

Preliminary Design of Hybrid Aircraft Thermal Management System (Tms) Experimental Test Campaign in Cira Icing Wind Tunnel (IWT)

Marco Invigorito

CIRA - Italian Aerospace Research Centre

Michelangelo Serpico

CIRA - Italian Aerospace Research Centre

Umberto Mercurio

CIRA - Italian Aerospace Research Centre

Carmine Caputo

CIRA - Italian Aerospace Research Centre

Gianpaolo Elia

CIRA - Italian Aerospace Research Centre

Roberto Scigliano

`r.scigliano@cira.it`

CIRA - Italian Aerospace Research Centre

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Abstract

Within the EU-funded AMBER project [1], CIRA (Italian Aerospace Research Centre) is responsible for the experimental validation of the Thermal Management System (TMS) for a hybrid aircraft. To support this activity, CIRA is upgrading its Icing Wind Tunnel (IWT) [8], taking advantage of its large test section and its ability to reproduce realistic flight conditions, including altitude, temperature, and mass flow rates ranging from 1.5 to 55 kg/s. The upgraded facility will enable full-scale testing of TMS units with power levels up to approximately 0.5 MW, providing realistic operating conditions for accurate performance evaluation and optimization. This paper presents the ongoing upgrades to the IWT, including the required equipment, subsystems, and experimental requirements for TMS characterization. A preliminary numerical performance characterization of a realistic TMS for an hybrid aircraft has been performed leading to a better definition of the facility operating envelope. Once completed, the upgraded IWT will provide a unique capability for testing thermal management systems under representative hybrid aircraft operating conditions, supporting the development and validation of advanced thermal performance solutions.

INTRODUCTION

Achieving the climate targets set by the Paris Agreement and the EU Green Deal requires the aviation sector to adopt cleaner and more sustainable technologies. Although aviation currently accounts for around 2.5% of global CO₂ emissions [9], its overall climate impact is even greater due to non-CO₂ effects, such as nitrogen oxide emissions [10] and contrail formation [11]. As air traffic continues to increase, hybrid-electric and all-electric propulsion systems are emerging as promising solutions for reducing emissions, particularly on short-haul and regional routes. In line with this vision, the aviation industry aims to achieve net-zero carbon emissions by 2050, with Europe targeting a significant share of intra-EU flights powered by hydrogen or electric propulsion.

The transition to electrified propulsion, however, introduces major thermal management challenges. High-power components—including batteries, electric motors, fuel cells, and power electronics—generate significant heat and require efficient cooling to ensure safe and reliable operation. In hybrid aircraft, the coexistence of conventional and electric propulsion systems creates complex and highly variable thermal loads, demanding advanced Thermal Management Systems (TMS) capable of maintaining optimal operating temperatures under all flight conditions [12][13][14][15].

Since thermal management directly affects aircraft safety, efficiency, and component lifetime, experimental validation is essential. While numerical simulations support the design process, they must be complemented by experimental testing under representative operating conditions to accurately assess TMS performance and validate design methodologies. This paper first introduces a numerical heat exchanger characterization and secondly presents the experimental facility upgrade.

1. METHODOLOGY AND EXPERIMENTAL FRAMEWORK

The thermal management of next-generation electrified aircraft requires a precise understanding of heat exchanger (HEX) performance across highly variable environmental conditions. This study employs a hybrid

analytical-experimental approach to characterize a liquid-to-air compact heat exchanger intended for fuel cell cooling.

1.1 Baseline Characterization and numerical check methodology

The first phase of the study focused on establishing a high-fidelity baseline using manufacturer datasheet specifications for a 300 kW nominal capacity unit.

To define the "virtual twin" of the heat exchanger, the global reference transmittance (UA_{ref}) was extracted by relating the nominal heat rate (Q) to the air and coolant temperature streams. The heat transfer was modeled using the Log Mean Temperature Difference (LMTD) method [1], [3], where the baseline UA_{ref} is expressed in (1).

$$UA_{ref} = \frac{Q}{F * \Delta T_{lm}} = 11.9 \text{ kW/K}$$

1

In this formulation, ΔT_{lm} represents the log-mean temperature difference for an equivalent counter-flow arrangement. To account for the specific single-pass cross-flow geometry of the real unit, a correction factor $F = 0.97$ was applied. This factor was determined by interpolating the standard performance charts for unmixed crossflow configurations provided in the seminal work of Kays and London [3], based on the heat capacity rate ratio and the exchanger effectiveness.

