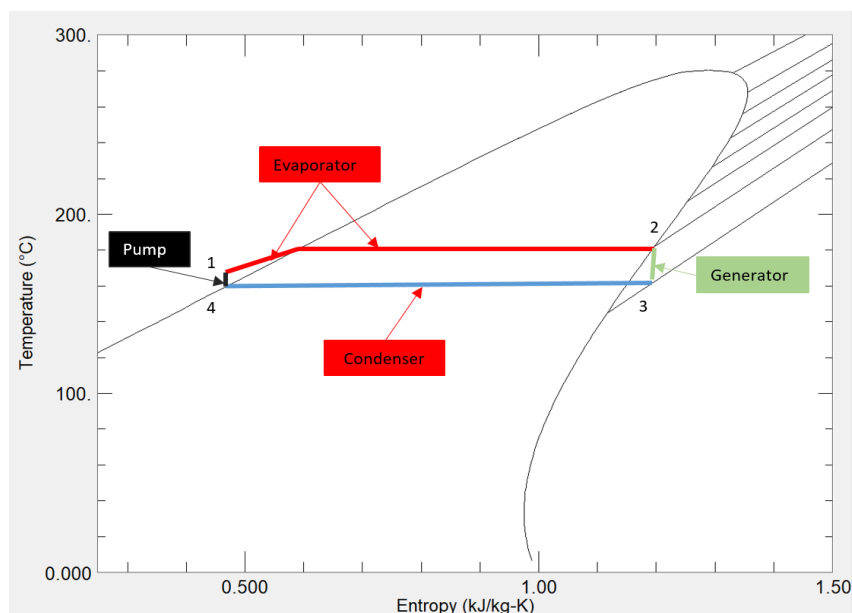


Fig. 290.—T-s diagram for P2P system with WHR.

TABLE 44.—COMPARISON OF P2P WITH AND WITHOUT WHR

TMS configuration	Evaporator duty, kW	Condenser duty, kW	Pump power, kW	Net power out, kW	Percent conversion
P2P	1,000	1,005.3	5.3	-----	-----
PWP with WHR	1,000	842.5	0.5	157.5	15.6%

4.2.5 Lithium Capacitors (LiCs)

This section discusses the technical capability of LiCs for the CHEETA aircraft concept. It is important for the CHEETA fuel-cell-based power system to have a lightweight means to buffer fast transients. Fig. 291 shows the CHEETA electric system schematic. Four fuel cells are connected in parallel on each side of the transfer bus to generate electricity. Four motors are connected to each side of the distribution bus, each passing through a variable speed drive and a DC-AC inverter. The energy storage components (i.e., battery in this figure) are also connected to the distribution bus to provide and absorb transient power during operation. As is well known, fuel cells provide inherently constant or slowly varying power. Transient changing of load current will cause fuel cell degradation and even irreversible damage in a few ways:

- Voltage fluctuations: Fuel cell systems have specific voltage operating ranges, and significant variations outside these ranges can stress the system. This can lead to reduced efficiency and potential damage if the voltage exceeds safe limits (Ref. 280).
- Thermal stress: Rapid changes in current can lead to localized hotspots or temperature variations, which can degrade the fuel cell components. Thermal stress can cause mechanical deformation, cracks, or accelerated aging of materials, ultimately reducing the lifespan of the fuel cell.
- Water management issues: Fuel cells require proper water management for efficient operation. Transient load changes can disrupt the water balance within the fuel cell system. If the water content deviates from the optimal range, it can affect the performance of the PEM and catalyst layers, leading to decreased efficiency and potential damage (Ref. 281).
- Contaminant buildup: Fuel cells are susceptible to the accumulation of contaminants, such as carbon monoxide (CO) or sulfur compounds, which can degrade the catalyst and reduce the fuel cell's performance. Transient load changes can influence the concentration of contaminants in the system, potentially accelerating their buildup and further impacting the fuel cell's operation.
- Mechanical stress: Sudden changes in load current can cause mechanical stress on various components of the fuel cell system, including seals, gaskets, and structural elements. This mechanical stress can lead to leaks, increased resistance, or even structural failure, compromising the overall performance and durability of the fuel cell (Ref. 282).

The CHEETA aircraft power system has energy buffer components to mitigate these effects. Whenever load variation is faster than the fuel cell dynamics, the energy buffer component will either supply or absorb the difference between them, which can happen during load increases and decreases, bus transitions, loss of fuel cell, motors in regenerative mode, and so forth.

4.2.5.1 Storage Component and Technology

This section compares the three selected types of energy components, namely lithium-ion battery (LiB), electric double-layer capacitor (EDLC), and LiC.

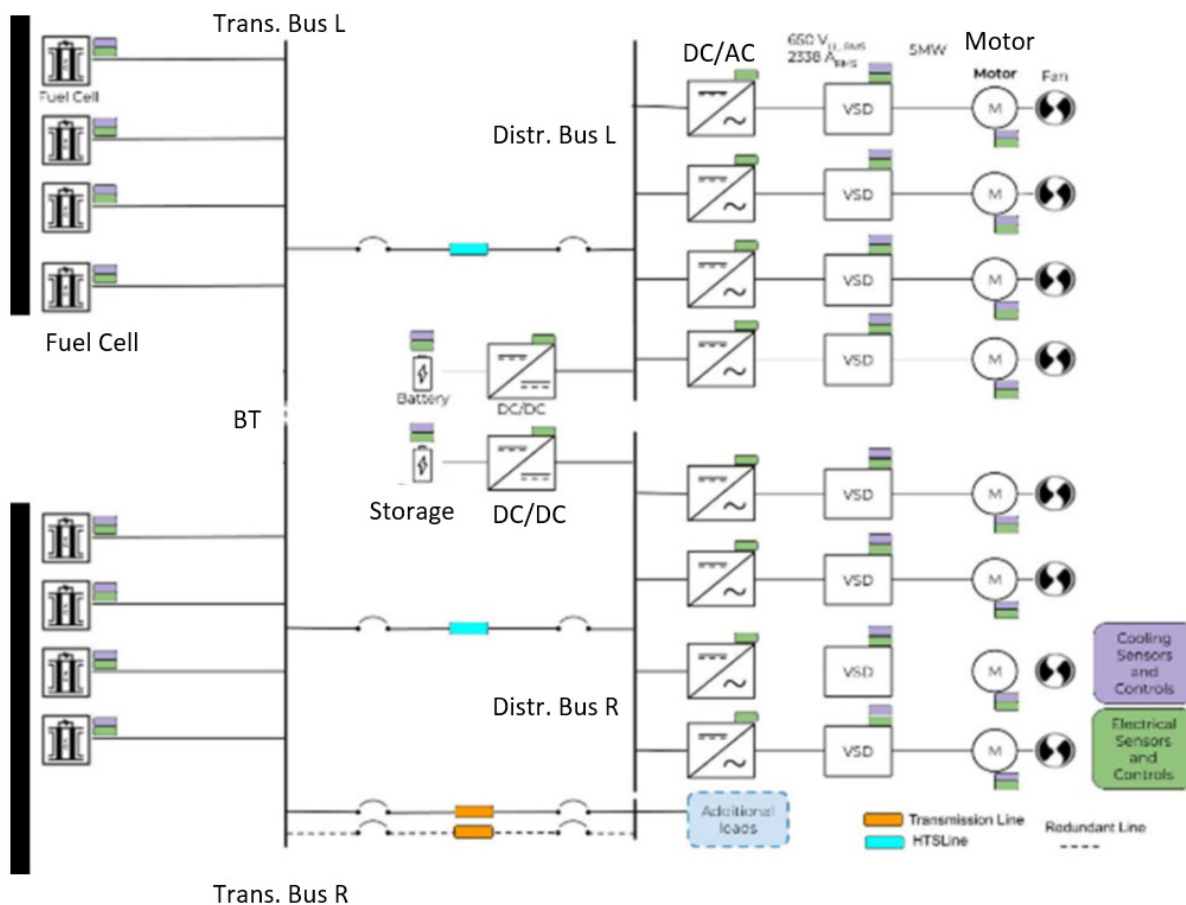


Fig. 291.—CHEETA aircraft concept electrical schematic diagram.

4.2.5.1.1 Electric Double-Layer Capacitor (EDLC)

An EDLC, also known as an ultracapacitor or supercapacitor, is an energy storage device that uses electrostatic double-layer capacitance and sometimes pseudocapacitance to store energy. EDLCs have a very high specific power and a long cycle life, but low specific energy or energy density due to energy storage mechanism and the material itself, and linear discharge curve yields high residual energy at minimum required voltage.

4.2.5.1.2 Lithium-Ion Battery (LiB)

A LiB is a rechargeable battery that uses lithium ions as the main constituent of its electrochemical system. LiBs have a high specific energy but issues with degraded performance or even malfunctioning at low temperatures, thermal runaway safety concerns at high temperatures, and limited battery life span.

4.2.5.1.3 Lithium Capacitor (LiC)

LiCs combine characteristics of battery and capacitor and yield a higher energy density than EDLC cells, with longer life and higher power capability than battery cells. However, there are a limited number of products on the market. LiCs generally have a wider operational temperature range than LiBs. For example, some COTS LiCs can operate down to -25°C , while other LiCs can operate up to 105°C , suitable for different applications. LiCs are rated for 100 to 500 K charging/discharging cycles, which

yields a longer maintenance-free service life. Like the EDLC, the voltage of the LiC varies linearly according to its state of charge. In applications that require a stable operating voltage, a buck/boost converter will be required. Unlike EDLCs, LiCs are generally limited to a minimum voltage of 2.2 V to prevent damage, whereas EDLCs can be discharged to zero volts. LiCs possess a long cycle life and high energy density, and their discharge time can be extended by several minutes instead of seconds, which may be a significant difference for power quality solutions. Examples of LiC manufacturers are Eaton, Taiyo Yuden, Tecate Group, and VinaTech, among others.

4.2.5.1.4 Comparison of Storage Technologies

Fig. 292 compares various storage technology capabilities that are applicable to the CHEETA aircraft concept. Fig. 293 (Ref. 283) shows the storage mechanism and chemical structure of different technologies. The LiC has an asymmetric configuration, as it uses two types of electrodes. Anions are adsorbed onto the positive electrode surface or detached from the positive electrode surface, whereas lithium ions are embedded in or detached from the negative electrode host.

4.2.5.2 Simulation Modeling and Study

This section describes the modeling and simulation developed in MATLAB®/Simulink® (The MathWorks, Inc.) for power transient studies of PEMFC and LiC power systems. It can be used to study power sharing between a fuel cell and an energy buffer component, power transient migration, power flow control, sizing of components, and more. The CHEETA aircraft power system is further simplified and scaled down into a fuel cell system, LiC, and load without losing the essence of the study purposes, as shown in Fig. 294.

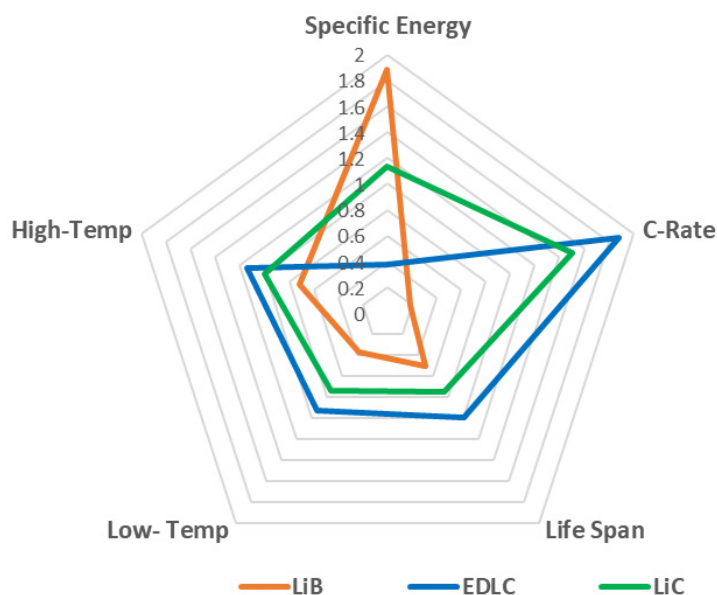


Fig. 292.—Comparison of storage technologies.

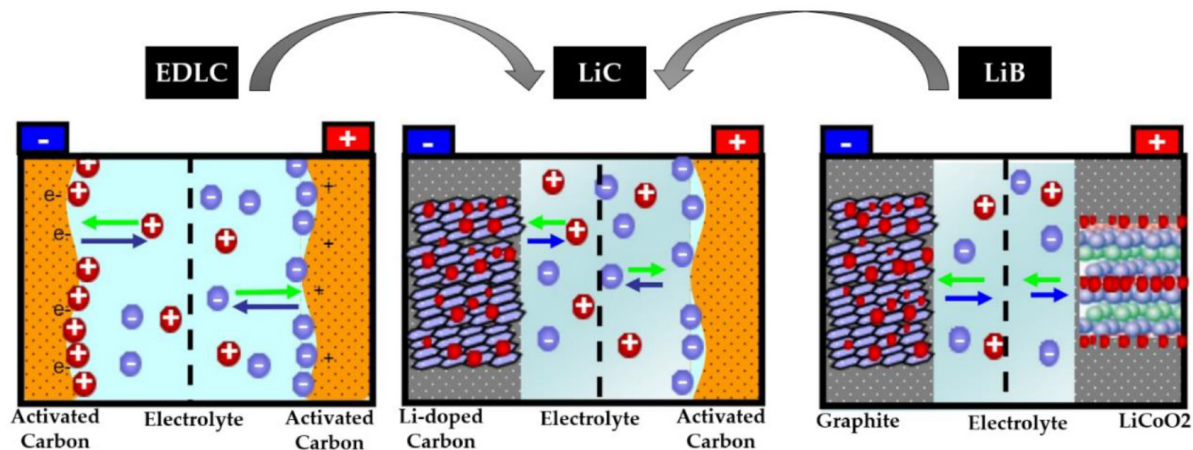


Fig. 293.—Storage mechanism and chemical structure of LiBs, LiCs, and EDLCs.

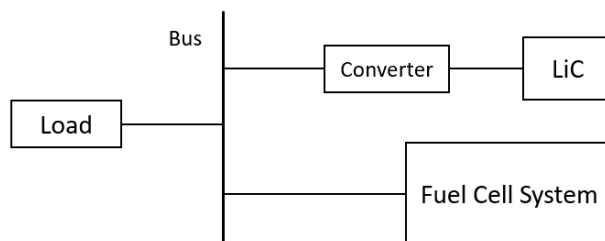


Fig. 294.—Simplified fuel cell/LiC power systems.

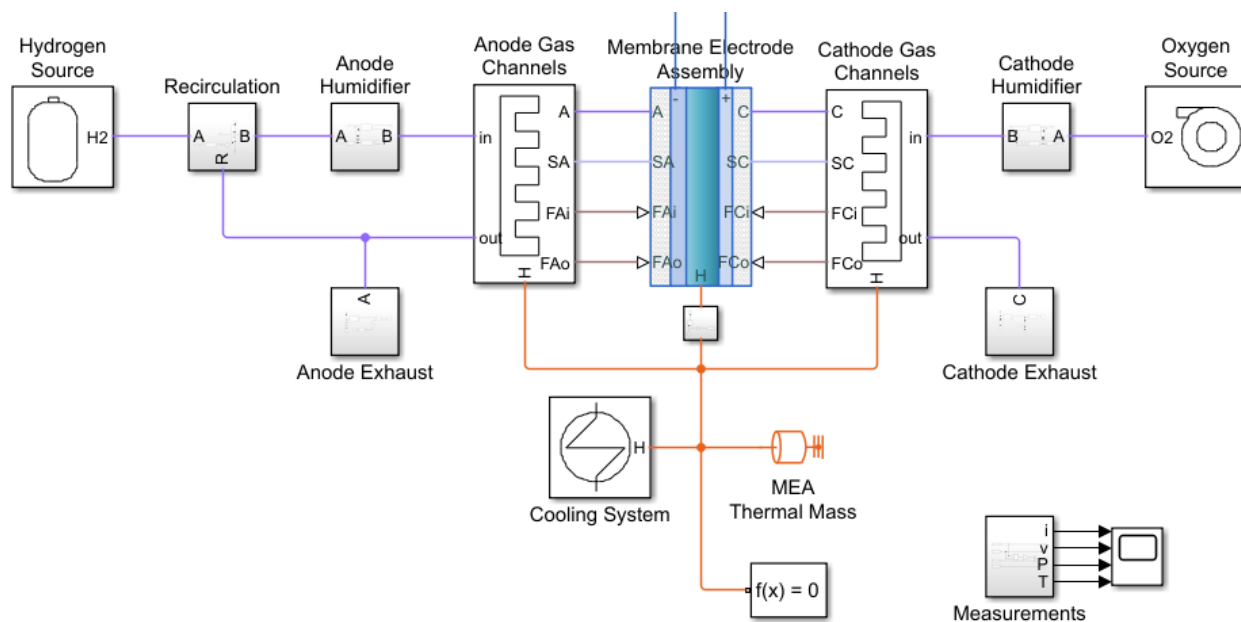


Fig. 295.—Simplified fuel cell power systems.

4.2.5.2.1 Modeling of Fuel Cell System

Fig. 295 shows the modeling of a PEMFC system. The two moist air networks represent different gas mixtures. The anode network consists of N_2 , H_2O vapor, and H_2 , representing the fuel. The H_2 is stored in the fuel tank at 70 MPa. A pressure-reducing valve releases H_2 to the fuel cell stack at around 0.16 MPa.

Unconsumed H_2 is recirculated back to the stack. The cathode network consists of N_2 , H_2O vapor, and O_2 , representing air from the environment. A compressor brings air to the fuel cell stack at a controlled rate to ensure that the fuel cell is not starved of O_2 . A back-pressure relief valve maintains a pressure of around 0.16 MPa in the stack and vents the exhaust to the environment. The temperature and relative humidity in the fuel cell stack must be maintained at an optimal level to ensure efficient operation under various loading conditions. Higher temperatures increase thermal efficiency but reduce relative humidity, which causes higher membrane resistance. Therefore, in this model, the fuel cell stack temperature is kept at 80 °C. The cooling system circulates coolant between the cells to absorb heat and rejects it to the environment via the radiator. The humidifiers saturate the gas with water vapor to keep the membrane hydrated and minimize electrical resistance. The model is available in Simulink® using Simscape block (Refs. 284 to 285).

4.2.5.2.2 Modeling of LiC

Many electrical modeling approaches have been developed (Ref. 283). One approach to modeling involves the integration of physics and chemistry interfaces, known as the electrochemical model. This sophisticated method considers the intricate interactions between these two domains. While electrochemical models provide highly accurate results, they come with a drawback of extensive computational time. This is primarily because they capture the intricate reaction processes using partial differential equations. Consequently, their prolonged computation time limits their practical utility in areas such as energy management and real-time control, where efficient decision-making is crucial. Another approach is to use resistor capacitor (RC) networks to model the electrical behaviors of LICs. There are two key factors to improve the modeling precision of LiCs: optimizing the circuit configuration and increasing the number of components. In this study, the RC modeling method is adopted to fulfill the purpose of the current study of energy management and rapid simulation time. Fig. 296 shows the model of the LiC. Capacitors C_1 , C_2 , and C_3 have fixed capacitances. The capacitance of capacitor C_v depends on the voltage across it. Resistors R_1 , R_2 , R_3 and $R_{selfdischarge}$ have fixed resistances.

4.2.5.2.3 Modeling of Converter With Control

The power converter regulates the voltage to match between the load side and the source (supply) side. To maintain a balance between input power, output power, and losses, the power converter draws the necessary power from the supply side. However, in certain scenarios, the converter can also support regenerative power flow, allowing power to flow back from the load to the supply, thereby supporting bidirectional operations. The converter model will automatically regulate the voltages of two ports and has a user parameter allowing input of a fixed converter efficiency value.

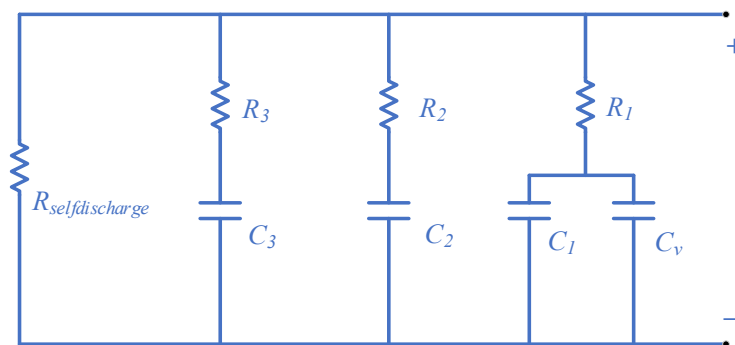


Fig. 296.—Equivalent circuit for a single-cell LiC.

4.2.5.2.4 Simulation and Results

Fig. 297 shows the simulation schematic of the fuel cell/LiC power system in MATLAB®/Simulink®, which represents the system described in Fig. 294. The fuel cell connects to the load directly, and the LiC connects to the load through a DC-DC converter. This schematic will represent a simplified system for the studies of dynamic behavior on the CHEETA aircraft concept. A single fuel cell stack is 2.5 MW, whereas in this simulation the fuel cell is scaled down and set as 100 kW rated electrical power, with the stack area 280 cm² and the limiting current density 1.4 A/cm². The fuel cell maximum current changing rate (i.e., the current slew rate) is set as 0.163 A/s, which is assumed to be the same value for both fuel cell power rising and closing down in this study. As is well known, the maximum current changing rate depends on various factors such as the design, electrochemical processes involved, cell size and geometry, and operating conditions of the fuel cell, and is worthy of further investigation.