To ensure the reliability of this extraction, the model was cross-validated using the effectiveness-number of transfer units (ϵ -NTU) method [1], [2]. For unmixed geometries, the analytical effectiveness was calculated as $\epsilon = 0.48$, which successfully closed the thermal balance at 300.6 kW, confirming the accuracy of the 11.9 kW/K baseline UA_{ref} .

1.2 Flight mission profile and kinematic states

To evaluate the HEX under realistic operational stresses, we integrated a regional route flight profile [4]. Three distinct mission points were selected to represent the core operating envelope:

1. Ground Baseline

- Aircraft Speed $\approx 35 \text{ m/s}$ $\rightarrow$ Mean air core velocity: 10 m/s (*corresponding to design conditions under an air-side pressure drop of 1500 Pa*);

2. Climb Transition (2000 m)

- Aircraft Speed $\approx 70 \text{ m/s}$ $\rightarrow$ Mean air core velocity: 18 m/s (*Altitude transient state with an air-side pressure drop of 3900 Pa*);

3. Top Climb (5500 m)

- Aircraft Speed ≈ 92 m/s $\rightarrow$ Mean air core velocity: 28 m/s (*Altitude transient state with an air-side pressure drop of 6900 Pa*)

A key challenge in this analysis was defining the relationship between the aircraft's freestream velocity and the actual air velocity through the HEX core. For this study, an aerodynamic coupling was established based on the design point specifications. At the ground baseline, it was determined that a freestream velocity of 35 m/s (representative of the aircraft's initial kinematic state) induces a mean core velocity of 10 m/s, resulting in a design pressure drop of 1500 Pa. This relationship accounts for the installation effects and the flow resistance of the ducting system. The kinematic states for the subsequent mission points (Point 2 and Point 3) were then derived by scaling the internal flow conditions relative to the aircraft airspeed. The corresponding air-side pressure drops for the "Climb Transition" and "Top Climb" conditions were subsequently resolved as a function of these calculated core velocities and the varying air density at altitude. This approach allows for a computationally efficient estimation of the mass flow rate required for the convective re-scaling methodology, while capturing the essential physics of the ram-air effect across the flight envelope.

1.3 Aerothermal scaling and Colburn power Law

The transition from ground to high-altitude flight introduces a complex coupling between air density, International Standard Atmosphere (ISA) temperature variations, and boundary layer behavior. To predict the UA variation, a re-scaling methodology based on the empirical Colburn factor curves ($j = St \bullet Pr^{2/3}$) plotted against the Reynolds number was implemented. A key assumption of this model, supported by the high thermal conductivity of the aluminum fins and the constant high-flow rate of the water-glycol coolant, is that the internal convective resistance and wall conduction are negligible compared to the air-side resistance. Consequently, the scaling law was simplified to:

$$UA \propto \dot{m}^{(1-m)} \quad \rightarrow \quad \frac{UA_{Altitude}}{UA_{ref}} \approx \left(\frac{\dot{m}_{altitude}}{\dot{m}_{ref}} \right)^{(1-m)}$$

2

where the exponent m is derived from the negative characteristic slope of the compact fin-and-plate matrix curves documented in the classic Kays & London experimental datasets [3].

1.4 Performance HEX evaluation

By iteratively solving the ϵ -NTU model across the mission profile, a counter-intuitive phenomenon was observed. At the Top Climb (5500 m), despite the lower air density, the massive temperature delta (96°C between the coolant and ambient air) causes the maximum heat rejection capacity up to 932 kW: more than triple the nominal ground capacity. A summary of the performed calculation is shown in Table 1.

Table 1
Exchanger performance evaluation.

Mission point	Air core velocity	Air temp.	Estimated UA	Effectiveness (ϵ)	Max Heat rejection allowed	Coolant inlet temperature	Coolant outlet temperature
1 (ground Baseline)	10 m/s	35°C	11,9 kW/K	0,48	300 kW	75,0°C	67,2°C
2 (2000 m)	18 m/s	2°C	16,0 kW/K	0,39	705 kW	75,0°C	56,7°C
3 (5500 m)	28 m/s	-21°C	16,3 kW/K	0,38	932 kW	75,0°C	50,8°C
Results indicate that the maximum thermal capacity reaches 932 kW, a 210% increase compared to the ground baseline. While a high cooling margin is generally desirable, in the context of Proton Exchange Membrane Fuel Cells (PEMFC), this introduces severe operational risks:							

- **PEMFC Hypothermia:** Proton Exchange Membrane Fuel Cells (PEMFC) operate in a narrow window (60–90°C). The analytical results show the liquid outlet dropping to 50.8°C, which could lead to electrochemical quenching and loss of efficiency [5];
- **Thermal Management System (TMS) Complexity:** To mitigate over-cooling, the system would require complex control architectures, such as active 3-way bypass valves and flow modulation. These components increase the total system mass and introduce additional failure points, contradicting the "fail-safe" requirements of aeronautical certification;
- **The Weight-Drag Penalty:** If the HEX core is sized solely for ground takeoff conditions, it becomes a massive "dead weight" during the cruise and climb phases. Furthermore, the oversized frontal area generates unnecessary cooling drag, penalizing the aircraft's overall range and aerodynamic efficiency.

1.5. Experimental Rationale: Bridging the Gap at CIRA Icing Wind Tunnel

The discrepancies between idealized macro-models and real-world aerothermal behavior necessitate a dedicated experimental campaign. Standard laboratory test benches often fail to replicate the complex installed performance of a radiator. Consequently, the proposed validation will be conducted at the CIRA Icing Wind Tunnel (IWT), which offers a unique environment for system-level simulation.

The IWT facility upgrade allows for the integration of the HEX within a flight-representative nacelle-like duct. This setup is crucial for several reasons:

1. **Passive Flow Induction:** Unlike standard benches that use auxiliary fans, the IWT replicates true in-flight physics by relying on the differential aerodynamic pressure between the intake and the exhaust. This

ensures that the mass flow rate through the core is naturally driven by the ram-air effect, as it would occur in a real nacelle.

2. **Cooling Drag and Installation Effects:** The experimental setup enables the isolation of cooling drag from the total nacelle drag. This allows for a detailed assessment of how inlet/outlet design and non-uniform flow distribution across the core impact the overall vehicle efficiency.
3. **Off-Design Conditions:** Analytical models based on the Colburn law are typically limited to 1D assumptions. The wind tunnel campaign will extend this by evaluating performance degradation under yaw and sideslip conditions, providing the necessary data to generate high-fidelity operational performance maps [6].

Ultimately, these tests will validate the tightly coupled interaction between external aerodynamics and internal radiator pressure losses, providing a flight-proven characterization that is essential for the sizing of future zero-emission propulsion systems.

2. Overview of CIRA IWT facility

The Icing Wind Tunnel (IWT) at the Italian Aerospace Research Centre (CIRA) is a state-of-the-art, closed-loop, refrigerated, and pressurized facility designed to reproduce in-flight atmospheric icing conditions for aeronautical applications. Its original mission was to support aircraft certification by investigating ice accretion on critical components and assessing its impact on aircraft safety and performance under controlled, repeatable flight conditions. Since its inauguration in September 2002, the IWT (Fig. 4) has become one of the world's most advanced icing wind tunnels.

The IWT is distinguished by a unique set of capabilities that place it among the most advanced icing wind tunnels worldwide. It is the largest refrigerated wind tunnel currently in operation, the highest-speed icing facility, with test speeds up to Mach 0.7, and the only installation capable of simultaneously reproducing both altitude and temperature conditions representative of real flight. These characteristics make the facility an exceptionally flexible platform for aerodynamic, environmental, and icing research.

The tunnel is extensively used for the qualification and certification of a wide range of ice protection technologies, including pneumatic boots, hot bleed-air systems, and electrothermal anti-icing and de-icing solutions. Its testing capabilities are further enhanced by a dedicated engine flow simulation system, which allows the assessment of icing phenomena on aircraft and helicopter engine intakes over a broad operating envelope. A schematic layout of the Icing Wind Tunnel and its main subsystems is shown in Fig. 5.