The LiC stack has 60 cells in series and 10 strings in parallel, and each LiC cell has a rated capacity of 2600 F. The initial voltage of each LiC cell is set as 4V. The equivalent capacitor of the entire stack is 433 F, and stack voltage is 240 V. The converter efficiency is set as a fixed value of 95%. The control algorithm will also monitor the LiC stack voltage; if the LiC voltage is below 240 V, the controller will direct the fuel cell to charge it as long as the fuel cell has extra current bandwidth beyond the load demand. Fig. 298 shows the pulse current load profile. The period is 1500 s, with pulse width equaling 50% of the period, and the amplitude is 268 A, corresponding to a fuel cell output power of 100 kW. The instantaneous step-up and step-down of load power demands are challenges to a fuel cell system.

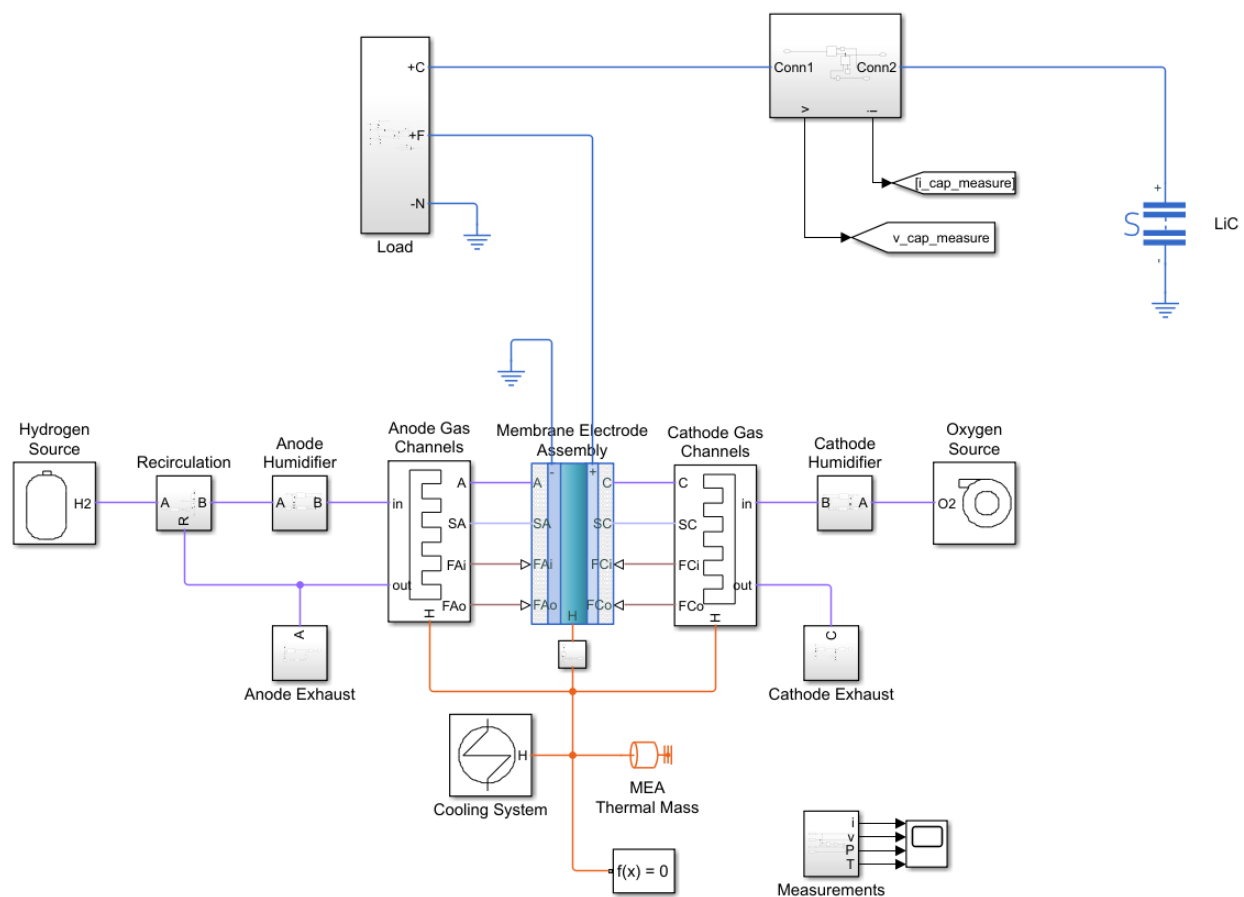


Fig. 297.—Overall schematic of fuel cell/LiC power system.

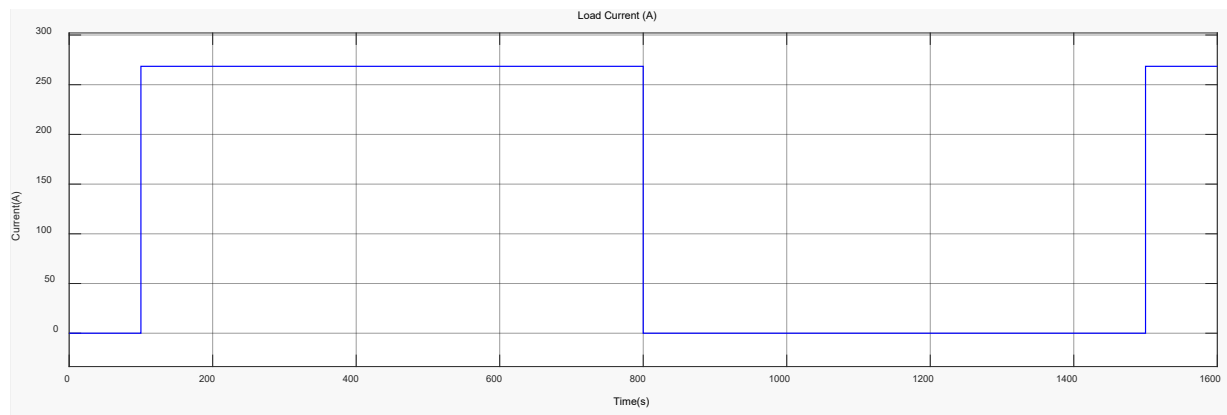


Fig. 298.—Pulse load profile.

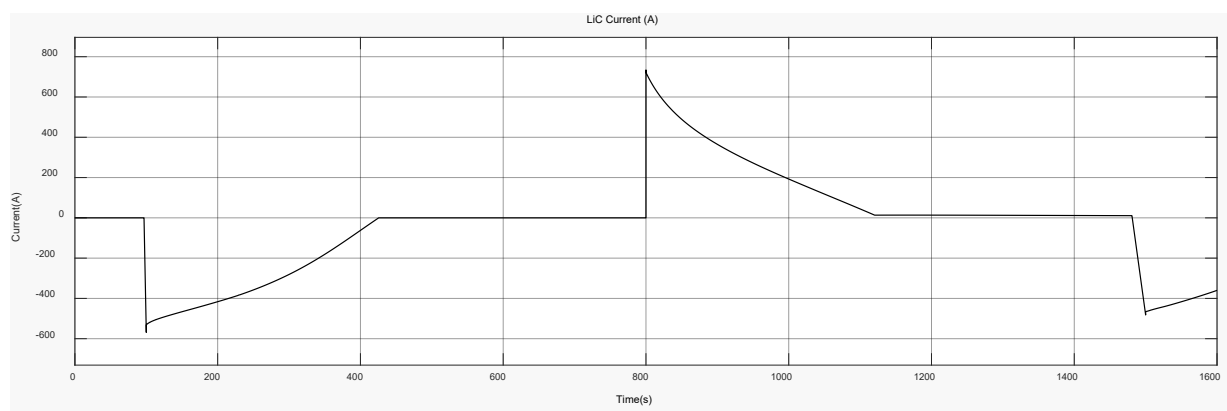


Fig. 299.—LiC current profile during simulation.

Fig. 299 shows the LiC current profile during the simulation. From 0 to 100 s, the current remains at zero, as the load is also zero during this period. Subsequently, from 100 to 425 s, the LiC starts a discharge process with the current instantly rising to 500 A and gradually declining to zero. This discharge phase allows the fuel cell system to gradually increase its output power. Between 425 and 800 s, although the load current is high, the LiC current remains at 0 A since the fuel cell caters to the entire power requirement of the load. From 800 to 1120 s, the load current becomes zero, and the LiC begins to charge as the fuel cell current gradually decreases. Finally, from 1,120 to 1,500 s, the LiC undergoes charging with a small current value. This occurs because the control algorithm detects that the LiC voltage is lower than 240 V and initiates fuel cell charging to replenish the LiC.

Fig. 300 depicts fuel cell current during the simulation. Between 100 and 425 s, the load current experiences a step increase and stabilizes at 268 A, while the fuel cell current gradually rises from 0 A to reach a final value of 268 A. The LiC compensates for the difference between the load and fuel cell currents, as depicted in Fig. 299. Throughout the period from 425 to 800 s, the load current remains high, and the fuel cell supplies the entire load current. From 800 to 1120 s, the load current becomes zero, and the fuel cell current gradually decreases from 268 to 7 A. The energy generated during this phase is transferred to the LiC for charging. Subsequently, between 1120 and 1500 s, the fuel cell current is maintained at a low value of approximately 7 A. This current level is regulated by the control algorithm to charge the LiC and ensure that the LiC voltage remains at 240 V.

Fig. 301 shows LiC voltage during the simulation. From 100 to 425 s, the LiC undergoes a discharge process, resulting in a significant drop in voltage from 240 to 135 V. The voltage of 135 V is close to the

minimum voltage of 132 V (calculated as 60 multiplied by 2.2). This indicates that the LiC is nearly depleted by 425 s. From 425 to 800 s, the LiC voltage shows a recovery of approximately 10 V. This recovery is not because of the fuel cell charging the LiC, as all of the current from the fuel cell flows to the load (refer to Fig. 299 and Fig. 300). Instead, this recovery is attributed to the structural characteristics of the LiC material, where a portion of the stored charge deep within gradually diffuses toward the top layer (refer to the LiC model in Fig. 296). Between 800 and 1120 s, the load is inactive, and the fuel cell recharges the LiC, bringing its voltage to a final value of approximately 238 V. Subsequently, from 1,120 to 1500 s, the LiC undergoes continuous charging by the fuel cell with a small current until the voltage reaches 240 V.

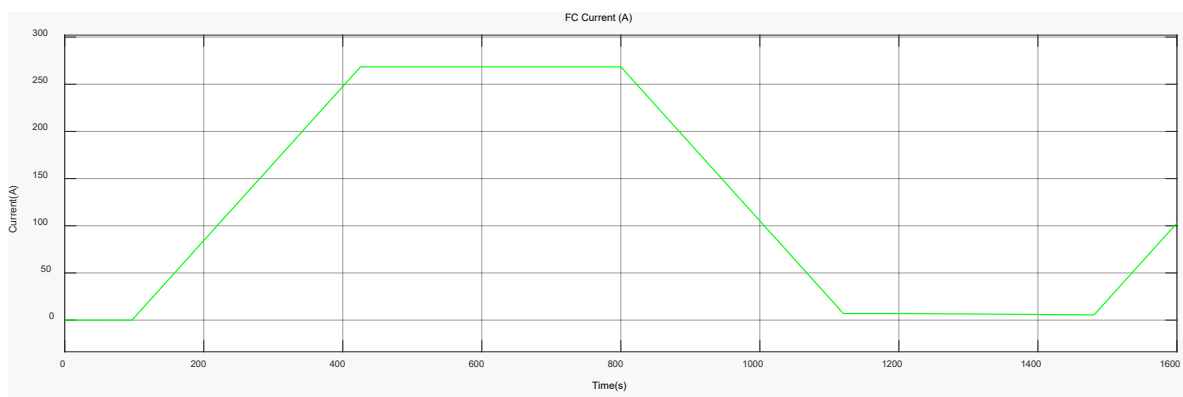


Fig. 300.—Fuel cell current during simulation.

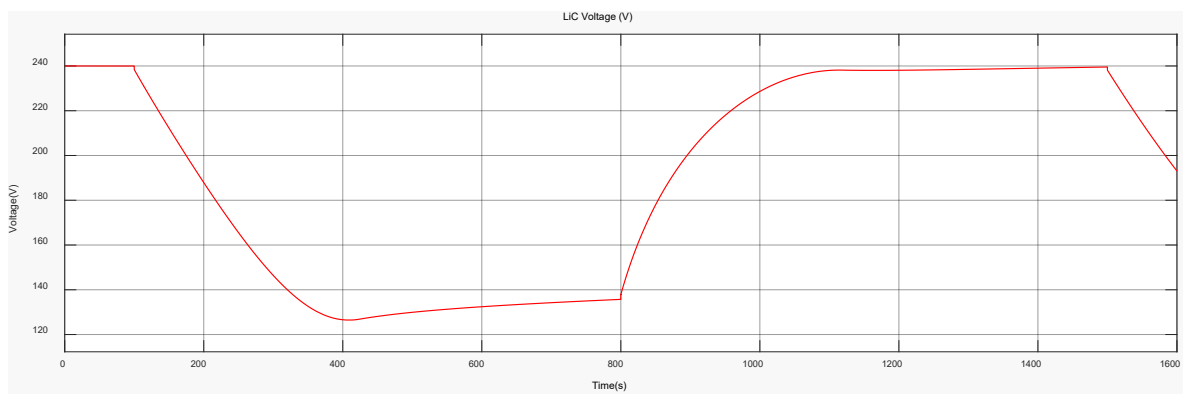


Fig. 301.—LiC voltage during simulation.

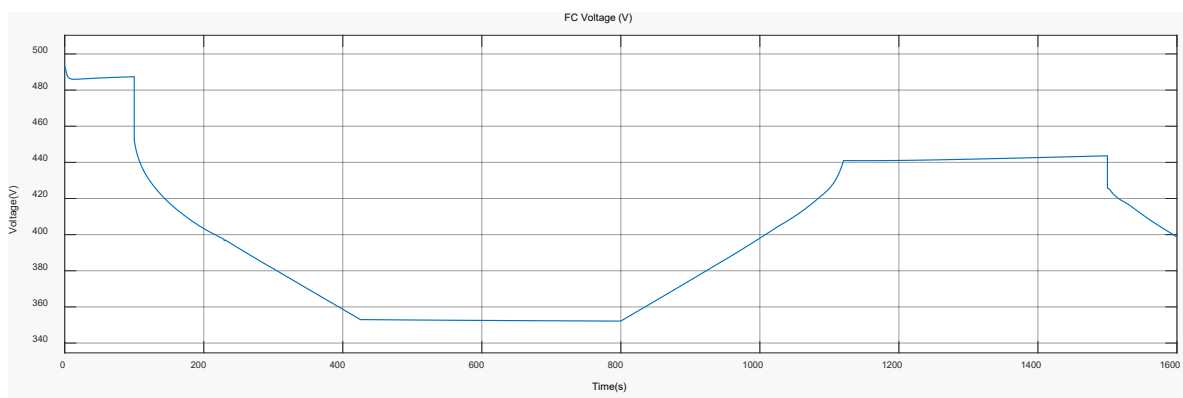


Fig. 302.—Fuel cell stack voltage during simulation.

Fig. 302 shows fuel cell stack voltage during the simulation. The voltage drops to about 350 during high power demands and recovers to about 440 when charging the LiC.

Fig. 303 illustrates the electrical power generated by the fuel cell stack alongside the power consumed by the cathode air compressor and the coolant pump, necessary for maintaining stable and efficient operation of the system. Consequently, the net power output of the system is slightly lower than the power generated by the stack. It is important to note that this model assumes the use of an isentropic compressor. Considering the efficiency of the compressor would further decrease the net power gain by a couple of additional percentage points. Furthermore, the figure displays the excess heat generated by the fuel cell stack, which needs to be dissipated by the cooling system.

Fig. 304 shows the thermal efficiency curves of the fuel cell based on the HHV (higher heating value) and LHV (lower heating value) of H_2 , as well as the reactant utilization fraction. The thermal efficiency represents the proportion of the energy contained in the H_2 fuel that the fuel cell has converted into useful electricity. For a PEMFC, the theoretical maximum efficiency is 83%. However, due to internal losses, the actual efficiency typically reaches around 60%. The reactant utilization fraction refers to the portion of the reactants, namely H_2 and O_2 , flowing into the fuel cell stack that has been consumed by the fuel cell. Increasing the utilization fraction enhances the utilization of gases passing through the fuel cell. However, it also results in a decrease in the concentration of reactants, leading to a reduction in the voltage produced. Unconsumed O_2 is released into the environment, as it is not reusable, while unconsumed H_2 is redirected back to the anode to prevent wastage. In practice, H_2 is periodically purged to eliminate any contaminants that might have accumulated.

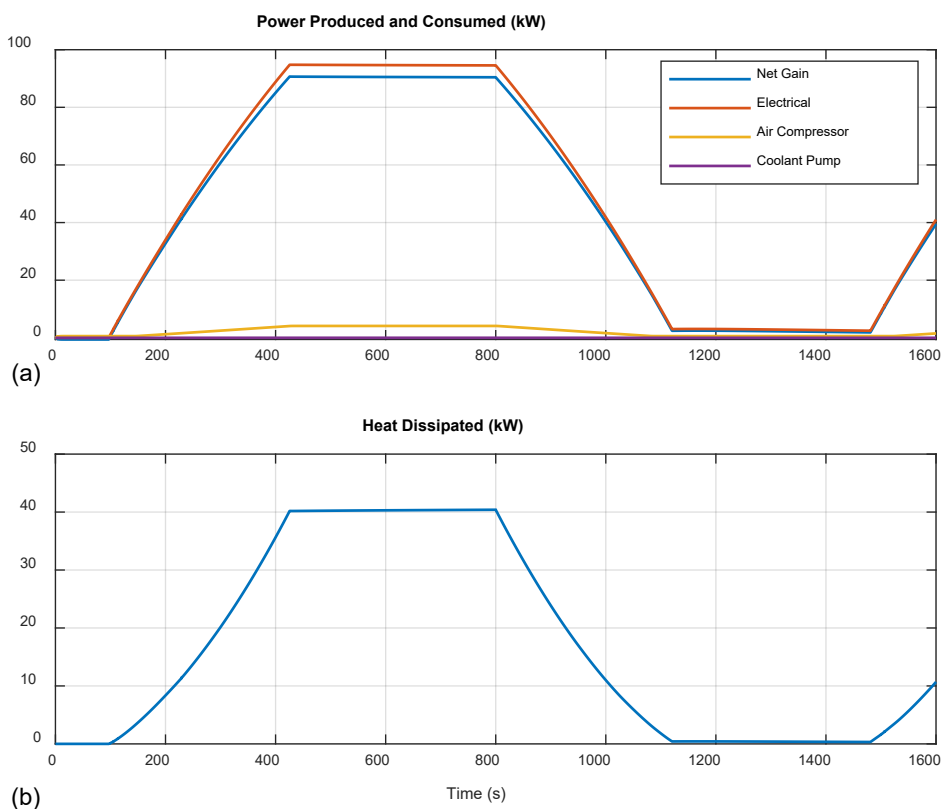


Fig. 303.—Fuel cell system power production (a) and consumption and heat generation (b).

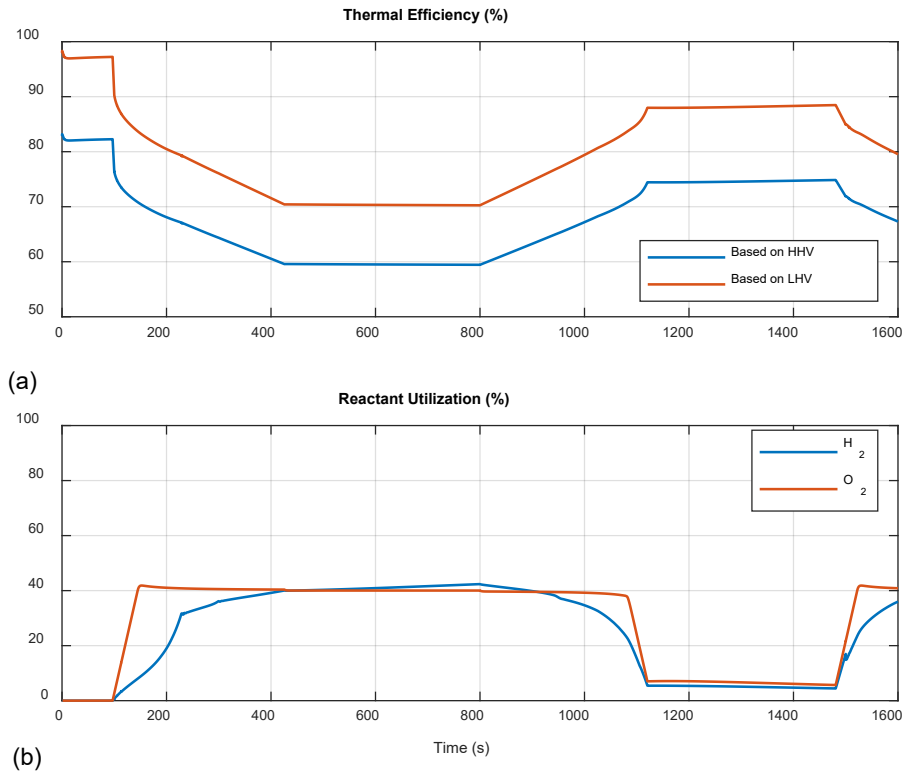


Fig. 304.—Fuel cell thermal efficiency (a) and reactant utilization (b).