The IWT is designed with a highly adaptable test circuit that includes three interchangeable test sections—the Main Test Section (MTS), Secondary Test Section (STS), and Additional Test Section (ATS)—together with an Open Jet configuration. This flexibility allows the facility to support a broad range of aerodynamic and icing investigations. The available test sections cover different flow conditions and dimensions, with cross-sectional areas ranging from $1.15 \times 2.35 \text{ m}^2$ for the STS up to $3.6 \times 2.35 \text{ m}^2$ for the ATS. The maximum achievable Mach numbers are 0.70 in the STS, 0.40 in the MTS, and 0.25 in the ATS. Depending on the experimental requirements, both closed-wall and slotted-wall configurations (7% porosity) can be adopted.

The air cooling system relies on a twin-row heat exchanger, which allows the airflow temperature to be reduced to minimum static values of – 40°C in the STS and – 32°C in the other configurations. A dedicated pressurization and evacuation system provides accurate control of the tunnel pressure, enabling operation from 39 kPa, corresponding to an altitude of approximately 7,000 m, up to 145 kPa for high-Reynolds-number aerodynamic investigations.

Artificial icing conditions are generated through the Spray Bar System (SBS), composed of up to 20 aerodynamically optimized stainless-steel spray bars equipped with a total capacity of 1000 nozzles. Positioned in the settling chamber upstream of the test section, the system is supplied with pressurized water and air to create a controlled supercooled water droplet cloud. The 18 m long contraction section between the settling chamber and the test area ensures sufficient time for the droplets to reach thermal equilibrium with the cold airflow before interacting with the test article, providing a stable and repeatable icing environment.

The SBS enables accurate control of key icing parameters, including Median Volume Diameter (MVD) and Liquid Water Content (LWC), allowing the simulation of a wide range of icing conditions defined by certification standards such as CS-25/29 Appendix C and selected Supercooled Large Droplet (SLD) scenarios. The main characteristics of the different test sections are reported in Table 2.

Table 2
IWT test sections main characteristics

TEST SECTION	Dimension [m]	Max speed*(Mach)	T [°C]	Altitude* [m]
MAIN	2.25 x 2.35	0.41	-32 < Ts < + 40	0 ÷ 7000
SECONDARY	1.15 x 2.35	0.70	-40 < Ts < + 40	0 ÷ 7000
ADDITIONAL	3.60 x 2.35	0.25	-32 < Ts < + 40	0 ÷ 7000
OPEN-JET	2.25 x 2.35	0.34	-32 < Ts < + 40	0 ÷ 7000

Owing to its versatility and broad operating capabilities, the facility is now being adapted to support the experimental validation of advanced technologies, including Thermal Management Systems (TMS) for next-generation hybrid-electric aircraft.

3. Icing Wind Tunnel Upgrade description

The implementation of Thermal Management Systems (TMS) in aircraft components requires comprehensive verification and validation campaigns before their integration into flight operations. To address this requirement, the Italian Aerospace Research Centre (CIRA) is enhancing its Icing Wind Tunnel (IWT) facility to provide a representative test environment for investigating the thermal management needs of future hybrid-electric aircraft. The upgraded facility will enable the simulation of thermal loads associated with propulsion systems at cruise conditions, allowing the assessment of TMS performance under realistic operating scenarios.

As part of this upgrade, the IWT fluid conditioning capabilities will be expanded to complement the existing icing and anti-icing testing functionalities. A dedicated thermal generation system will be introduced, capable of delivering up to 500 kW of thermal power through a water-glycol heat transfer fluid, with operating conditions reaching 10 bar pressure and a flow rate of 1200 l/min. The system will provide controlled thermal loads to different types of heat dissipation devices, including conventional liquid-to-air heat exchangers and innovative concepts such as integrated structural heat exchangers, skin heat exchangers, and nanofluid-based solutions designed to improve heat transfer efficiency.