Fig. 305 presents temperature readings from different locations within the fuel cell system and coolant mass flow rate. The fuel cell stack temperature is regulated to a maximum limit of 80 °C through the cooling system. Fuel entering the anode is heated by the recirculated flow, while air entering the cathode is warmed by the compressor. Maintaining an optimal temperature is crucial for proper functioning of the fuel cell because higher temperatures can reduce relative humidity, leading to an increase in membrane resistance. Hence, controlling temperature within the desired range is essential. In this model, the cooling system operates by adjusting coolant flow rate through a simple control mechanism. Fig. 305 shows the temperature of the coolant after it has absorbed heat from the fuel cell stack and after it has released that heat through the radiator.

4.2.6 Battery Cell Selection and Dynamic Characterization Testing

While the fuel cell systems described previously provide the majority of the power for propulsion and other aircraft-associated systems, inclusion of batteries can be beneficial to ensure that rapid changes in power demand can be met during off-nominal operating points. Given the continued need to provide a lightweight power and energy systems architecture, advanced high-power battery systems were considered.

To provide a forecast for the potential specific energy of high-power (100 C) batteries relevant to the CHEETA program in 2050, we analyze the historical data of various high-energy (1 C) battery technologies from the 1900s to the present and project the industry's potential advancements. Additionally, we will estimate the specific energy of high-power batteries in 2050 by employing the current high-energy versus high-power battery specific energy ratio, which will support the fuel cell/battery hybrid system modeling in the CHEETA program. Given the scarcity of historical high-power battery specific energy data, primarily due to technical and application limitations, we rely on high-energy battery data for our advancement forecast.

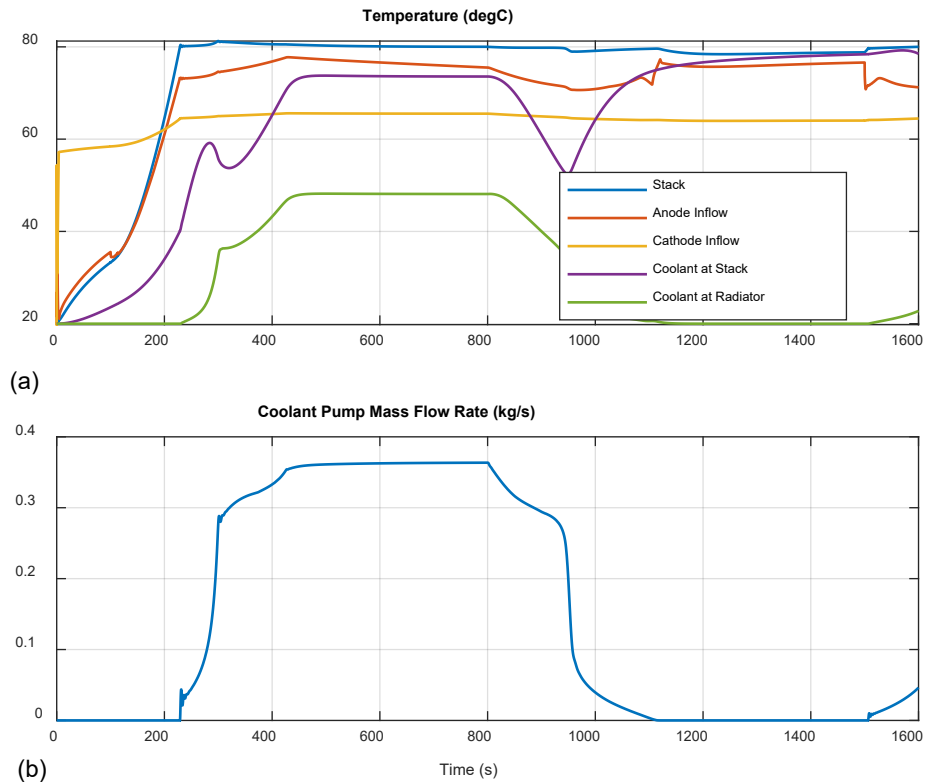


Fig. 305.—Fuel cell system temperatures (a) and coolant mass flow rate (b).

To comprehend the evolution of battery specific energy, we begin by examining the historical data of the pre-LiB era, including lead-acid batteries, Ni-Cd batteries, and nickel-metal hydride (NiMH) batteries from 1900 to the present. We then explored the specific energy trends observed in commercial high-energy LiBs since their introduction in 1991. Fig. 306 illustrates the collected historical data of lead-acid batteries, Ni-Cd batteries, and NiMH batteries, showcasing a yearly specific energy improvement of approximately 1.1%. Although these chemistries have demonstrated gradual advancements over time, they have not achieved significant specific energy improvements when compared to LiBs.

LiBs have experienced remarkable progress, with an impressive annual specific energy improvement of approximately 4.8%. This substantial growth can be attributed to various factors, including the significant demand from the consumer electronics market, government incentives, and advancements in material characterization, computer modeling, and related technologies. By advancing material characterization techniques to incorporate in situ and operando methods, and by further integrating computer modeling with machine learning and automated experimentation, the potential for achieving even higher annual specific energy improvements becomes attainable. These advancements in characterization, computational, and experimentation methodologies serve as the underpinning for accelerated progress in battery technology into the future.

Government incentives also play a critical role in driving battery technology advancements. For instance, the launch of the U.S. DOE's Battery500 program in 2017 has set ambitious goals for the field. The program aims to achieve a cell-level specific energy of 500 Wh/kg by 2030. Notably, within just three years, demonstrations of cell-level specific energy approaching and exceeding 400 Wh/kg have already been accomplished. Moreover, in 2023, a commercial cell with an energy density of 450 Wh/kg was successfully realized, further highlighting the rapid progress in battery technology.

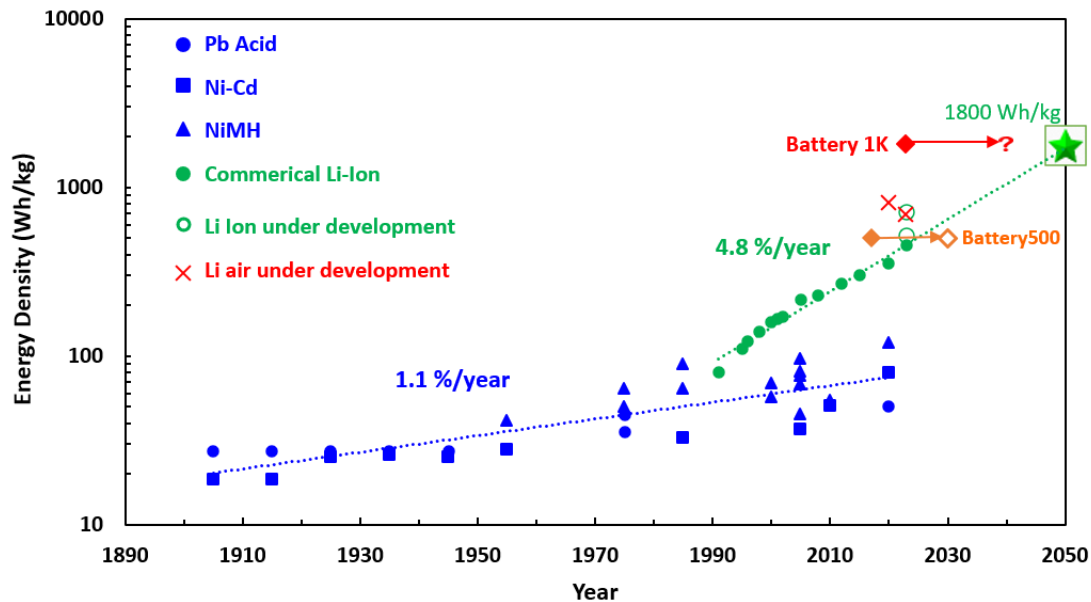


Fig. 306.—Past, current, and projected battery energy densities.

Assuming that the current annual specific energy improvement rate of 4.8% continues, it is reasonable to estimate that batteries will reach a specific energy of approximately 1800 Wh/kg by 2050. While this specific energy surpasses the theoretical limit of LiBs, other promising battery chemistries, such as magnesium-air (theoretically 2800 Wh/kg), Al-air (theoretically 4300 Wh/kg), and lithium-air (theoretically 5200 Wh/kg), far exceed the specific energy target.

Fig. 306 illustrates energy density improvement trends since the 1900s for various battery technologies, including lead-acid, Ni-Cd, NiMH, and LiB. For LiB, only the highest energy density value of commercial cells that year is shown. Representative U.S. Government programs and battery technologies under development are also shown.

These chemistries still face significant fundamental challenges that must be overcome to unlock their full potential, yet they serve as a benchmark for future battery technologies to strive toward. Moreover, the expanding battery market, driven by electric vehicles, grid energy storage, and potentially electrified aviation, along with the already substantial consumer electronics market, creates a conducive environment for driving technological advancements that enhance battery specific energy.

Taking advantage of this momentum, the U.S. Government is formulating the Battery1K program, which aims to achieve a specific energy of 1,000 Wh/kg at the pack level at the battery EOL. Assuming a typical cell-to-pack ratio of 70% and an EOL capacity to fresh cell ratio of 80%, this corresponds to a cell-level specific energy of 1,800 Wh/kg. The timeline for achieving this goal is yet to be defined, but it is generally believed to be several years before the 2050 timeframe to support aviation's net-zero 2050 goals.

To project the specific energy of high-power (100 C) batteries for 2050, we utilize the ratio between high-power and high-energy battery specific energy. Drawing from currently available data on commercial cells, this ratio stands at approximately 0.56. By extrapolating this ratio, we can estimate that the specific energy of high-power (100 C) batteries in 2050 will be approximately 1,000 Wh/kg.

In conclusion, our analysis of historical data and observed trends leads us to forecast a significant rise in the specific energy of high-power (100 C) batteries by 2050. The projected specific energy of 1,000 Wh/kg at the cell level, considering the advancements achieved through programs like Battery500, Battery 1K, and anticipated future technology and market developments, establishes a noteworthy milestone for the industry to aspire to.

TABLE 45.—SUMMARY OF HIGH-POWER CELL CHARACTERISTICS

Characteristic	Condition	Fullymax®				Samsung SDI	Toshiba
Nominal capacity, Ah	0.2-C discharge	1.3	4.93	3.3	3.2	5	10
Energy density, Wh/kg	0.2-C discharge	153	143	155	25.8	250	47
Power (C-rate)	Continuous max. C-rate	100	100	70	10	9	75
Temperature	Reference	10 to 45 °C	10 to 45 °C	10 to 50 °C	10 to 50 °C	Max. 80 °C	25 °C, 10 s
Cycle life	25 °C, discharge capacity ≥80%	≥500	~500	≥300	≥500	<250	~15,000

4.2.6.1.1 Battery Testing

Further market research, technology benchmarking, and supplier engagements are the key focus areas for battery cell development/selection and sourcing as per the requirements of any application. After understanding the CHEETA program requirements and system architecture, we engaged with our primary cell supplier Fullymax® (Shenzhen Fully Battery Co., Ltd.) to source the candidate cell with specifications of 1.3 Ah, 100 C continuous discharge rate in the available timeline and started performing characterization testing. In addition to Fullymax® 1.3-Ah cells, the following high-power cells have been identified (Table 45) that can be available for near-future procurement to do characteristic testing.

4.2.6.2 Dynamic Characterization Testing

It is important to do dynamic characterization of the newly developed cell to verify cell performance against published specifications using defined test conditions. It helps in the following aspects of candidate cell selection:

- Understand charge and discharge characteristics under various C-rate and temperature conditions.
- Analyze the thermal behavior of the cells to understand the temperature related performance limitation.
- Collaborate with the cell developer on the required characteristics.
- Evaluate cycle life and energy density using mission profiles.
- Provide inputs to the cell, module, and battery pack level modeling.
- Understand or derive requirement for TMS.
- Understand thermal runaway mitigation strategy.
- Provide inputs to system safety.

4.2.6.3 Test Setup and Results

The battery test setup is depicted in Fig. 307. The battery cell test results are summarized in Fig. 308, with discharge profiles and capacity profiles shown in Fig. 309 and Fig. 310, respectively. It should be noted that the received cell samples are the low-cost version of initially developed 100-C continuous discharge cells, hence the very low-capacity measurement during testing. Future work is recommended to source the high-performance version, which can meet 100-C continuous discharge specifications.

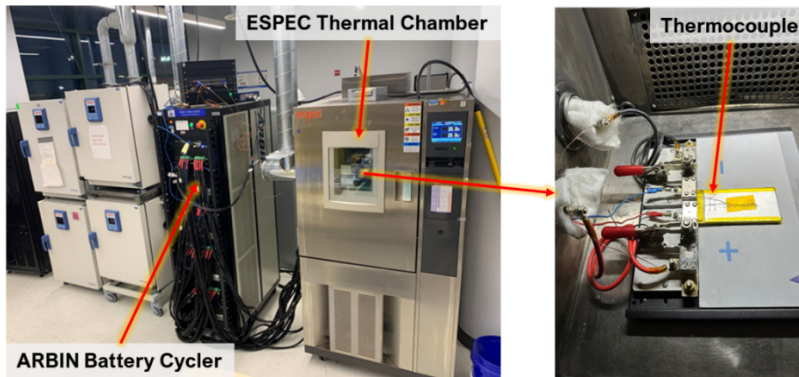


Fig. 307.—Battery cycler and temperature chamber setup.

Manufacturer: Fullymax					
Model: FBC773462					
Initial performance	Description	Specification	Test Result		Notes
Size	Max (T x W x L)	7.7x34.5x64.5	7.24x32.89x57.54	7.26x32.86x57.32	Vernier Calipers
Mass, g	Max	32.60	32.65	32.47	Scale
Nominal Capacity (1C), Ah	Ch: 1C to 4.2V with C/50, Dch: C/5 to 3.0V at 25°C	1.3	1.29	1.30	Min capacity = 1.27 Ah, Typical capacity = 1.33 Ah
Nominal voltage, V		3.7	3.84	3.84	
Energy, Wh		4.8	4.98	4.98	
Standard charge current, A		1.3	-	-	1C
Max charge current, A		2.6	-	-	2C (Ref temp: 10~45°C)
Standard discharge current, A		1.3	-	-	
Max continuous discharge current, A		130	0.14	0.46	100C (Ref temp: 10~45°C)
Max Peak discharge current, A		195	Failed at 90% SOC	Failed at 90% SOC	150C (< 3 sec)
Specific energy, Wh/kg		148	152	153	
Energy density, Wh/L		281	-	-	
Operating temperature, Charge		10 to 45°C	-	-	
Operating temperature, Discharge		-20 to 60°C	-	-	
ACIR as received, mΩ	1kHz with Hioki at 25°C	< 3.3	2.46	2.60	
DCIR, mΩ	Pulse discharge at 100C for 3 sec at 10% SoC, 50% and 90% SoC.	-	Failed at 90% SOC under 100C	Failed at 90% SOC under 100C	SoC adjusted during discharge at C/5
Performance					
Crate discharge capacity, % vs C/5	Ch: 1C to 4.2V with C/50, Dch: C/5, 1C, 10C, 50C, 100C at 25°C,	> 80% at 100C	100C failed	100C failed	vs. Minimum Capacity
Temperature discharge capacity	Ch: 1C to 4.2V with C/50, Dch: 1C at -20, 25 and 60°C, rest: 1 hour	60%, 100%, 150%	On testing	On testing	
Cycle life 1, cycle number until < 80 %	Ch: 1C to 4.2V with C/50, Dch: 1C to 3.0V at 25°C, rest: 10 min	500 cycles			DCIR measurement every 100th
Cycle life 2, cycle number until < 80 %	Ch: 1C to 4.2V with C/50, Dch: 1C to 3.0V at 25°C, rest: 30 min				DCIR measurement every 100th
Cycle life 3, cycle number until < 80 %	Ch: 1C to 4.2V with C/50, Dch: 1C to 3.0V at 45°C, rest: 10 min				DCIR measurement every 100th
Calendar life at 25°C, %	Retention after 1 month at 4.2V	85%			
Calendar life at 25°C, %	Recovery after 1 month at 4.2V	90%			
Calendar life at 25°C, %	Recovery after 1 year at 50% SOC	85%			

Fig. 308.—Summary of test results for Fullymax® cells.

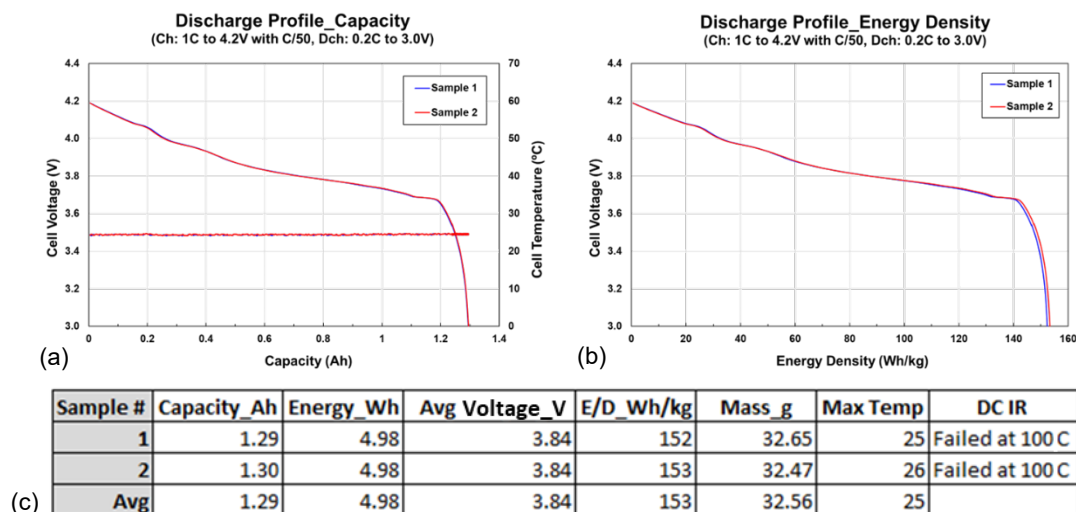


Fig. 309.—Cell discharge profiles. (a) Capacity. (b) Energy density. (c) Table.

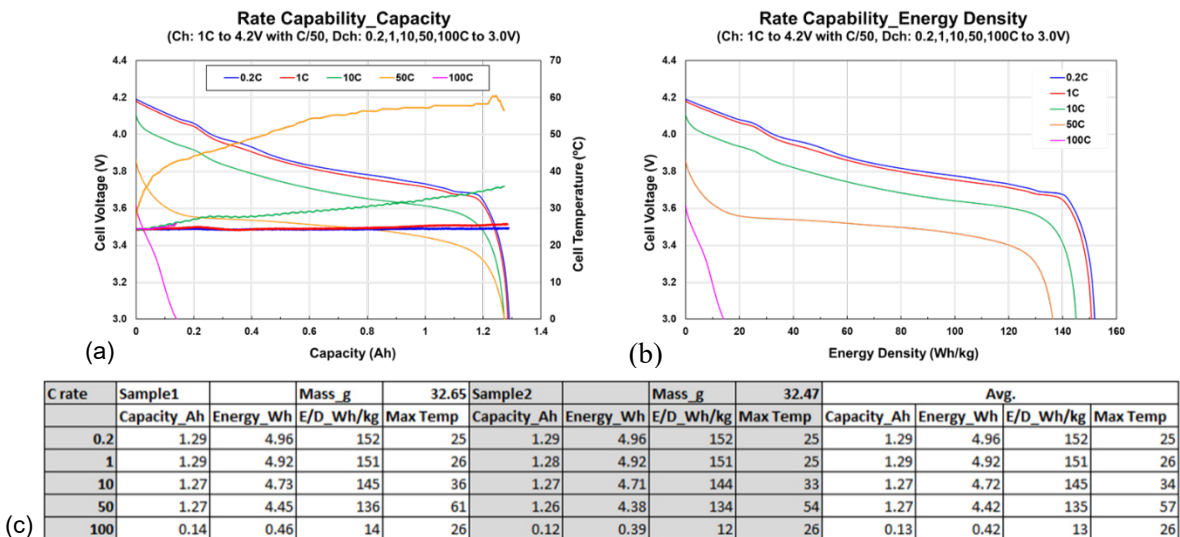


Fig. 310.—Initial capacity summary. (a) Capacity. (b) Energy density. (c) Table.

4.2.6.4 Electrical Power System: Conclusions

Test results, and collaboration with Fullymax[®], conclude that the 1.3-Ah cell and other higher power cells have the potential to deliver 100-C continuous power with 153-Wh/kg energy density. However, we need to continue monitoring the industry trend and aggressively engage and partner with existing and other potential cell developers to push for new cell development targeting up to 1,000-Wh/kg energy density by 2050.