The new thermal generation unit will allow precise control of the heat load applied to electrical components, continuous measurement of key parameters such as flow rate, pressure, and temperature, and experimental validation of cooling strategies under realistic thermal stresses. The main advantage of this approach is the possibility of testing TMS components in a representative flight environment, reproducing aircraft speed, altitude conditions, pressure and temperature variations, and, when required, icing effects on heat exchanger surfaces. The system is based on an electrically driven heating unit that supplies the heat transfer fluid to the test section under controlled temperature and pressure conditions managed by the facility control system. It is designed to reproduce the hot coolant flows generated by electric motors, energy storage systems (batteries and fuel cells), and power electronics, enabling the evaluation of thermal dissipation solutions in realistic operating conditions. The expected operating range and maximum thermal power capability of the system are reported in Table 3.

Table 3
Operating conditions and accuracy of the heat load simulation system

Heat load simulator operating conditions	
Maximum heat power	500 kW
Maximum continuous operating temperature	120°C
Maximum operating temperature (peak only)	130°C
Maximum operating pressure	10 bar
Maximum mass flow rate	1200 l/min
Accuracy of the control system	
Temperature control accuracy	± 1°C
Flow rate control accuracy (based on set value)	± 2%
Pressure control accuracy	± 0.2 bar
Minimum adjustable thermal power	10 kW

To ensure precise control of the temperature, pressure and flow rate of the refrigerant, the system has been equipped with pressure (PT in Fig. 6) for absolute pressure measurement, PDI for differential pressure measurement) and temperature sensor (TT in Fig. 6.) at all the key points, particularly upstream and downstream of each component within the setup, in order to thermally characterize each individual

component. It will also include flow meters (FT in Fig. 6), both to directly measure the fluid flow rate towards the test section—providing feedback to the electronic control unit—and to ensure that there are no leaks inside the test room during testing.

A filtering system for the heat transfer fluid is included to prevent the accumulation of dust and scaling deposits on the electric resistances or in the article during the test. Since the system is built with adequate thermal insulation to minimize heat losses to the environment during the test, an electro-fan radiator, referred in the Figure as the "Cooling Radiator", has been included to facilitate rapid cooling of the heat transfer fluid and be able to quickly repeat tests such as cold start transients from ambient temperature up to maximum operating temperature without using the tunnel or the test article to cool the fluid to initial conditions.

Finally, the planned IWT upgrade will lead to three different complementary kind of tests:

- **Test of TMS Performance**
 - **Simulated Operating Conditions:** Replication of flight conditions matching regional aircraft mission profile points.
 - **Variable Testing Parameters:** Altitude (0–7'000 m), temperature (-40°C to + 40°C), air speed (M0.1-M0.7).
 - **Performance Metrics:** Heat transfer efficiency, pressure drop, thermal response time, system stability at critical points across the entire mission profile.
- **Icing Wind Tunnel Testing**
 - **Air Intake + Heat Exchanger Integration:** Analysis of icing impact on combined system performance.
 - **Ice Accretion Effects:** Evaluation of ice growth on heat exchanger surfaces and performance degradation.
 - **Advanced Anti-ICE Systems:** Innovative solutions utilizing TMS hot fluid for efficient Ice prevention.
- **Aerodynamic Assessment**
 - **Drag Evaluation:** Measurement of aerodynamic drag induced by heat exchangers.
 - **Propulsive Thrust Contribution:** Analysis of aero-thermo-propulsive performance of air-cooling systems (Meredith effect).

Conclusions and future perspective

Present work shows how next generation of Hybrid and Electric aircrafts will need to develop systems to manage the thermal loads developed on board. Moreover, to ensure safety and reliability and to optimize the overall dimensions and performance of TMS, it is useful to carry out tests in a simulated flight environment. CIRA is involved in the development of new Heat Exchanger based on ALM Technology and the new test

capabilities of IWT can support the activity. Indeed, CIRA undertakes a significant update of its Icing Wind Tunnel facility to allow tests of full-scale TMS components in real flight conditions, even cloud formation.

Next goal will be the creation of a robust and repeatable test methodology for the testing and integration of thermal load dissipation terminals of an aircraft. Finally, it can be certainly concluded that the opportunity of simulation in Icing Wind tunnel of whole TMS opens to several testing activities such as innovative anti-ice systems powered from hot fluid coming from the components to be cooled.