5.0 System Integration

A key component of the CHEETA program was determining how the integration of the various innovative technologies studied can be performed in a way that best leverages the collective strengths and overcomes inherent challenges of these technologies at a system level. Studies were performed in the areas of aircraft conceptual design and sizing, TMS integration, and trade studies of alternate H₂ power and propulsion architectures.

5.1 Aircraft Conceptual Design

For this program, a series of conventional design approaches were combined with a model-based approach to resolving the operating characteristics of the novel power and energy systems, as well as careful engineering analysis, to explore the aircraft design space. This approach allowed for the selection of key aircraft conceptual features and anticipated aircraft performance characteristics.

5.1.1 Fuel Cell Aircraft Sizing

While modern transport-class aircraft utilize high-bypass-ratio turbofan engines as a means for power and propulsion, the CHEETA vision vehicle aircraft is defined with a fuel cell power system and electric ducted fan propulsion architecture. By utilizing this configuration, the standard methods used for sizing airframe and propulsion systems are not readily applicable. For this reason, a variety of models were developed to explore the conceptual design landscape of the CHEETA vision vehicle aircraft.

With subsonic transports serving as the dominant contributor to the fuel consumption and carbon footprint of global aviation, the need for environmentally responsible transportation has been met with

increased research activity in the field of aircraft propulsion electrification across industry, government, and academic organizations. Most recently, a number of studies have revealed the viability of turboelectric and hybrid-electric propulsion systems for future commercial aircraft (Refs. 286 to 296), though these studies have definitively identified battery energy storage as a key technological bottleneck to the near-term and mid-term feasibility of commercial transport-class aircraft hybridization. While a substantial body of work is currently underway to improve battery specific energy capabilities, previous projections of the capabilities of aerospace battery systems have placed the pack-level specific energy of these lithium-ion systems for the year 2030 at 500 Wh/kg (Ref. 297), well below the anticipated 800 Wh/kg levels that would make battery storage feasible for hybrid-electric propulsion of single-aisle commercial-transport aircraft (Ref. 101). As an alternative means for meeting the energy storage requirements for aircraft, many R&D efforts across the aeronautics community are actively considering the use of alternate fuels such as H₂ as an energy carrier for aviation. Although H₂ offers ultra-high specific energy (energy per unit weight), given the low energy density (energy per unit volume) of the medium in gaseous form, most applications have turned to LH₂ for commercial transport applications.

5.1.1.1 Design Methods

5.1.1.1.1 Requirements Definition

The current H₂-electric study considered design requirements and system-level integration impacts associated with the vehicle platform and, as such, did not capture energy and environmental impacts associated with fuel production. The importance of lifecycle impacts of fuel production is acknowledged, as the energy-related emissions for H₂ are highly sensitive to the production pathway and the degree of renewable power generation in the grid (Ref. 298). An entry-into-service timeframe of 2050 was also assumed. The aircraft design was constrained to produce zero CO₂ and NO_x at the vehicle level while targeting the same payload-range performance, comparable cruise Mach number, and equivalent gate compatibility of a Boeing 737-800. As such, these requirements are used to represent a replacement aircraft to existing markets, rather than a new vehicle intended to enable new market models. The resulting list of configuration requirements for the CHEETA aircraft design are further summarized in Table 46. The position on the payload-range diagram of the Boeing 737-800 used to define the performance requirements is shown in Fig. 311.

TABLE 46.—PERFORMANCE REQUIREMENTS OF CHEETA
CONCEPT, BASED ON SELECT BOEING 737-800 MISSION

Parameter	Value
Passengers (single-class).....	180 pax
Cabin	Single-aisle
Span (at airport gate)	Within Group III (79 to 118 ft)
Takeoff field length (standard day).....	8,200 ft
Range.....	2,935 NM at EOL
Cruise Mach.....	0.78
Cruise altitude.....	37,000 ft
Reserves.....	200 NM divert, 30 min loiter, 5% remaining fuel after landing
Regulation.....	FAR 25 compliant
CO ₂ emissions.....	0.0 lb/pmi (0.2 lb/pmi for B737-800)
NO _x emissions	0.0 lb/LTO-cycle (27 lb/LTO-cycle for B737-800)

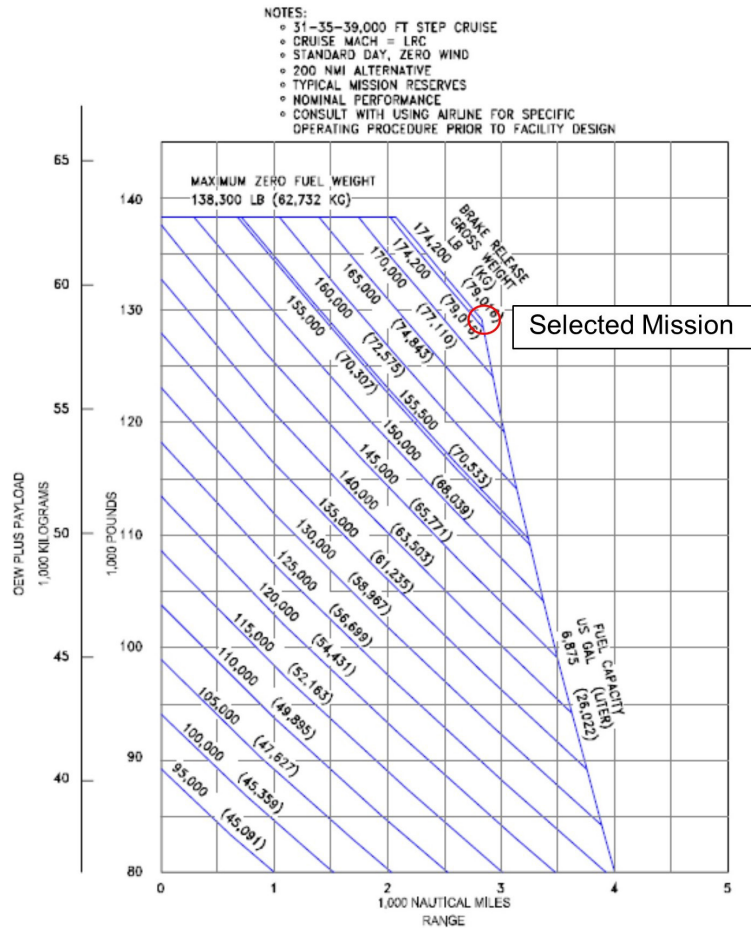


Fig. 311.—Payload-range diagram of Boeing 737-800 for long-range cruise (after Ref. 299) with selected mission performance for CHEETA design indicated.

5.1.1.1.2 Notional Configuration and Internal System Concept

The CHEETA aircraft concept, as sketched in Fig. 312, is a high-wing, V-tail, 180-passenger, single-aisle, transport-category aircraft (Ref. 300). Nine electric motor-driven fans, with three sets of propulsors over each wing and at the aft section of the fuselage, provide thrust for the vehicle. Fig. 313 depicts the major internal systems of the aircraft. Fuel cells, in green, are located near the propulsors to shorten the distance between electrical generation and usage. This proximity is also convenient for thermal system integration; the motors, inverters, and electrical transmission lines are cooled by the cryogenic LH₂ fuel (Ref. 31).

An unpressurized lifting centerbody was configured above the pressurized passenger cabin, such that the high-volume LH₂ storage tanks are placed within the aircraft OMLs. The tanks were positioned to be laterally outside of the passenger compartment regions to mitigate the risk of liquid or gaseous H₂ leaks into passenger compartments or when the unpressurized section vents H₂ out of the aircraft, as seen in Fig. 314. The placement of the H₂ storage system above the pressurized main fuselage was selected primarily to avoid catastrophic incidents that may result with undercarriage H₂ storage. Such cases could result from a failed landing gear system, where a wheels-up landing or gear puncture of the aircraft skin may result in ignition of the H₂ fuel stored within the tanks. Placing the tanks high on the aircraft provides a clear pathway for the highly buoyant H₂ gas to quickly move away from the aircraft if venting is

necessary. Placement of H₂ tanks in the centerbody region immediately adjacent to the overwing propulsor banks was avoided to reduce the risk of tank puncture in the event of an uncontained rotor or fan blade failure. These considerations were implemented pursuant to 14 CFR §25.963 regarding fuel tank integration, as shown in Fig. 314 (Ref. 301).

The configuration of the H₂-electric power system is also shown in the notional diagram in Fig. 313. The cylindrical LH₂ fuel tanks are shown in blue, and the batteries are shown in purple. The SC power transmission lines are shown in yellow, with circuit protection components shown, where the transmission lines are routed from the fuel cell power plants to the propulsor busbars in gray. Two sets of SC lines are provided between the powerplant and busbars to promote redundancy. Tie lines between busbars are provided to assist with power distribution in the event of failure in one of the fuel cell power plants. The power inverters, SC motors, and fan systems are also shown across the three banks of propulsors. Fuel cells were placed within the wing structure with an inlet compression system to facilitate the proper air/O₂ supply to the fuel cell stacks, while also ensuring that flow of H₂ was routed outward from the H₂ tanks, away from the main passenger compartment. Thus, there are no instances where H₂ lines were required to cross over portions of the fuselage occupied by passengers or crew, which would be a safety concern. Placing the fuel cells within the wings also allowed the weight of these components to be used to offset lift-induced bending loads that would otherwise manifest across the wing root. A fuel cell system was also positioned across the aft end of the aircraft to supply the adjacent propulsor bank. The aft-mounted power system components can be electrically isolated from the rest of the overwing power system, if desired in future design iterations. Even with extensive circuit protection in place, this modification could be necessary to lower the potential for catastrophic, cascading failures that would otherwise render all three propulsor banks inoperative. To counterbalance the center-of-gravity position associated with this aft-mounted fuel cell system, the battery system was placed in an electronics equipment bay across the forward portion of the aircraft, an arrangement similar to that found on the Boeing 787 family of aircraft.

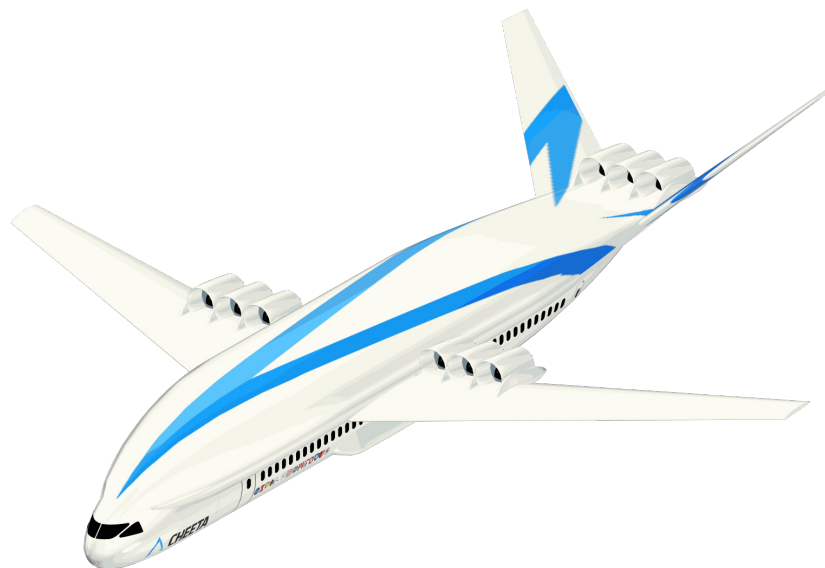


Fig. 312.—Isometric view of CHEETA aircraft concept.

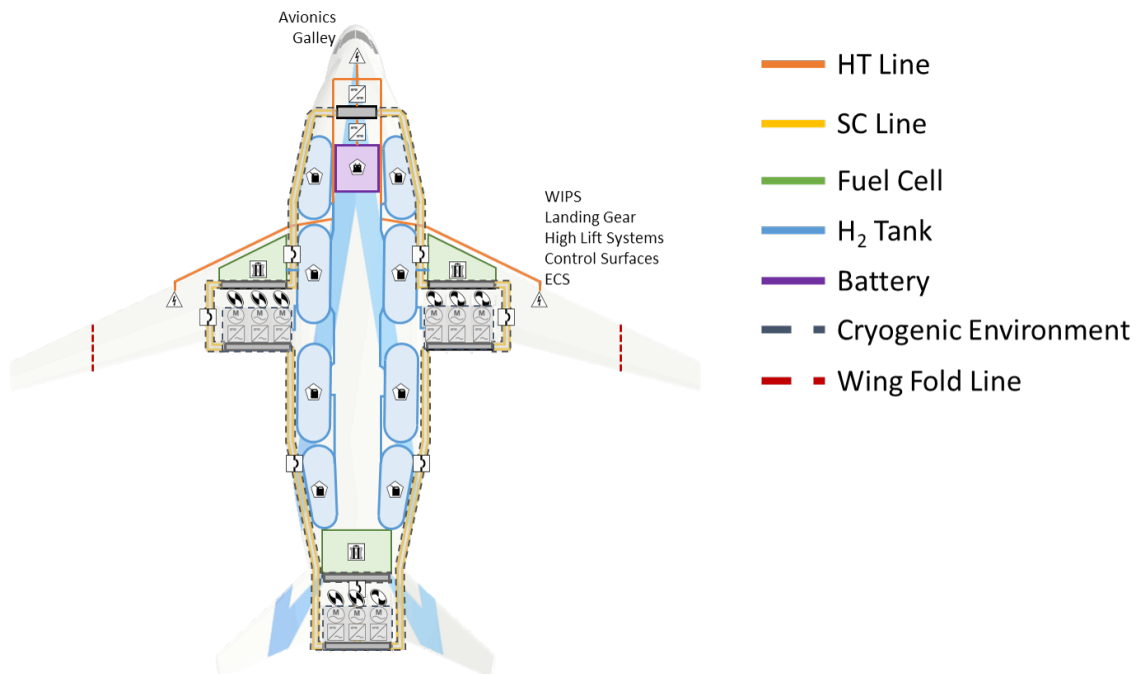


Fig. 313.—System overview of CHEETA aircraft concept.

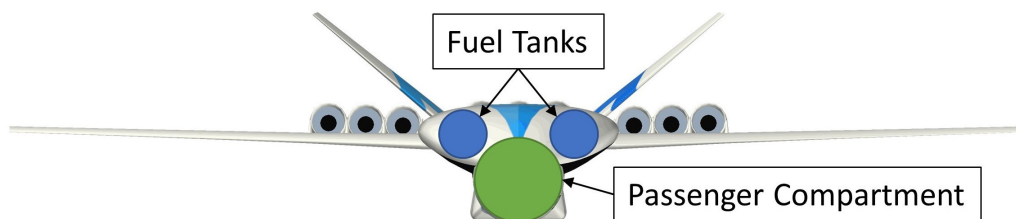


Fig. 314.—Front view of CHEETA aircraft concept demonstrating passenger and fuel tank location.

Given the reference to a Boeing 737-800, the cabin/cargo layout and dimensions of this reference aircraft were maintained for the CHEETA configuration. Due to the location and size of the aft-centerbody propulsor bank, a V-tail was selected for ease of integration. Other tail configurations can also be considered. Since the 737-800 is a Gate III-capable aircraft, CHEETA was configured with the same constraint, which limited the gate span to 118 ft. A 15% span increase was assumed to be viable for the CHEETA aircraft, however, by leveraging folding wing tips. This approach was considered feasible given similar wing-tip extension schemes used on contemporary aircraft (Ref. 302) and the much greater wing-tip extension scheme proposed in the Subsonic Ultra Green Aircraft Research configuration (Ref. 303).

5.1.1.1.3 Weight Estimate

A weight buildup for primary aircraft structures was performed using Torenbeek's methods published in Roskam (Ref. 304), where weight estimations for structures and systems analogous to historical aircraft, such as wings, landing gear, and avionics, were used. No reductions in component weights due to modern or future materials, such as lightweight composite structures, were assumed. As a result, the direct comparison to the Boeing 737-800 was deemed appropriate, though further reductions in aircraft weight and energy utilization are feasible with the incorporation of advanced structures. The selected weight estimation methods were validated using a complete mass estimate for the 737-800 operational empty weight and

maximum takeoff weight (MTOW) to within 1% of published values. The General Dynamics weight estimation method, as published in Roskam (Ref. 304), was also utilized, though greater deviation was observed when estimating the mass of the Boeing 737-800 reference aircraft. As such, Torenbeek's approach was chosen as the primary weight estimation method for the majority of the CHEETA configuration. However, to capture a broader set of sizing considerations, an average of both the Torenbeek and the General Dynamics wing weight methods was used. These estimates were then combined with the remaining primary empty-weight components of the aircraft as estimated by Torenbeek's method. The weight estimation method was set up as parametric functions to allow for iteration across variations of aircraft geometry. For systems without historical baselines, such as the fuel cell, SC motors, and LH₂ fuel tanks, new weight estimation techniques were developed. Fuel cell weight estimates are described in greater detail in Section 5.1.1.1.6. Fuel tank estimates were produced by assuming a tank weight fraction ($W_{\text{tank}}/W_{\text{H}_2}$) of 0.65, based on the anticipated weight scaling of novel LH₂ tank designs created by GE Research (Ref. 305) as a part of the broader CHEETA research program. Similar weight forecasts for fully SC electric motors (Ref. 21), power electronics (Refs. 306 and 307), SC power transmission systems (Refs. 308 and 309), and distributed ducted fan systems including fans, casings, and nacelles (Refs. 31 and 13) were drawn from related contributions to the CHEETA research program based on forecasted technology capabilities for the year 2050, alongside trends obtained from available data.

5.1.1.1.4 Aerodynamics Estimate

Aerodynamics estimates were performed using Roskam's (Ref. 310) methods based on geometry-specific parameters. Fuselage, tail, and propulsor geometry remained constant between different configurations. Nacelle drag was also calculated for the anticipated DP layout, which was constant between configurations. Estimates for wave drag were developed using the Delta method (Ref. 311), which has been shown by Vargas and Vos (Ref. 312) to be an accurate conceptual design-level wave drag estimation method for aircraft of this class.

5.1.1.1.5 Performance Estimate

A performance analysis was conducted using an Euler method to simulate the rigid-body motion of the aircraft through time across a mission, where a look-up table was used to represent the aircraft system aerodynamics, buildup of weights, and fuel cell power system using the aforementioned modeling methods. The performance simulation framework leverages an iterative scheme to determine the fuel weight required to complete the mission, recomputes the aircraft weight based on changes in fuel weight from previous iterations, and closes on a design.

The takeoff thrust requirements were derived using the Raymer (Ref. 313) takeoff parameter method, verified with an Euler method time-step simulation of the rigid-body equations of motion. A takeoff C_L was assumed as an input. Since the aircraft has three independent banks of propulsors on the body and left and right wings, as shown in Fig. 312, a three-propulsor (tri-jet equivalent), one-engine-inoperative (OEI) thrust profile was chosen to best represent the propulsor failure mode with respect to the Federal Aviation Regulations (FAR) balanced field takeoff length. It is assumed that the fuel cells, drive inverters, SC wiring, and the rest of the powertrain are designed to be sufficiently redundant or independent for this OEI consideration.

Note that second segment climb requirements were not evaluated at this time.

5.1.1.1.6 Fuel Cells and Degradation

A conceptual diagram of a PEMFC is shown in Fig. 315. A set of compressors is used to provide the necessary O₂ mass flow rate to the fuel cell system to achieve the desired power setting. As the supplied

H₂ ionizes and crosses from the anode to the cathode side of the fuel cell electrolyte, it is combined with the supplied O₂ on the cathode side of the fuel cell. The unutilized H₂ is recirculated on the anode side of the fuel cell membrane, and unutilized air and the water vapor products are exhausted across the outlet. PEMFCs were chosen based on a previous study which indicated SOFCs lacked the specific power for transport-class aircraft applications (Ref. 300). Notably, high-temperature PEMFC systems were leveraged for the current study due to the greater heat rejection capability of the higher (180 °C) operating temperature.

Contemporary PEMFCs have been demonstrated to produce stack-level specific power of 5.5 kW/kg (Ref. 314). However, once balance of plant, humidifier, and compressor systems are added, the system-level specific power drops precipitously to 1 kW/kg. Technology projections to year 2050 were made with industry advice (Ref. 315), based on the recent progress rates of low-temperature PEMFC systems scaled to high-temperature systems. To estimate the weight, a notional 1-MW stack was conceptually developed, with system components including a fuel cell membrane, balance of plant, compressor, motor, and radiator. The breakdown of fuel cell component weights is shown in Table 47.

Like any conventional power plant, fuel cell systems gradually degrade with continued operation. The fuel cell degradation process varies greatly based on operating condition, such as load and temperature, as well as H₂ purity, resulting in degradation rates between 0.5 to 20 $\mu\text{V}/\text{h}^{-1}$ for steady-state lifetime tests (Ref. 316). The degradation process manifests as an aggregate decrease in cell voltage over time, which subsequently impacts aircraft performance through increased fuel use and reduced maximum power. The degradation rate will define the maximum number of flights that can be completed on a given stack until a certain lower performance limit is reached, signifying the fuel cell EOL.

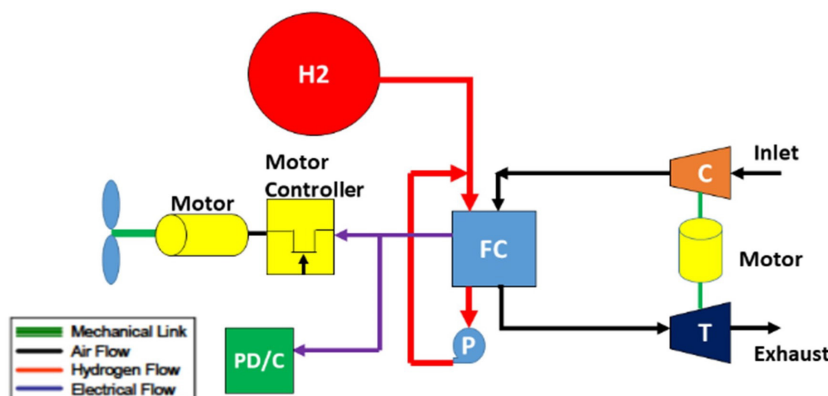


Fig. 315.—PEMFC architecture based on (Ref. 317).

TABLE 47.—BREAKDOWN OF FUEL CELL COMPONENT WEIGHTS

Component	Net weight, kg
Fuel cell	175
Balance of plant	102
Radiator	48
Compressor	28
Compressor motor	12
Total weight	365

To find the impact that fuel cell degradation has on aircraft performance, a range of degradation rates across 20,000 flight hours was assumed to find an aggregate voltage drop. The range of fuel cell degradation rates was curtailed to 0.5 to 5 μVh^{-1} to create a resulting voltage degradation range of 10 to 100 mV. This range was selected because it represents the cumulative degradation voltages that enable aircraft design to close, as will be demonstrated in Section 5.1.1.2.4. While a fuel cell under real-world conditions does not produce a constant degradation rate, cumulative fatigue damage theory assumes the total decay rate is treated as a linear combination of decay produced by different conditions (Ref. 318). Therefore, the aggregate voltage drop found is indicative of potential real-world degradation and performance.

A previously used PEMFC model was modified to include degradation (Ref. 319), which was modeled with a constant voltage decrease across the operating current densities of the fuel cell. While this method underestimates fuel cell output at low power, it matches previously published predictions for maximum power (Ref. 320). This approach allows for the most accuracy at the fuel cell sizing points at maximum power at the cost of an underestimate of fuel cell capability at minimum power, leading to an overestimation of fuel consumption by approximately 1% at cruise. An aggregate degradation of 100 mV on an individual cell and the resulting impact on total power generated for a 1-MW gross power stack are shown in Fig. 316. This degradation scenario represents the worst-case conditions of all degradation rates simulated.

Fig. 316(a) shows the impact that voltage degradation has as a function of current density. Here, degradation was applied as a constant voltage drop across the entire range of current density. Fig. 316(b) shows the performance of the fuel cell with power on the x-axis and net efficiency on the y-axis at takeoff conditions. The effect of the voltage degradation leads to both a reduction in efficiency throughout the operation of the fuel cell and a reduction in peak power of 150 kW per stack.

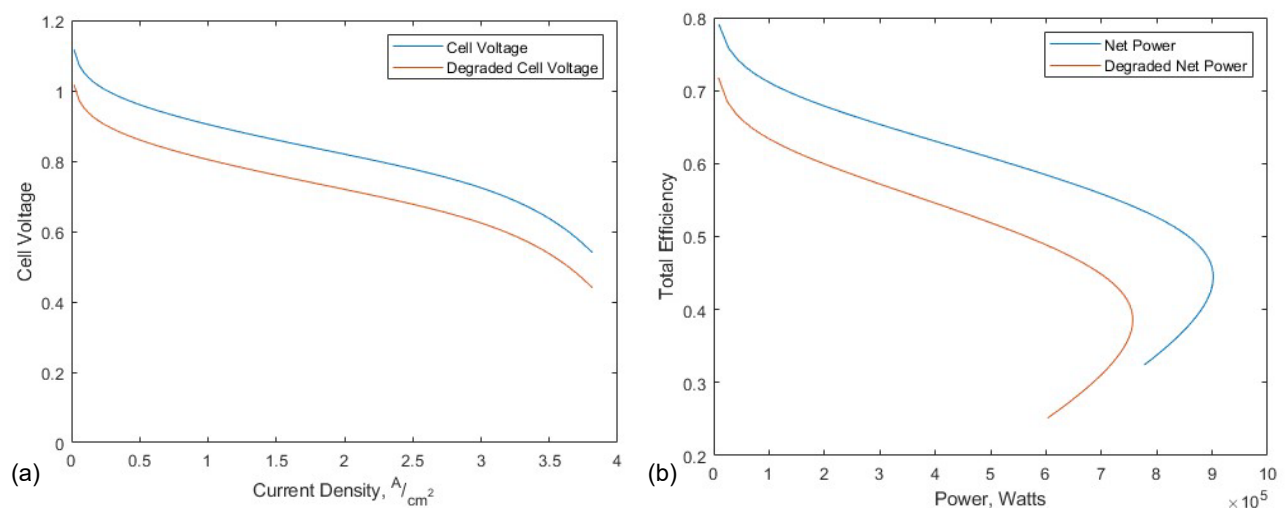


Fig. 316.—Degradation impact on (a) cell voltage and (b) stack power.

For the initial aircraft conceptual design, the critical sizing point for the fuel cell system was determined as the greater of (1) the power required to meet the balanced field takeoff requirement of 8,200 ft on a standard day at sea level or (2) the top-of-climb requirement for a service ceiling of 37,000 ft on a standard day. Hotel loads of 1 MW to run the cabin pressurization, flight controls, icing protection systems, actuators, galley loads, and other critical systems were included in both sizing conditions. When considering these sizing points, a major difference between fuel cell systems and jet engines is the difference in power and thrust lapse with altitude. Fundamentally, for a turbofan engine, the power generated by the turbine is mechanically linked to the compression system and primary propulsive device, such that the power-producing components cannot be decoupled from the power expenditure as thrust and compressive work. However, for a fuel cell system, the electrochemical power production capability only depends on the provided conditions (density and pressure) of the supplied O_2 and H_2 . Although the ambient pressure and density of the air decrease as a function of altitude, an independent inlet compression system can be used to provide an air supply to the fuel cell system at a variable pressure ratio with minimal impact on the power setting and fan performance. Thus, the fuel cell system can be configured to maintain a significant proportion of rated power production at the cruise altitude relative to what is produced at sea level, though this approach also requires a parasitic electrical power draw from the fuel cell system in order to run the compressor and compensate for precompression drag on the vehicle system. The effect of this system is analogous to that of a turbocharger on a reciprocating engine.

This consideration for propulsive power lapse rate is indicated in Fig. 317, where the relative power capability of a typical gas turbine engine is compared to the fuel cell model used in this study. On Fig. 317(a), if the peak power requirements are sized based on the takeoff conditions, the fuel cell system will retain approximately 84% of this peak power available at the cruise altitude, whereas the jet engine will only retain approximately 29% of the equivalent sea-level power available. This leads to a large amount of excess available power at cruise for the fuel cell configuration. Conversely, on Fig. 317(b), if the top of climb condition is used to size the power requirements of the aircraft, when projected back to the equivalent sea-level power capability, the fuel cell system can be sized with a significantly reduced peak power overall as compared to a jet engine configuration. The result of this variable-compression approach is a reduced lapse rate in the fuel cell gross power available with altitude as compared to a comparable thermal engine system, though reductions in available power at higher altitudes can be observed due to the increase in parasitic load required from the fuel cell stack to pressurize the inlet air as the altitude increases.

This trend allows the aircraft power system to be sized based on an unconventional tradeoff between top-of-climb and takeoff sizing parameters, as compared to those typically observed for thermal engines. The result is a system that produces less power at sea level but more power at top of climb than its thermal engine equivalent. As a consequence, for a fixed takeoff-lift coefficient, an aircraft with a fuel cell powerplant sized by top of climb may require a larger wing area or extensive high-lift systems to meet balanced field length requirements when compared to those using conventional turbofan systems. An additional consequence of this variation in power lapse behavior is that the initial climb rate for the H_2 fuel cell aircraft is much slower than that of a conventional jet airliner at lower altitudes. However, while this initial climb rate is slower than those typically observed for turbofan-powered aircraft, the climb rate remains relatively constant at about 1,000 ft/min until an altitude of 30,000 ft, where reduced air density and pressure and acceleration to cruise speed results in a reduced climb rate at these higher altitudes.

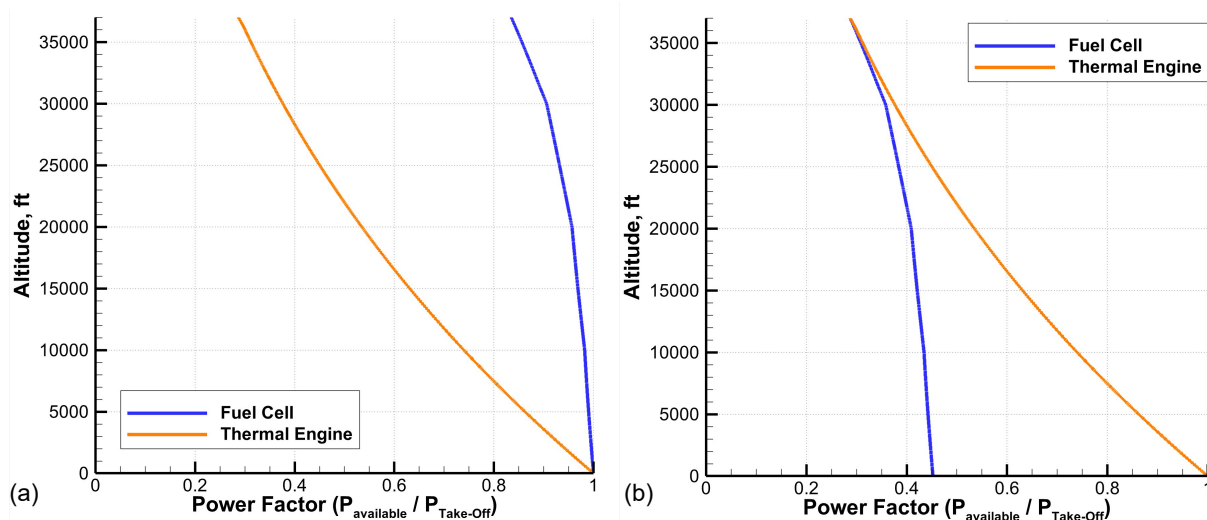


Fig. 317.—Variation in relative power available with changing altitude for fuel cell power system and jet engine. (a) Power system sized by takeoff. (b) Power system sized by top of climb.

5.1.1.1.7 Heat Exchanger

Initial estimates were performed with a simplified pressure-drop HEX model. This model used an assumed area of 1 m^2 per 1-MW fuel cell stack and estimated the drag assuming a pressure drop relative to the magnitude of heat rejected. These initial estimates are juxtaposed by further refinements that typify the importance of utilization of the low-grade waste heat in fuel cell TMSs, such as through the use of a dedicated ducted HEX. A ducted HEX was configured based on the methods discussed in a previous study (Ref. 319) and was integrated to the baseline aircraft. This HEX utilizes the Meredith effect to increase the pressure of a fluid passing through the HEX. A variable exit nozzle is then used to increase the velocity at the nozzle. This accelerated flow provides a thrust element that, while not sufficient to provide a net thrust once nacelle and internal flow drag forces are considered, reduces the drag impact of the HEX. The frontal radiator area of the HEX is defined by α , the ratio of HEX area to propulsor area. To characterize the impact of heat utilization on aircraft performance, the ducted HEX model (denoted as the isolated HEX in Ref. 319) has been implemented into the sizing methods used to create the CHEETA configuration. Thermal system weight was estimated using the HEX weight scaling factor ($\psi = 53.7 \text{ kg/m}^2$) and Roskam's method for turbojet nacelle weight estimation was applied to account for the fairing and structure around the radiator (Ref. 321). The combined drag estimation was used from a previous study (Ref. 319), using a $C_D = 0.04$, using the frontal radiator area, A_r , as the reference area.

5.1.1.2 Sizing Results

Using the sizing methods described previously, a series of parametric studies were conducted to survey the aircraft design space. First, a pressure-drop radiator model was used for the HEX, allowing for a rapid exploration of the design space. This approach was used to produce a carpet plot showing the impact of varying wing area and aspect ratio on MTOW and fuel weight. A design point was chosen, and the impact of integrating an isolated, ducted HEX was explored by varying α , the HEX area to propulsor area ratio. After selecting an optimal α for the fuel cell TMS, the aircraft sizing process was repeated to explore the impact that heat utilization had on the MTOW and fuel weight. Then, using the selected configuration with an isolated, ducted HEX, an oversizing of the fuel cell peak power available was explored to reduce fuel weight at the expense of increasing MTOW. To characterize the impact of degradation on aircraft performance, fuel cells of various cumulative degradations were then considered.

5.1.1.2.1 Aircraft Conceptual Design Space

Candidate design configurations were selected based on minimizing either aircraft MTOW or LH_2 fuel weight. The former configuration was assumed to provide a surrogate representation of an aircraft configuration that minimized aircraft acquisition costs, whereas the latter configuration was assumed to provide a configuration with minimum operating costs. Carpet plots were constructed to visualize the effect of varying aspect ratio, wing area, and takeoff lift coefficient on MTOW and fuel weight. These parameters were identified as the primary drivers to the power required at the sizing points of takeoff and top of climb. The lightest MTOW configurations were found at the transition between the two sizing points, which implicitly resulted in the least excess power at any given point and thus the smallest power plant required. The lightest fuel weight configurations were consistently observed as being sized for takeoff, where the larger peak power available from the fuel cell results in a higher fuel cell electrochemical conversion efficiency at cruise.

Fig. 318 shows carpet plots for the baseline sizing approach, using a basic pressure drop model for the HEX radiator drag. The takeoff lift coefficient is set to be 2.2, which is assumed to be approximately the same as for a Boeing 737-800 (Ref. 286). The carpet plot includes variation in aspect ratios and wing surface areas between 6 to 12 and 1,600 to 2,500 ft^2 , respectively, to show the impact of changing aerodynamic performance on the aircraft MTOW and fuel weights. The purple lines show the Group III wingspan of 118 ft (left) and the Group III + 15% wingspan constraint (right) of 135.7 ft. The carpet plot has color contour changes shown at steps of 500 lb in MTOW, and dashed lines show the change of fuel weight in steps of 500 lb. The yellow star indicates the minimum MTOW configuration. The least fuel configuration is indicated by the red circle. A yellow line denotes the transition of configurations that are sized by top of climb (above the yellow line) to takeoff (below the yellow line).

The baseline configuration was selected from the subset of the design space within the maximum span constraint. A maximum admissible aspect ratio of 10 was chosen as it represents the maximum aspect ratio of contemporary transport aircraft (Ref. 302), as shown by the bold black line on Fig. 318. It was assumed that the standard sizing methods used in the current study were unable to sufficiently capture the aeroelastic impacts and additional design considerations above this aspect ratio. Due to the flat design space, an engineering judgment was made to select a wing area of 1,840 ft^2 , thus the baseline configuration has the maximum allowable wingspan of 135.7 ft. The location of the baseline on Fig. 318 is shown by the orange triangle. The judgment was made to leverage a minor MTOW increase to produce a fuel weight decrease within the other constraints provided. The net effect of this was a 78-lb increase in MTOW with a 27-lb decrease in fuel weight. Both of these values are less than the 1% of their respective values, well within the error for the weight estimation techniques applied. Results comparing these configurations are shown in Table 48.

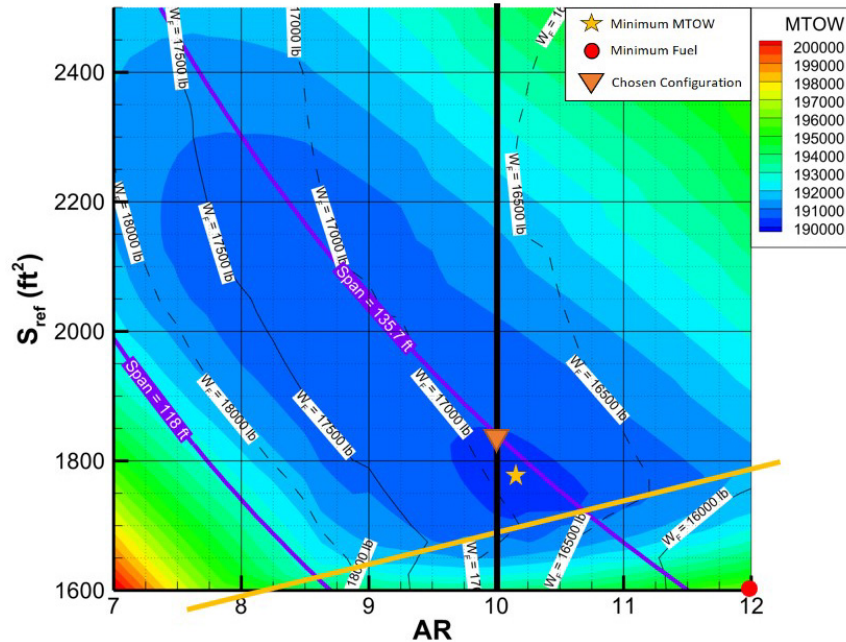


Fig. 318.—MTOW and fuel weight variation with parametric variation of reference area and aspect ratio ($C_L = 2.2$) for HT-PEMFC.

TABLE 48.—COMPARISON SHOWING THE MINIMUM MTOW, MINIMUM FUEL WEIGHT, AND CHOSEN BASELINE CONFIGURATIONS

	Min. MTOW	Min. fuel	Chosen configuration (baseline)
AR	10.125	11.5	10
S_{ref} , ft ²	1,775	1,625	1,840
Fuel cells, lb	20,175	23,351	20,081
Fuel Tanks, lb	10,988	10,150	10,971
Fuel Weight, lb	16,905	15,616	16,878
MTOW, lb	190,406	194,860	190,479

5.1.1.2.2 Aircraft Conceptual Design Space with Ducted HEX Systems

To integrate the ducted HEX to replace the pressure-drop HEX model used in the previous section, a new value, α , must be defined. To characterize its impact on MTOW and fuel weight, sizing studies were performed for the baseline configuration ($AR = 10$, $S_{ref} = 1,840$ ft²), across a parametric variation in α from 0.5 to 2 were swept. The resulting MTOWs and fuel weights are shown in Fig. 319. Configurations that did not close were removed, curtailing the α range to $0.55 \leq \alpha \leq 1.4$. The minimum MTOW occurs at $\alpha = 0.65$.

The impact of the ducted HEX was studied by revisiting the aircraft sizing to identify MTOW and fuel weight produced by varying wing aspect ratio and S_{ref} at a constant $\alpha = 0.65$. Fig. 320 shows the wing planform area on the abscissa and aspect ratio on the ordinate in a fashion consistent with the baseline sizing study. Variations in MTOW are shown in increments of 500-lb steps by the color contours and fuel weight of 500-lb steps by the isocontour lines. The wingspan constraints of 118 and 135.7 ft are shown by two purple lines. A yellow line across the design space denotes the separation between top-of-climb-sized configuration (above) and takeoff-sized configurations (below).

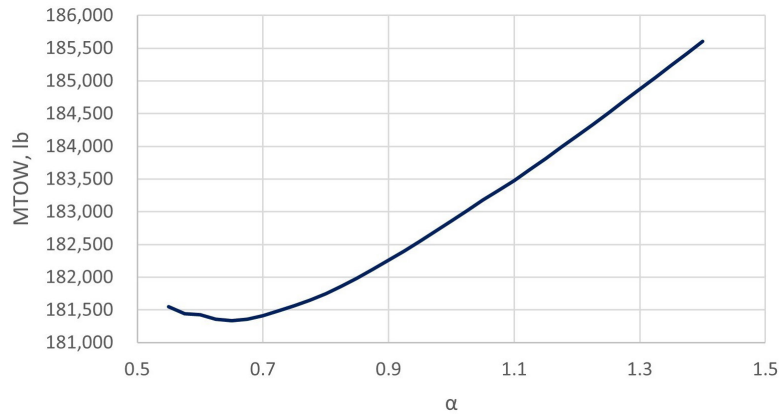


Fig. 319.—Impact of HEX-fan area ratio on MTOW.

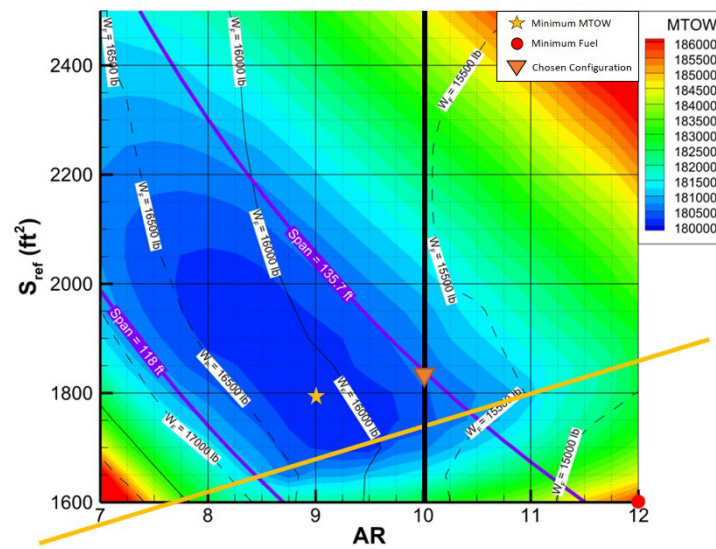


Fig. 320.—Carpet plot showing the impact of wing area and aspect ratio on MTOW and fuel weight at $\alpha = 0.65$.