NOMENCLATURE

A	Exchanger Area
C	Heat capacity rate (W/K)
CIRA	Italian Aerospace Research Center
ϵ	Heat exchanger effectiveness
HEX	Heat Exchanger
ISA	International Standard Atmosphere
IWT	Icing Wind Tunnel
j	Colburn factor
LMTD	Log Mean Temperature Difference
m	Characteristic slope exponent of the fin-and-plate matrix
$\dot{m}$	mass flow rate
NTU	Number of transfer units
PEMFC	Proton Exchange Membrane Fuel Cell
Pr	Prandtl number
Q	Thermal Power
Re	Reynolds number
St	Stanton number
T	Temperature
TMS	Thermal Management System

U Global exchanger transmittance

v Velocity

Declarations

Ethics, Consent to Participate, and Consent to Publish

Not Applicable

declarations

The datasets generated during and/or analysed during the current study are available from the corresponding author on reasonable request

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Figures

Real EXH datasheet:

- Dissipated Power: **300 kW**
- Coolant: **Water/glycole 50%**
- Coolant flow: **600 lpm**
- Coolant inlet temperature: **75 °C**
- Coolant outlet temperature: **67,2 °C**
- Ambient temperature: **35 °C**

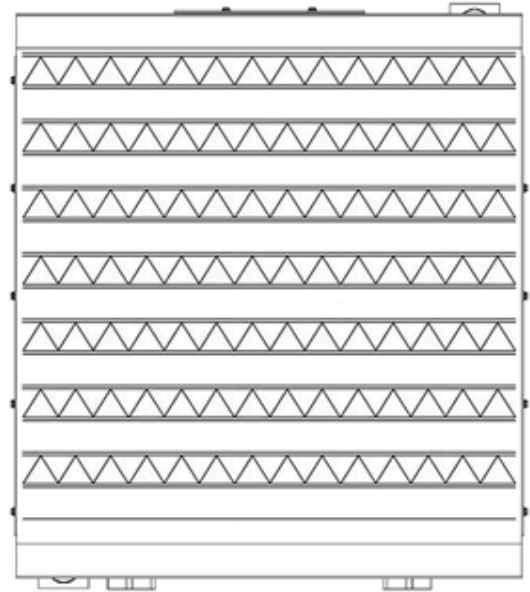


Figure 1

Real working parameter of Crossflow Heat Exchanger.