Three configurations are shown. The red dot in the lower right of Fig. 320 depicts the minimum fuel configuration, the yellow star denotes the minimum MTOW configuration, and the orange triangle denotes the selected configuration. As before, the configuration was judiciously selected to have a maximum admissible wing aspect ratio of 10, denoted by the thick black line, and a maximum wingspan constraint of Group III + 15% (135.7 ft.), matching the previous baseline design point with $S = 1,840 \text{ ft}^2$ and aspect ratio = 10. This configuration provides a 434-lb fuel weight reduction (2.68%) for a 534-lb MTOW increase (0.30%) as compared to the minimum MTOW point. Comparing the baseline aircraft with a pressure-drop radiator to that produced using an isolated, ducted HEX aircraft, the fuel weight decreases by 1,131 lb (6.70%) and the MTOW decreases by 9,878 (5.19%). The primary driver of these reductions is a reduction in peak power, owing to the reduction in HEX drag at the peak power and sizing point of top of climb. This overall weight reduction leads to a significant decrease in fuel weight. A summary of these configurations is shown in Table 49. The values labelled “Baseline” represent a configuration sized with a HEX system using a pressure drop radiator. The other columns show the impact of utilizing waste heat in an isolated HEX system with $\alpha = 0.65$, all of which show significant decreases in MTOW.

TABLE 49.—SUMMARY OF CONFIGURATIONS
UTILIZING WASTE HEAT WITH DUCTED HEX SYSTEMS

	Baseline, pressure drop HEX	Chosen configuration	Minimum MTOW	Minimum fuel weight
Aspect ratio	10	10	8.5	12
S_{ref} , ft ²	1,840	1,840	1,900	1,600
Fuel cells, lb	20,081	15,723	16,585	20,000
Fuel weight, lb	16,878	15,747	16,181	14,726
MTOW, lb	190,479	180,601	180,067	185,860

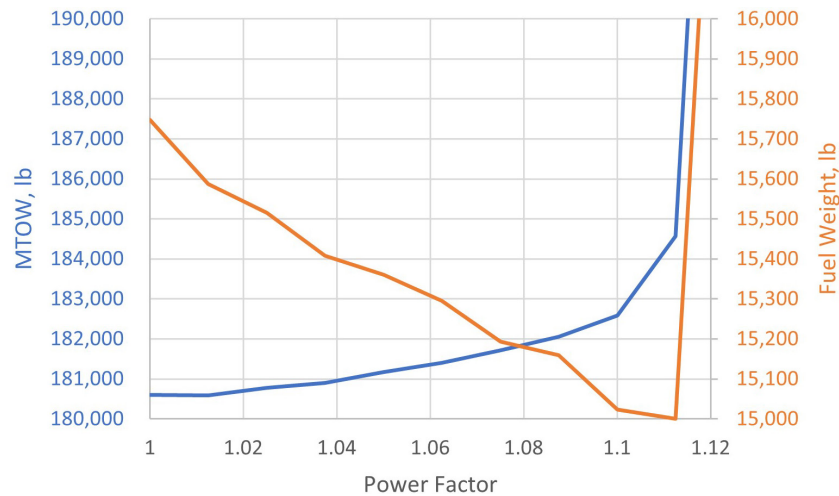


Fig. 321.—Increasing power factor reduces fuel weight until critical point.

5.1.1.2.3 Power Factor

As seen in a previous study (Ref. 300), a fuel cell system can be configured to have a maximum power available in excess of that required for the critical sizing point. Scaling the peak sizing power requirement by a power factor ($P_F = P/P_{req}$) causes a decrease to the fuel weight at the expense of an increase in MTOW. Power factor can also be used to appropriately size the fuel cell power system for beginning-of-life (BOL) power, preparing for EOL degradation, which will be further expounded in the following section.

To observe the influence of power factor on a configuration, the value of $\alpha = 0.65$ was used with the chosen configuration discussed previously, and power factors from 1.0 to 1.5 were applied. Fig. 321 shows how increasing power factor increases MTOW and reduces fuel weight. The MTOW increase is due to increased fuel cell weight to provide the additional power available. Weight estimates for structural elements such as the fuselage and landing gear also increase to accommodate the increased powerplant weight. Since the power required at cruise is nearly constant, as fuel cell weight increases, the power demand per cell at cruise decreases, allowing the fuel cell to operate at a more efficient condition. This increase in efficiency for the cruise leg of the flight is the primary driver of the reduced fuel weight. However, if a power factor of 1.11 is exceeded, a rapid and substantial increase in weight is observed. This trend is observed because fuel cells no longer provide sufficient power to meet the takeoff sizing constraint with the fixed propulsor size. This limitation is reconcilable by increasing takeoff C_L or increasing wing area, but assuming a constant geometry, including α , $PF = 1$ represents the maximum practical value of the fuel cell power available for this configuration.

5.1.1.2.4 Degradation Study

To compare configurations across different degrees of fuel cell cumulative degradation, an additional sizing study was conducted, shown in Fig. 322. Here, the power factor was defined based on the fuel cell maximum power output sized at BOL in order to accommodate the design mission from Fig. 311 with EOL degraded performance. To accommodate larger fuel cells, large HEXs are required to expel the greater peak heat loads. Therefore, α also varies, creating the carpet plot shown in Fig. 322. Changes in MTOW are shown in color steps of 1,000 lb, and changes in fuel weight are shown in 250-lb increments by the isocontour lines. For a given α , there is a strict maximum power factor that can be supported, as was seen in Fig. 319.

Using the cumulative fatigue theory, different EOL cell-level voltage losses due to degradation were investigated. Steady-state degradation rates of 0.5 to $5.0 \mu\text{Vh}^{-1}$ across a 20,000-h design life accumulate to cell-level voltage losses of 10 to 100 mV. These results are then overlaid as yellow lines on Fig. 322 to show the corresponding size of fuel cell and HEX that is required to achieve design mission performance at the EOL.

The background color map is an underprediction; it does not account for increased fuel tank size, and associated weight, to accommodate for the EOL fuel capacity. However, using the figure as a comparative tool provides a starting point to interpret and compare the results of variable stack-level voltage losses due to fuel cell degradation. To accommodate these losses, the onboard fuel cell size must increase, as shown by the increase in power factor. A subsequent rise in α occurs to accommodate the increased peak heat loads. For a constant power factor, as α increases, a rise in MTOW is observed due to the increased size and weight of the radiator nacelles. A reduction in fuel weight is also seen with power factor, as at BOL the additional power available means that at cruise the fuel cells can perform for less demand per cell, increasing efficiency.

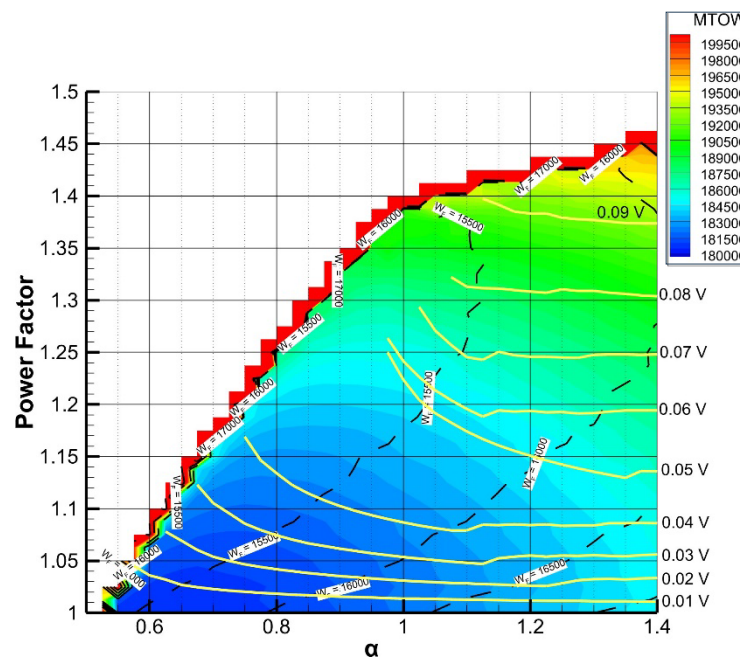


Fig. 322.—Degradation points plotted on α versus power factor carpet plot.

TABLE 50.—COMPARISON OF DEGRADED CONFIGURATIONS

Fuel cell degradation, V/cell	0.00	0.03	0.06	0.09
α	0.650	0.725	1.025	1.125
Oversize	1.000	1.096	1.226	1.398
Fuel cell weight, lb	15,723	16,979	18,878	21,623
Fuel tank weight, lb	10,301	10,852	11,951	12,689
Fuel weight, BOL, lb	15,747	15,523	15,727	15,803
Fuel weight, EOL, lb	15,747	16,695	18,386	19,521
MTOW, lb	180,601	184,164	190,333	196,200

Four configurations were chosen to serve as a comparison that highlights how the conceptual aircraft performance changes due to fuel cell degradation considerations. In each case, the minimum MTOW design was chosen across the parametric variation in α . These points were chosen at cumulative degradations of 0.00, 0.03, 0.06, and 0.09 V/cell, respectively. Each configuration has the same OML, with the exception of radiator size, with the same wing planform of 1,840 ft² and an aspect ratio of 10. The results are tabulated in Table 50.

At larger values of cumulative degradation, the fuel cell size must substantially increase to meet the peak power requirement for the design mission at EOL. This increase in fuel cell size is met with increased efficiency at cruise due to the decrease in power required per cell, contributing to decreased fuel weight at BOL. The magnitude of this fuel decrease is such that all configurations sized for degraded fuel cells, despite having higher MTOW, have lower fuel consumption for the design mission at BOL. The isocontour lines show that this effect tapers off with power factors above 1.2, as the additional weight starts to require additional power.

Significant tradeoffs are observed during this configuration process. For example, a 50% increase in degradation from 0.06 to 0.09 V/cell leads to a 15% increase in fuel cell weight and a 3% increase in MTOW. Alternatively, the increase from 0.03 to 0.06 V/cell leads to an 11% increase in fuel cell weight and a 3% increase in MTOW.

5.1.1.2.5 Payload-Range Performance

Since the effects of degradation significantly reduce the peak power and decrease cruise efficiency of the CHEETA aircraft at EOL, this suggests that at BOL the aircraft is capable of exceeding the range of the design mission. To investigate this possibility, a payload-range diagram was created, comparing the CHEETA aircraft against the Boeing 737-800. The selected CHEETA configuration assumes a fuel cell sized to accommodate a 0.06-V/cell degradation as shown in Table 50. The resulting payload-range performance of the configuration is shown in Fig. 323. The red line denotes the undegraded fuel cell performance, corresponding with a start-of-design-life aircraft, and the orange line denotes the degraded fuel cell performance, corresponding to end-of-design-life aircraft performance. The blue payload-range envelope represents the payload-range capabilities of the reference Boeing 737-800 aircraft. The gray circles represent the U.S. domestic routes flown by the Boeing 737-800 aircraft in 2019 (Ref. 322) and are used to present a comparison for the payload-range capabilities of the CHEETA fuel cell aircraft throughout its design life.

For the design payload of 35,000 lb, the aircraft with 0 V/cell degradation has a range of 3,500 NM, as compared to the 2,935-NM range of the 0.06 V/cell degraded aircraft and Boeing 737-800 aircraft. This increase in payload-range capability covers 99.99% of historical routes flown by Boeing 737-800 aircraft in 2019. As degradation occurs, the performance of the aircraft decreases to results comparable to the Boeing 737-800, allowing the aircraft to cover 99.96% of previously flown routes.

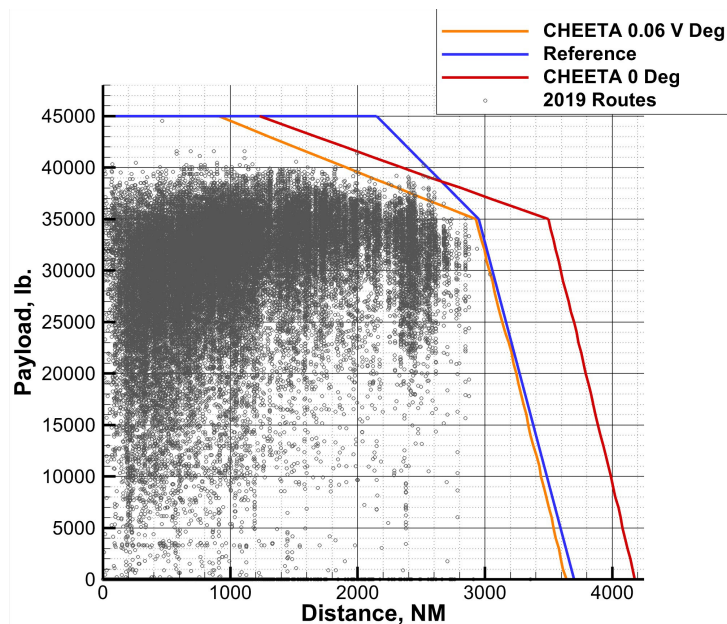


Fig. 323.—The net change of aircraft payload-range performance due to degradation, as compared to an incumbent reference aircraft.

5.1.1.3 Aircraft Conceptual Sizing: Conclusions

This study shows the impacts that a LH₂-fuel-cell-based electric propulsion system has on the configuration and layout of a single-aisle, transport-class aircraft with the consistent performance capability of a Boeing 737-800. Through an iterative design process, it is demonstrated that fuel cell power systems lead to an increase in MTOW relative to modern kerosene-and-turbofan-powered systems. However, higher electrochemical efficiency and a decreased lapse rate with altitude that are achieved by using an independent inlet compression system enable comparable performance with contemporary aircraft. Utilization of low-grade waste heat from HT-PEM fuel cell systems, such as through use of ducted HEX systems to decrease the drag penalty of a radiator system, improves aircraft performance, reducing MTOW and fuel consumption.

Like many other power systems, fuel cells degrade over time. While fuel cell voltage losses diminish fuel cell peak power and efficiency capability, this effect can be incorporated to LH₂ fuel cell aircraft sizing. HEX systems must be codesigned with projected EOL fuel cell performance in mind to accommodate the EOL heat loads. Provided a goal for EOL payload-range performance, a LH₂ fuel cell aircraft can be designed to satisfy performance requirements at this point and will see significantly improved BOL performance. This design method demonstrates the feasibility of the design of LH₂ fuel cell aircraft throughout a defined life cycle. While precise degradation rates of fuel cells in aviation operations are unknown, the application of cumulative fatigue theory provides a way to capture the net effects of variable degradation.

5.1.2 Thermal System Integration

Throughout the development of the CHEETA aircraft conceptual design, it was recognized that definition of a suitable thermal management strategy for the H₂-electric power system would serve as a key governing factor in the overall feasibility of the aircraft system. For this reason, a purposeful analysis of vehicle-integrated strategies of heat rejection were considered.

For many electrified aircraft concepts, large amounts of waste heat are generated by propulsive power components such as batteries (Ref. 323), fuel cells (Ref. 324), power transmission cables, power

electronics, and electric machines (Ref. 31). Thermal management and utilization of waste heat is therefore an important issue for these configurations (Refs. 325 to 327).

Potential solutions to address this issue depend on the quality and magnitude of the waste heat thermal load. Quality (or grade) refers to the temperature of the rejected heat. High-temperature waste heat is more easily utilized, and therefore higher grade, than low-temperature heat. For example, high-quality heat (1,400 °C) produced via combustion in aero-engines directly generates thrust power via an exhaust jet (Ref. 328). For low-quality heat (80 °C), uses are limited to utilities such as wash water heating, wing deicing, and fuel preheating (Ref. 329). In this work, we consider the thermal loads associated with HT-PEMFCs, which operate at temperatures up to 200 °C (Ref. 330).

The thermal load magnitude also dictates what solutions are practical. For a turboelectric aircraft such as the STARC-ABL concept (Ref. 331), the kW-scale thermal loads could be managed by OML HEXs (Ref. 332). Limits on available aircraft surface area make this strategy impractical as the sole solution for the MW-scale thermal loads of a fuel-cell-powered aircraft such as CHEETA concept (Ref. 300). Conventional radiator-type cooling systems introduce substantial parasitic drag penalties, mainly due to skin friction on additional ducting and radiator surface area (Ref. 300). An avenue to mitigate the penalties, however, is through utilizing the waste heat from the fuel cell to produce thrust (Ref. 333). The heat addition in a ducted radiator creates thrust, as in a ramjet (Ref. 328), as has been quantified through combined heat transfer and internal flow analyses (Refs. 334 and 335).

In the following sections, we provide a conceptual overview of the CHEETA fuel cell system, present a thermal management solution to address CHEETA-like applications, quantify the impact of that thermal system on aircraft-level performance, and discuss multiple integrated thermal system architectures and their potential benefits.

5.1.2.1.1 PEMFC Power System

PEMFCs generate electrical power by combining H₂ fuel at the anode and ambient O₂ at the cathode to form water (Ref. 336). At the catalyzed membrane surface, protons H⁺ pass through the membrane while electrons e⁻ pass through the circuit connecting the electrodes. For an ideal fuel cell, the voltage is governed by the Gibbs free energy of formation for water at standard pressure (1 bar) and the current is governed by the fuel flow rate (Ref. 336). Fuel cells can be combined in series to form stacks of higher voltage (equal to the number of cells times cell voltage) and in parallel for increased power output at a fixed voltage.

In practice, several mechanisms limit fuel cell operation:

- Ohmic losses, related to the electrical resistance of the fuel cell and circuits
- Activation losses, related to the chemical kinetics at the anode and cathode surfaces
- Mass transport losses, related to the nonideal transport of reactants or products to or from the electrodes
- Fuel crossover and short-circuiting, related to the loss of energy from wasted fuel or electrons

Each of these loss mechanisms reduces the output power, and therefore the efficiency, of the fuel cell. We assume that all fuel energy not producing electrical power ($\dot{W}_{fc}$) produces heat in the fuel cell, which will be sent to a TMS at a temperature equal to the stack operating temperature. These losses are characterized by the fuel cell efficiency (η_{fc}), resulting in a thermal load ($\dot{Q}$).

$$\dot{Q} = \left(\frac{1 - \eta_{fc}}{\eta_{fc}} \right) \dot{W}_{fc} \quad (65)$$

We have applied this fuel cell performance model to the CHEETA power system based on high-temperature PEMFC forecasts described in Waddington et al. (Ref. 300), with our projection of stack specific power based on these forecasts and industry press releases (Refs. 337 to 339). The model parameters and resulting cell performance are presented in Fig. 324, which shows cell voltage, efficiency, and power density at varied current density for a PEMFC power system operating at an altitude of 37,000 ft. The y-intercept of cell voltage is set by the fuel crossover losses, which causes a voltage at idle that is less than the open-circuit reference voltage. The nonlinearities at low current densities are due to activation losses. The linear portion of the curve is driven by ohmic losses, up until the precipitous decrease in voltage due to nonideal mass transport (Ref. 336). The plot shows that fuel cell efficiency decreases with increasing power.

5.1.2.1.2 Ducted Heat Exchanger Thermal System

To utilize fuel cell waste heat necessitates a ducted radiator HEX. We have quantified the HEX performance, following the methodology of Kays and London (Ref. 334), using the effectiveness (ϵ) of a HEX based on the number of heat transfer units (N_{TU}). For the thermal systems considered, we assume polyethylene glycol to be the hot working fluid (Ref. 340) and ambient air to be the cold fluid.

To quantify the drag of such a HEX in isolation, we have used the combined heat transfer and internal flow analysis presented by Drela (Ref. 335). The HEX core drag contribution (D_{core}) refers to only the net drag of the core flow, defined as the airflow captured in the nacelle, sent through the HEX, and expelled through the outlet, as in Fig. 325. The net core flow force contribution includes effects of friction and heat addition. Friction manifests as a pressure drop over the HEX. The deposition of heat into the working fluid produces a density change and an acceleration. If the net drag force is negative, it will be referred to as a “HEX thrust.”