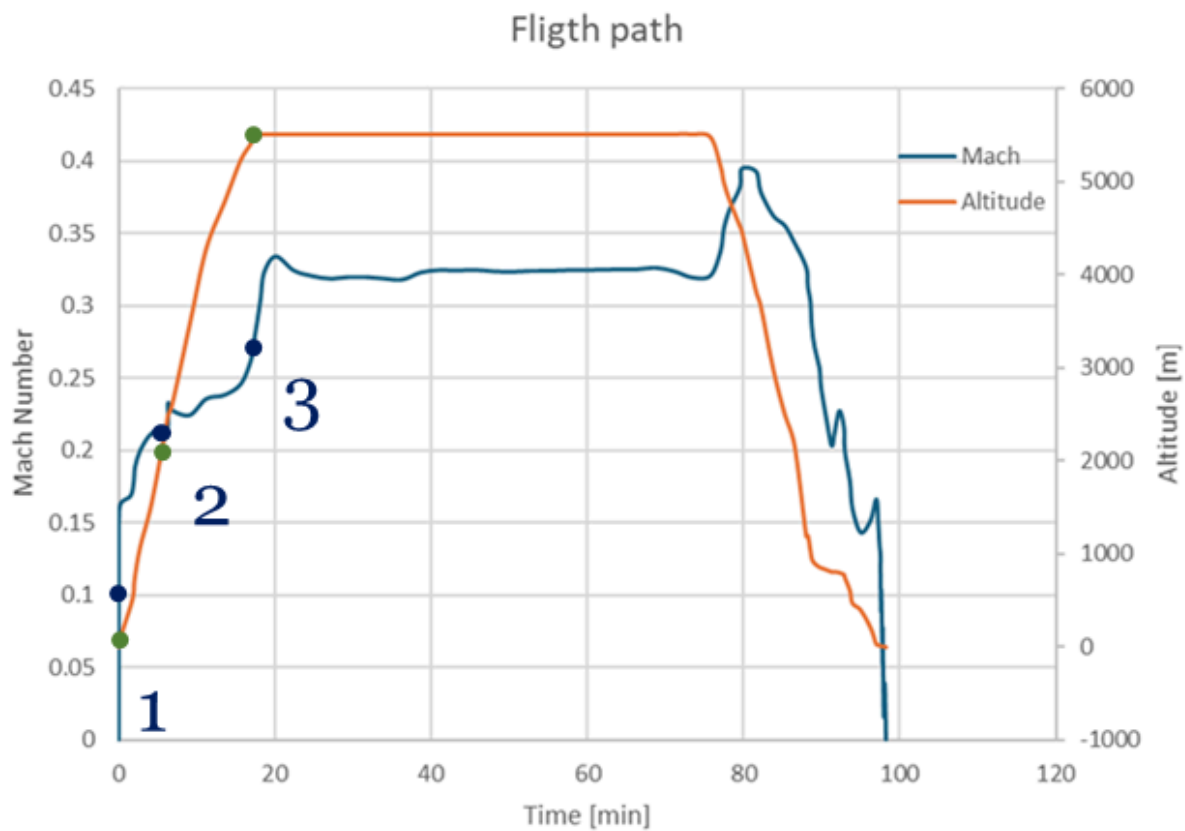


Figure 2

Regional mission profile.

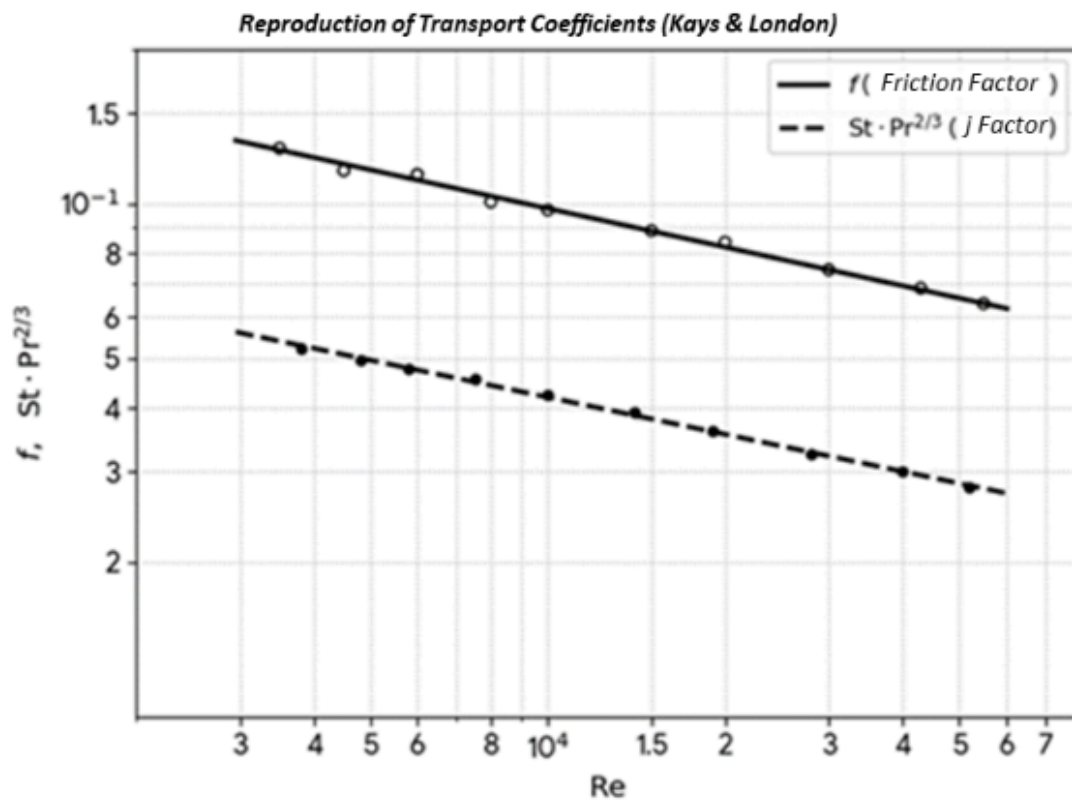


Figure 3

Colburn factor trend as a function of Reynolds number.