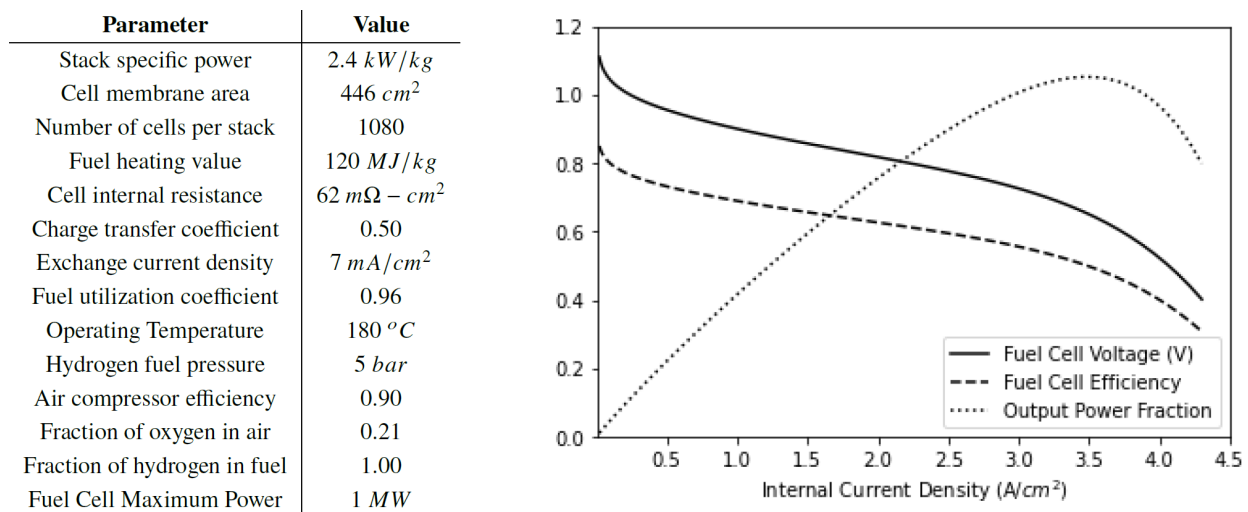


Fig. 324.—Fuel cell parameters and performance are shown for varied current density to highlight the impact of the various loss mechanisms for the CHEETA concept vehicle fuel cell system.

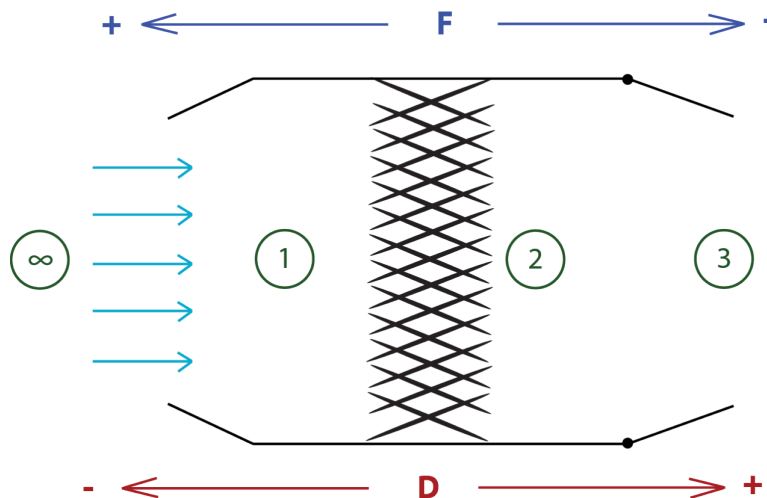


Fig. 325.—Notional 1D flow representation of an isolated ducted HEX.

The drag strongly depends on the HEX face velocity, u_1 , which is controlled by a variable-area exhaust, based on the mass flow rate required for cooling the fuel cell stack. We can characterize HEX performance and drag based on the empirical HEX effectiveness (ϵ) curves provided in Reference 334. It appears desirable to increase the HEX frontal area (A_r) to slow down the flow and reduce core drag, but this change creates increased drag on the duct inlet surface, which we call nacelle drag. This nacelle drag can be quantified using an empirical drag coefficient (C_d) for a specified flight condition (Ref. 341); we use a C_d of 0.04 at all conditions.

The use of a HEX also entails a weight addition. We assume the weight scales linearly with face area as given by Boeing (Ref. 315) to account for the HEX, plumbing, valves, and coolant circulation pumps. The added weight requires additional lift, thus an associated drag which we denote as weight drag, and its contribution is small relative to the force contributions of the core and nacelle. The sum of the drag contributions results in a drag power which is used to quantify the total thermal system drag at a given flight velocity. The optimal system is achieved by balancing the different drag sources with the thrust requirement at the cruise point.

5.1.2.2 Integrated Thermal/Propulsion Systems

The isolated HEX is only one of several configurations that could be installed, and in this section, we examine others using an incompressible 1D flow model, as described in Reference 319. Three integrated configurations are shown in Fig. 326. The intent is to assess the impact of varied HEX face area as well as different design choices (e.g., fan before HEX, HEX before fan, multiple flow paths) on system performance to inform future work toward an optimal CHEETA TMS. In the following subsections, we consider integrated thermal propulsors in series (fan and HEX share a flow path) and in parallel (fan and HEX have separate flow paths) in the context of the baseline and isolated HEX configurations.

5.1.2.2.1 Series-Integrated Thermal Propulsor

Fig. 326(a) and (b) show configurations with a fan and HEX integrated in the same duct with a shared flow path. Potential benefits include a reduced nacelle wetted area for decreased nacelle drag, attenuated inlet flow distortion through the screenlike influence of a HEX upstream of the fan, and augmented thrust due to heat addition.

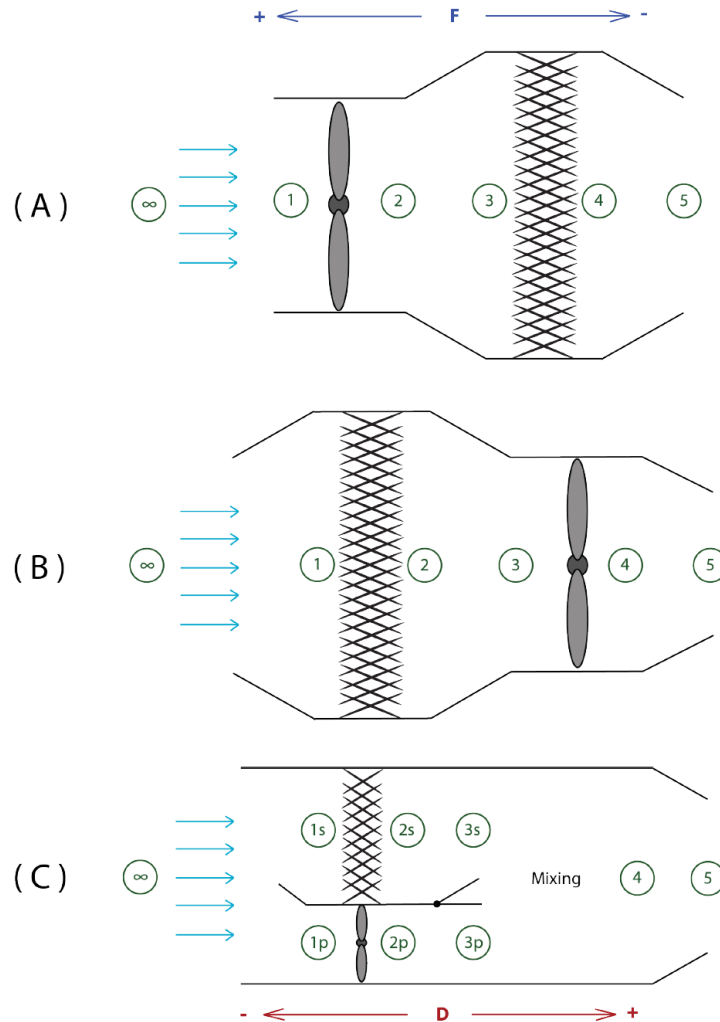


Fig. 326.—Notional 1D flow representations of (a) fan-HEX series-integrated, (b) HEX-fan series-integrated, and (c) parallel-integrated thermal propulsors, where series-integrated entails shared thermal/propulsion flow path and parallel-integrated entails split flow path with secondary cooling flow entrained by primary fan flow.

We assessed the impact of the HEX-fan area ratio (α) on the axial force coefficient (C_x) and power coefficient (K_p) of the series-integrated propulsors in Fig. 326(a) and Fig. 326(b). The HEX effectiveness was set as $\epsilon = 0.45$, the nondimensional HEX pressure drop ($\Phi = 20$) based on the upper range of typical devices (Ref. 335), the radiator temperature ($T_r = 180$ °C) based on high-temperature fuel cell operation (Ref. 330), and the nacelle drag coefficient ($C_d = 0.04$) based on experiments by Robinson et al. (Ref. 341).

Fig. 327 shows nondimensional net thrust per unit power, analogous to propulsive efficiency, for a range of α -values along with cruise and top-of-climb design points for the baseline and isolated configurations. For all α , the series-integrated thermal propulsor performs worse than an unintegrated HEX and propulsor. The cruise K_p of the baseline and isolated configurations is only enough to overcome the drag of the series-integrated configuration, and the integrated system requires twice the power of an isolated configuration to achieve the same net thrust. We also see that the power required to achieve a certain thrust is slightly higher for an upstream HEX. Due to the reduction in air density, because of temperature rise from heat addition, the fan requires a larger input work for a specific pressure rise.

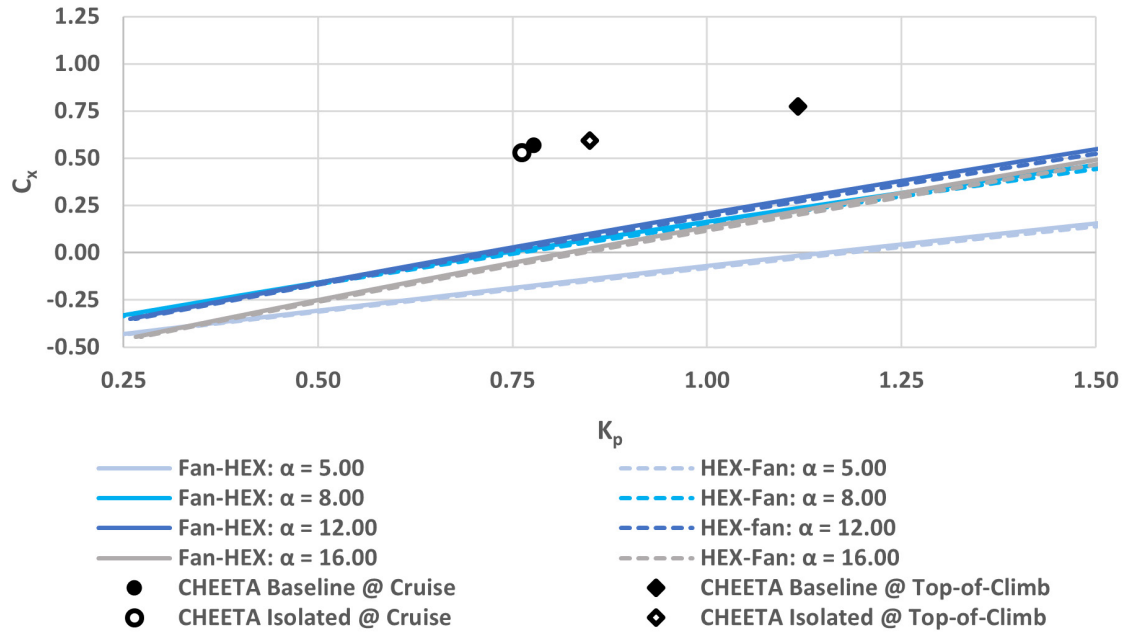


Fig. 327.— Nondimensional net thrust per power for series-integrated configurations compared to baseline and isolated configurations.

5.1.2.2.2 Parallel-Integrated Thermal Propulsor

Another thermal system is a parallel-integrated configuration, which has a primary (p) fan stream and a secondary HEX stream (s), as shown in the 1D flow diagram in Fig. 326(c). In practice, the primary and secondary streams would be arranged annularly within the duct, so the secondary stream is entrained by the primary stream of flow over the fan. We need to ensure enough air flows through the secondary stream to cool the fuel cell at any operating condition, and this is done using variable internal geometry where the secondary stream exhausts to the mixing chamber.

As with the series-integration configuration, we assessed the impact of the HEX-fan area ratio α on the axial force coefficient C_x and power coefficient K_p using the following assumptions: stack efficiency $\eta_{fc} = 0.60$; HEX effectiveness $\epsilon = 0.60$; nondimensional HEX pressure drop $\Phi = 20$, based on the upper range of typical devices (Ref. 335); radiator temperature $T_r = 180^\circ\text{C}$, based on fuel cell data (Ref. 330); and nacelle drag coefficient $C_d = 0.04$, based on experiments by Robinson et al. (Ref. 341).

In Fig. 328, we see C_x begin to decrease beyond a certain K_p for low α -values ($\alpha \leq 2.00$, for the range of K_p considered) due to the larger thermal load at high-power conditions, which necessitate higher secondary flow rate. For these low- α cases, the elevated flow rate corresponds to high flow velocity, thus increasing pressure drop and therefore core drags. For large α , HEX pressure drop is reduced, lessening the core drag and increasing the range of useful K_p . As with the series-integrated configuration, the external nacelle drag serves as an upper bound, limiting thrust-producing configurations to $\alpha \leq 16.00$ for this application.

The optimal α for the parallel-integrated thermal propulsor is between 0.5 and 2.0, depending on the stack efficiency, cruise power required, and thrust required at top-of-climb. Relative to the series-integrated propulsor, these results highlight the importance of limiting airflow over the HEX to only what is required for cooling.

This configuration performs comparably to the isolated configuration, which is unsurprising given that the only practical difference is the mixing of primary and secondary flows in the parallel case. The optimal α -values for the parallel-integrated propulsor are slightly larger than those of the isolated configuration, primarily due to our conservative estimate of nondimensional pressure drop, taking the upper range of typical values ($5 \leq \Phi \leq 20$) (Ref. 335). Considering compressibility effects, the air density would be slightly higher throughout the system due to the ram effect, enabling a reduction in HEX face area and work required from the fan. The results indicate that a parallel-integrated thermal propulsor is feasible and that a compressible assessment is required to fully characterize the performance of such a system.

5.1.2.3 Aircraft Thermal Management Integration: Conclusions

The reduction in thermal system drag due to heat addition reduced overall power requirements at the sizing point (top-of-climb), enabling a reduction in power system size. In other words, decreased stack efficiency caused an increase in propulsive efficiency. This coupled effect enables an aircraft with lower MTOW, power requirements, and fuel consumption for the same mission relative to the baseline aircraft, indicating that the propulsive contribution of the thermal system must be considered when sizing the power system. Thermal system design for MW-scale electric aircraft thus cannot be considered as a simple drag penalty to be quantified after preliminary design is complete; power, propulsion, and thermal systems must be designed together.

Our assessment shows the benefit of placing the HEX downstream of the fan as well as the drag penalties of passing excessive flow over the HEX. Our inference is that an optimal thermal system design would be a bypass configuration, as in Fig. 329, with variable internal geometry to control the entrained secondary airflow rate. Future work should focus on the development of a higher fidelity flow model to be incorporated into our aircraft sizing model and an aircraft design optimization framework.

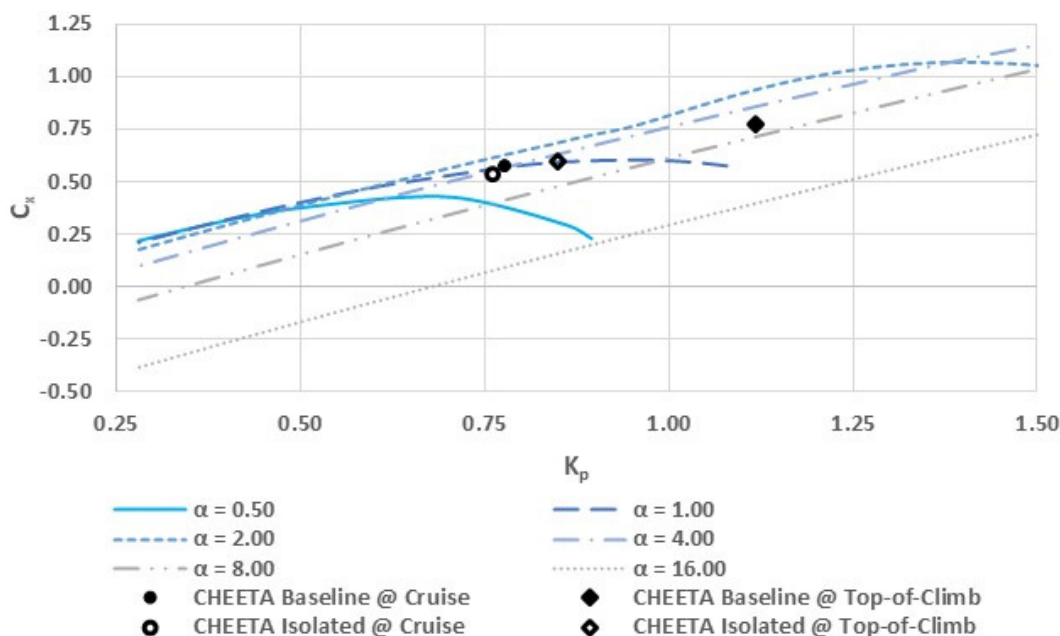


Fig. 328.—Axial force coefficient is plotted versus power coefficient for a parallel-integrated thermal propulsor. The range of nondimensional power at a specific HEX-fan area ratio (α) is limited by the airflow the HEX can pass before core drag becomes excessive. This range can be overly restrictive for low area ratios ($\alpha \leq 1.00$).

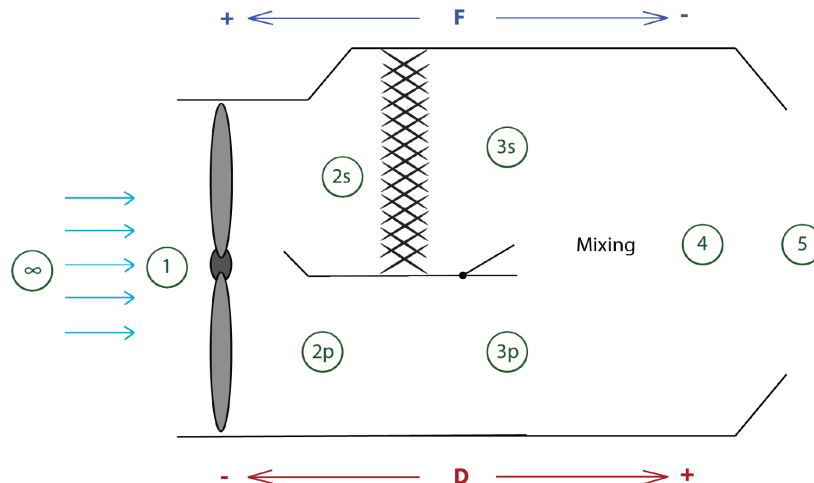


Fig. 329.—Notional 1D flow representation of bypass-integrated thermal propulsor.

5.1.3 H₂ Propulsion Trade Space Assessment

The system integration studies performed under the CHEETA program successfully provided a conceptual aircraft design for 2050 entry into service, leveraging advanced technology, that produces zero direct CO₂ and NO_x emissions from the aircraft. However, it was also recognized that alternate power system architectures may also provide further improvements in energy capability when utilizing H₂ energy. To further understand the impact of alternate H₂ propulsion architectures on the energy requirements of the given CHEETA concept mission, a trade space assessment was considered.

Meeting global climate goals (Ref. 342) will require major advances in the aviation industry, such as those described in a recent National Academies report (Ref. 101) that defined high-priority approaches for greening aviation, including (1) propulsion-airframe integration, (2) improved gas turbine engines, and (3) electrified propulsion systems. We will discuss and assess these approaches in the context of the H₂-fueled, electrified CHEETA aircraft concept.

5.1.3.1.1 Propulsion-Airframe Integration

Propulsion-airframe integration refers to the interactions between the thrust-producing devices and the body of the aircraft (Ref. 101). Today, turbofan engines are mounted sufficiently far from the airframe that the two can be considered, at least conceptually, in isolation. There is interest, however, in more integrated configurations where this is not the case, such as designs in which the propulsor ingests and reaccelerates a portion of the airframe boundary layer, augmenting the propulsive efficiency and reducing fuel burn (Ref. 343). Hall et al. quantified the aerodynamic benefit of this phenomenon, referred to as “boundary-layer ingestion” (BLI), for the D8 aircraft concept, which showed a 9% reduction in propulsor mechanical power relative to a non-BLI baseline due to aerodynamics alone (Ref. 11). In the present work, we quantify the performance benefit from BLI using the power balance method (Refs. 64 and 344), as in Reference 11.

5.1.3.1.2 Improved Gas Turbine Engines

Gas turbine engine overall performance is characterized by the product of thermal and propulsive efficiency (Ref. 101). The former is limited by engine operating temperature, which is constrained by material and cooling limitations (Ref. 101). The latter can be increased by decreasing fan pressure ratio (Ref. 101) or via propulsion-airframe integration. High-bypass-ratio turbofan engines, which achieve thermal efficiencies up to 55% and propulsive efficiencies up to 70% (Ref. 345), serve as a baseline with which we compare electrified propulsion systems.

5.1.3.1.3 Electrified Propulsion Systems

Turboelectric systems use an electric generator to convert shaft power from a gas turbine engine into electrical power, which is transmitted to motor-driven fans (Ref. 101). Reductions in fuel consumption may be possible through DP enabled by such systems, such as Perullo et al. for a regional turboelectric jet (Ref. 346). Despite wide use in the automotive industry, battery energy storage has been shown to be impractical for air transport due to insufficient specific energy (Refs. 347 and 348), and transport aircraft will require on-board energy conversion. In this context, fuel cells present an opportunity for energy conversion with zero CO₂, NO_x, or particulate matter emissions (Ref. 349). PEMFCs produce electricity by combining H₂ fuel and ambient O₂ over a catalytic membrane to produce electricity and water (Ref. 336). PEMFCs were selected for examination based on previous work (Ref. 92). They have high specific power and technological readiness relative to other types of fuel cells (Refs. 337 to 339).

5.1.3.1.4 Propulsion System Model

In this section, we present the multiple propulsion system configurations that were examined and the optimization framework used to compare them. This trade-space analysis (Ref. 350) has been carried out in a manner similar to that of Hall et al. (Ref. 347), who assessed the electrified aircraft trade-space in the context of battery-electric propulsion systems. Snyder et al. (Ref. 349) presented a similar assessment for H₂-fueled aircraft, including gas turbine and fuel cell power systems. We expand upon this previous work by (1) reassessing fuel-cell-powered aircraft for 2050 technology projections and (2) considering additional electrified systems, such as turboelectric and hybrid-electric aircraft.

We seek an aircraft that minimizes the fuel required for a given mission (range, payload, and cruise velocity). We perform this minimization using a modular optimization framework called GPkit (Ref. 57). GPkit is a Python toolkit that minimizes a cost within the bounds of constraints that are specified to capture the physical behavior of the system and its components. GPkit has been used in previous aircraft design efforts (Refs. 55, 58, 347, 351, and 352), primarily due to its speed even with a large number of constraints, making it attractive for aircraft design optimization.

Illustrated in Fig. 330 are the four classes of propulsion systems we assessed for the CHEETA aircraft: turbofan, turboelectric, fully electric, and hybrid-electric. The various blocks give the major subsystems employed within each class of propulsion system. In the figure, a solid arrow denotes power transmission and a dotted arrow denotes heat transmission.

- Turbofan systems employ a gas turbine to generate shaft power to drive the fans.
- Turboelectric systems use a generator to convert turbine shaft power into electrical power, which is distributed to motor-driven fans.
- Fully electric systems also employ distributed, motor-driven fans but utilize fuel cells to generate electrical power directly. The fuel cells are cooled by ducted HEXs.
- Hybrid-electric systems employ a combination of powertrains (i.e., gas turbine + fuel cell, fuel cell + battery) in parallel.

5.1.3.1.5 Fuel Cells

Rather than converting chemical energy into mechanical shaft power, then electrical power via a generator, as in a turboelectric system, a fuel cell performs the electrochemical energy conversion directly. In PEMFCs, this conversion occurs by separating the proton and electron of the H₂ atom at the surface of a catalytic membrane; the protons pass through the membrane while the electrons pass through the circuit, ultimately combining with O₂ at the cathode to form water (Ref. 336). The overall reaction is exothermic, and the resulting waste heat is assumed to be managed by a ducted HEX (Ref. 319).

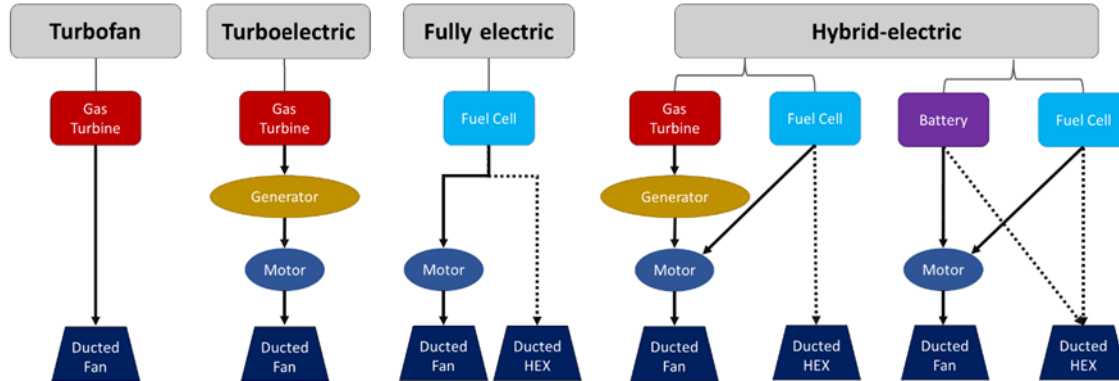


Fig. 330.—Four classes of propulsion systems assessed for the CHEETA aircraft: turbofan, turboelectric, fully electric, and hybrid-electric. Solid and dotted arrows denote power and heat transmission, respectively.

5.1.3.1.6 Ducted Heat Exchangers

Unlike gas turbines, where waste heat is expelled through the exhaust flow, fuel cells require a TMS to transport waste heat to a flow path for rejection (Ref. 336). Such TMSs can include liquid-air HEXs (or radiators), which add weight and surface area (i.e., drag sources) on the aircraft (Ref. 335), thereby decreasing performance. We analyze the thermal system performance of a zero-drag HEX concept, which uses the waste heat to produce a thrust contribution across a ducted radiator. This phenomenon, also known as the Meredith effect (Ref. 333), is analogous to a ramjet, where a change in air density due to heat addition causes flow acceleration and thrust production in a constant area duct. In this assessment, we draw heavily from the combined heat transfer and internal flow analysis presented by Drela (Ref. 335), which we incorporate in a 1D incompressible flow model of a ducted HEX (Ref. 319).

5.1.3.1.7 Gas Turbines, Motors, and Generators

We have sized the gas turbine engine by linearly relating engine weight and maximum power based on published data for the CFM-56 (Ref. 353). The performance of the engine core is taken from thermal efficiency projections of high-bypass-ratio engines (Ref. 345). The motors and generators are sized based on specific torque scaling factors, and the performance is quantified using published efficiency values: 20 N-m/kg and $\eta = 0.95$ for conventional motors (Ref. 354), 20 N-m/kg and $\eta = 0.98$ for generators (Ref. 355), and 70 N-m/kg and $\eta = 0.999$ for SC motors (Ref. 354). The SC motors and electrical transmission lines are cooled by the cryogenic H_2 fuel (Ref. 64).

5.1.3.1.8 Battery

The specific energy of batteries makes them insufficient as the sole power source for commercial transport aircraft (Ref. 348). However, hybrid-electric systems could take advantage of batteries during high-power conditions to supplement a gas turbine, as considered by Decerio et al. (Ref. 352) for a battery-turboprop hybrid regional aircraft. Given that fuel cell efficiency scales inversely with output power, a battery could supplement a fuel cell system to improve efficiency, and we consider this effect for a hybrid-electric configuration, as in Fig. 330. Other uses for batteries, such as transiently supplementing fuel cells during changes in throttle setting, are beyond the scope of this work.

5.1.3.1.9 Ducted Propulsors

The ducted propulsor (fan) converts mechanical shaft power ($\dot{W}_{shaft}$) into mechanical flow power ($\dot{W}_K$). We assessed the performance of a propulsor using an incompressible 1D flow model (Ref. 350). To estimate propulsor weight, we assumed half the weight of a jet engine is from the fan/nacelle and that this weight scales with the cube of d_{fan} . We estimated the benefit of BLI using the power balance method (Refs. 9 and 344).

5.1.3.2 Results

We compared the performance of several alternate propulsion systems for the CHEETA aircraft using the payload-fuel energy intensity (PFEI), the fuel energy required per the product of payload mass and range (Ref. 356). Use of PFEI allows a direct comparison of conventional aviation turbine fuel (ATF) and LH₂ fuel for each propulsion system configuration. PFEI is defined as

$$\text{PFEI} = \frac{(\text{Fuel Heating Value})(\text{Fuel Mass})}{(\text{Payload Mass})(\text{Range})} \quad (66)$$

A fully electric transport aircraft such as the CHEETA concept vehicle introduces several key technologies, which we highlight in the morphing chart in Fig. 331. This chart begins with an ATF-burning, non-BLI, twin-turbofan baseline aircraft, based on the 737-800 (Ref. 299) and projected efficiency improvements for advanced gas turbine engines (Ref. 345). Proceeding vertically, we apply each technological change sequentially. The first step is introducing BLI, which enables a 13% decrease in PFEI. In the second step, we switch to LH₂ fuel, which decreases PFEI by 24% relative to the baseline while eliminating in-flight CO₂ emissions. This performance benefit is partially negated in step three by the addition of two electric generators and nine motor-driven propulsors, which add weight and introduce additional losses. SC motors can mitigate these penalties by reducing motor weight and increasing efficiency, as in step four. Steps five and six show that electrifying the power system monotonically increases the required fuel energy, as a larger fraction of total power is produced by fuel cells rather than a gas turbine, indicating performance can be traded for NO_x emission reductions. The PFEI of the fully electric configuration (PFEI = 8.0) is 4% higher than that of the baseline (PFEI = 7.7). In the final step, batteries are added to supplement the fuel cells at high-power conditions (i.e., top of climb). While fuel cell efficiency improved at this condition, the additional weight increased power required during other phases of flight, causing an increase in PFEI.

We also considered the impact of key parameters on aircraft performance for the 180-passenger, 5,500-km mission, as in Fig. 332. The projected performance of each configuration is plotted along with conservative and optimistic scenarios for gas turbine thermal efficiency and fuel cell specific power. Each configuration plotted employs BLI. Fig. 332 shows that a H₂-burning turbofan aircraft requires the least fuel energy for the given mission, and that eliminating NO_x emissions using fuel cells comes with higher PFEI. There is also a monotonic increase in required fuel energy, as more power is produced using fuel cells rather than gas turbines, as observed for the turbo-, hybrid-, and fully electric systems.

The projected parameters, as well as conservative and optimistic estimates, are given in Table 51. The range of gas turbine thermal efficiencies are based on projections for high-bypass-ratio turbofan engines (Ref. 345). Values for motor specific torque are based on the review by Haran et al. (Ref. 354). The lower bound of specific power (2 kW/kg) is from tests conducted by Hypoint in 2021 (Ref. 337), and the upper bound (3 kW/kg) is their 2025 projection for a high-temperature PEMFC system (Ref. 339).

5.1.3.3 Aircraft Trade Space Exploration: Conclusions

Using a system optimization framework, we have assessed the performance of several propulsion system configurations for civil air transport: turbofan, turboelectric, hybrid-electric, and fully electric. We considered both ATF and LH₂ fuel in the context of a 5,500-km, 180-passenger mission. Performance was quantified using the PFEI, defined as the fuel energy required per product of range and payload. Key enabling technologies for future H₂-electric aircraft include (1) HT-PEMFCs, (2) SC motors and electrical transmission, (3) LH₂ storage tanks, (4) BLI, and (5) DEP.

Employing each of these enabling technologies, a fully electric, fuel-cell-powered propulsion system (PFEI = 8.0) performs close to a conventional ATF-burning turbofan system (PFEI = 7.7), and a H₂-

burning turbofan system with BLI performs best (PFEI = 5.9). Replacing turbofan engines with electric generators and motor-driven fans comes with weight/performance penalties, which are largely mitigated for SC motors. For a hybrid-electric configuration, increasing the ratio of fuel cell to gas turbine power capacity monotonically increases aircraft weight and PFEI. Employing a battery to supplement fuel cells at high-power conditions (takeoff, top of climb) does not improve performance for the assessed payload and range. The same result was observed for a fuel cell supplementing gas turbines.

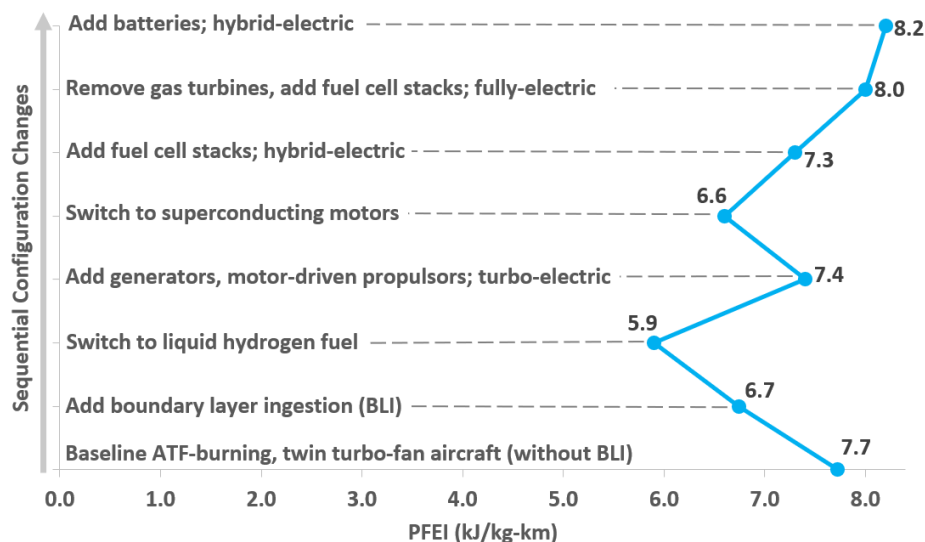


Fig. 331.—Morphing chart depicting aircraft performance for sequential changes to an ATF-burning, non-BLI, twin-turbofan baseline aircraft

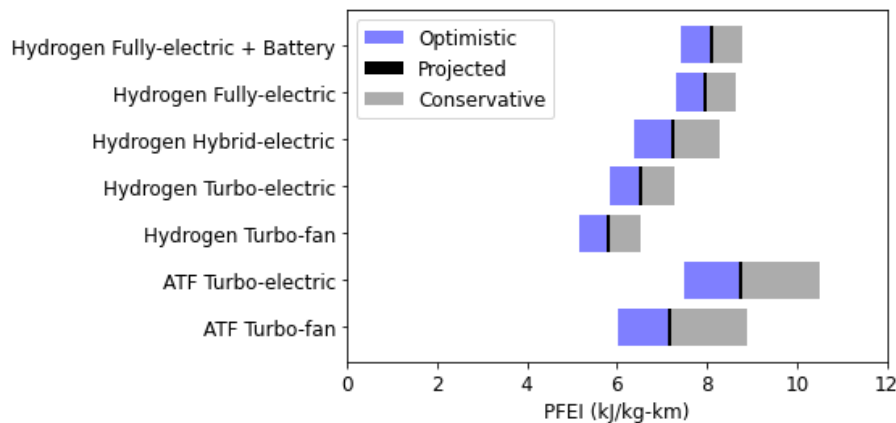


Fig. 332.—Payload-fuel energy intensity for each ATF and H₂-fueled propulsion system. Conservative and optimistic estimates are depicted in gray and blue ranges, respectively. Each system shown employs BLI.

TABLE 51.—PROJECTIONS OF KEY PARAMETERS

	Conservative	Projected	Optimistic
Gas turbine thermal efficiency	0.50	0.55	0.60
PEMFC stack specific power, kW/kg	2.0	2.4	3.0
Conventional motor specific torque, N-m/kg		20	
SC motor specific torque, N-m/kg		70	

Appendix—Acronyms

1D	one dimensional
2D	two dimensional
2L-CSI	two-level current source inverter
3D	three dimensional
3L-CSI	three-level current source inverter
AC	alternating current
ACR	alternating-current resistance
AD	actuator disk
AGARD	Advisory Group for Aerospace Research and Development
ANPC	active neutral-point clamped
APS	auxiliary power supply
ARPA-E	Advanced Research Projects Agency-Energy
ATF	aviation turbine fuel
BiCMOS	bipolar complementary metal-oxide-semiconductor
BJ	bipolar junction transistor
BLDC	brushless direct current
BLI	boundary-layer ingesting/ingestion
BOL	beginning of life
CABLES	Connecting Aviation By Lighter Electrical Systems
CAD	computer-aided design
CAPS	cryogenic auxiliary power supply
CC	cable cryostat
CERN	European Organization for Nuclear Research (Conseil Européen pour la Recherche Nucléaire)
CFD	computational fluid dynamics
CFRP	carbon-fiber-reinforced polymer
CHEETA	Center for High-Efficiency Electrical Technologies for Aircraft
CM	common mode
CMOS	complementary metal-oxide-semiconductor
CMTI	common mode transient immunity
CNT	carbon nanotube
CORC [®]	Conductor on Round Core
COTS	commercial off-the-shelf
CPC	cast polycarbonate
CPEC	cryogenic power electronics converter
CPU	central processing unit
CR	polychloroprene
CRM	Common Research Model
CS	common source
CSI	current source inverter
CSMM	Center for Superconducting and Magnetic Materials
CT	cryogenic temperature
CTE	coefficient of thermal contraction

DAQ	data acquisition
DC	direct current
DCR	direct-current resistance
DEP	distributed electric propulsion
DMOS	double-diffused metal-oxide-semiconductor
DNB	departure nucleate boiling
DOE	Department of Energy
DP	distributed propulsion
DPT	double pulse test
ECS	environmental control system
EDLC	electric double-layer capacitor
EMCB	electro-mechanical circuit breaker
EMI	electromagnetic interference
EOL	end of life
EPC	equivalent parallel capacitance
EPDM	ethylene propylene diene monomer
EPR	ethylene-propylene-rubber
ESR	equivalent series resistance
ETFE	ethylene tetrafluoroethylene
EUT	equipment under test
EVA	ethylene-vinyl-acetate
EWIS	electrical wiring interconnection system
FAR	Federal Aviation Regulations
FCL	fault current limiter
FEA	finite-element analysis
FEGV	fan exit guide vane
FEM	finite-element method
FEP	fluorene-ethylene-propylene
FET	field-effect transistor
FFD	fast fluid dynamics
FOM	figure of merit
FR	flame retardant
GA	genetic algorithm
GD	gate driver
GE	General Electric
GP	geometric program
GRP	glass-reinforced plastic
GSD	gravimetric storage density
HCB	hybrid circuit breaker
HDPE	high-density polyethylene
HEMT	high-electron-mobility transistor
HEX	heat exchanger
HFC	high flux core
HHV	higher heating value
HIRENASD	High Reynolds Number Aero-Structural Dynamics

HPAL	high-purity AL
HT-PEMFC	high-temperature proton-exchange membrane fuel cell
HTC	heat transfer coefficient
HTPEM	high-temperature proton-exchange membrane
HTS	high-temperature superconductor/superconducting
IC	integrated circuit
ICP	intrinsically coupled polymers
IGBT	insulated-gate bipolar transistor
JFET	junction-gate field-effect transistor
LDPE	low-density polyethylene
LHV	lower heating value
LiB	lithium-ion battery
LiC	lithium capacitor
LISN	line impedance stabilization network
LPF	low-pass filter
LT-PEMFC	low-temperature proton-exchange membrane fuel cell
LTO	landing and takeoff
LTPEM	low-temperature proton exchange membrane
MAC	mean aerodynamic chord
MDO	multidisciplinary design optimization
MEA	more-electric aircraft
MKI, II, III	mark I, II, III
MLI	multilayer insulation
MMEI	micro-multilayer multifunctional electrical insulation
MOSFET	metal-oxide-semiconductor field-effect transistor
MPP	molypermalloy powder
MPZ	minimum propagating zone
MQE	minimum quench energy
MRI	magnetic resonance imaging
MRL	manufacturing readiness level
MTOW	maximum takeoff weight
MWF	mass weight fraction
NP0	negative-positive-zero
NPSS	Numerical Propulsion System Simulation
NR	natural rubber
NSGA	Non-dominated Sorting Genetic Algorithm
NZPV	normal zone propagation velocity
OD	outer diameter
OEI	one engine inoperative
OML	outer mold line
P2P	pumped two-phase
PA	polyamide
PBI	phosphoric acid doped polybenzimidazole
PCB	printed circuit board
PEEK	polyether-ether-ketone

PEMFC	proton-exchange membrane fuel cell
PET	polyethylene-terephthalate
PFA	perfluoroalkoxy
PFEI	payload-fuel energy intensity
PFSA	perfluorosulfonic acid
PI	polyimide
PID	proportional–integral–derivative
PM	permanent magnet
PP	polypropylene
PPS	polypropylene sulfide
PS	polystyrene
PTFE	polytetrafluoroethylene
PUR	polyurethane
PVC	polyvinylchloride
PWM	pulse-width modulated or modulation
R&D	research and development
RANS	Reynolds-averaged Navier-Stokes
RC	resistor capacitor
RCD	resistor-capacitor-diode
REBCO	rare-earth barium copper oxide
RPM	revolutions per minute
RRR	residual-resistivity ratio
RT	room temperature
RVB	reverse voltage blocking
SAND	simultaneous analysis and design
SBR	styrene-butadiene rubber
SC	superconducting
SFCL	superconductive fault current limiter
SIR	silicone rubber
SMA	subminiature version A
SMCI	superconducting momentary circuit interrupter
SMES	superconducting magnetic energy storage
SOFC	solid oxide fuel cell
SP	signomial programming
SPF	silicide polysilicon fuse
SPT	stereo pattern tracking
SS-AC	solid-state alternating current
SS-DC	solid-state direct current
SSCB	solid-state circuit breaker
SSFCL	solid-state fault current limiter
STARC-ABL	Single-Aisle Turboelectric Aircraft with Aft Boundary-Layer Propulsion
STOL	short takeoff and landing
SUSAN	SUBsonic Single Aft eNginE
SVM	space vector modulation
SWaP	size, weight, and power

TBD	to be determined
THD	total harmonic distortion
TIM	thermal interface material
TMS	thermal management system
TRL	technology readiness level
UIUC	University of Illinois, Champaign-Urbana
ULI	University Leadership Initiative
VDE	Verband der Elektrotechnik, Elektronik und Informationstechnik
VJ	vacuum jacketed
VPI	vacuum pressure impregnation
VSI	voltage source inverter
VTOL	vertical takeoff and landing
WBG	wide bandgap
WHR	waste heat recovery
WIPS	wing ice protection system
XLPE	cross-linked polyethylene

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