Figure 4

Icing Wind Tunnel aerial view

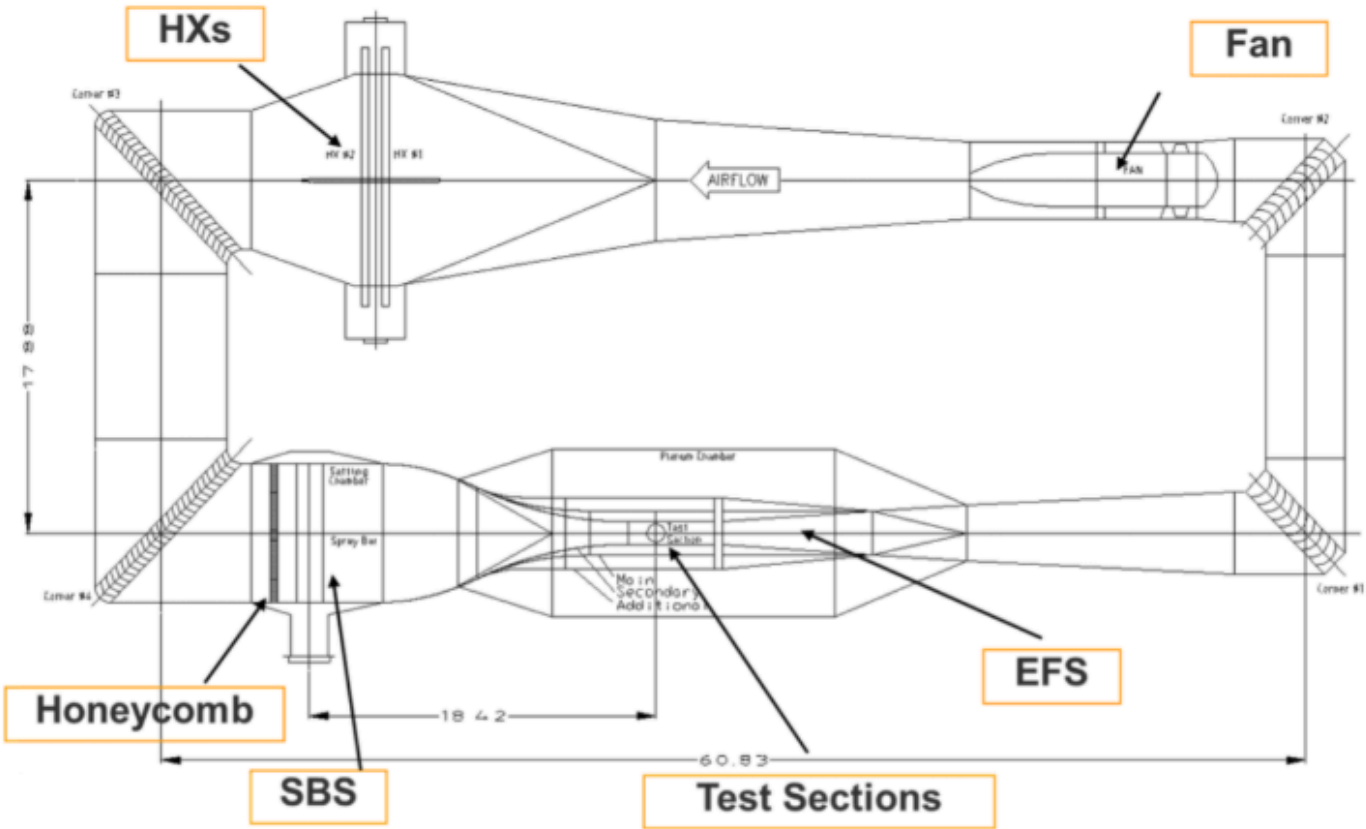


Figure 5

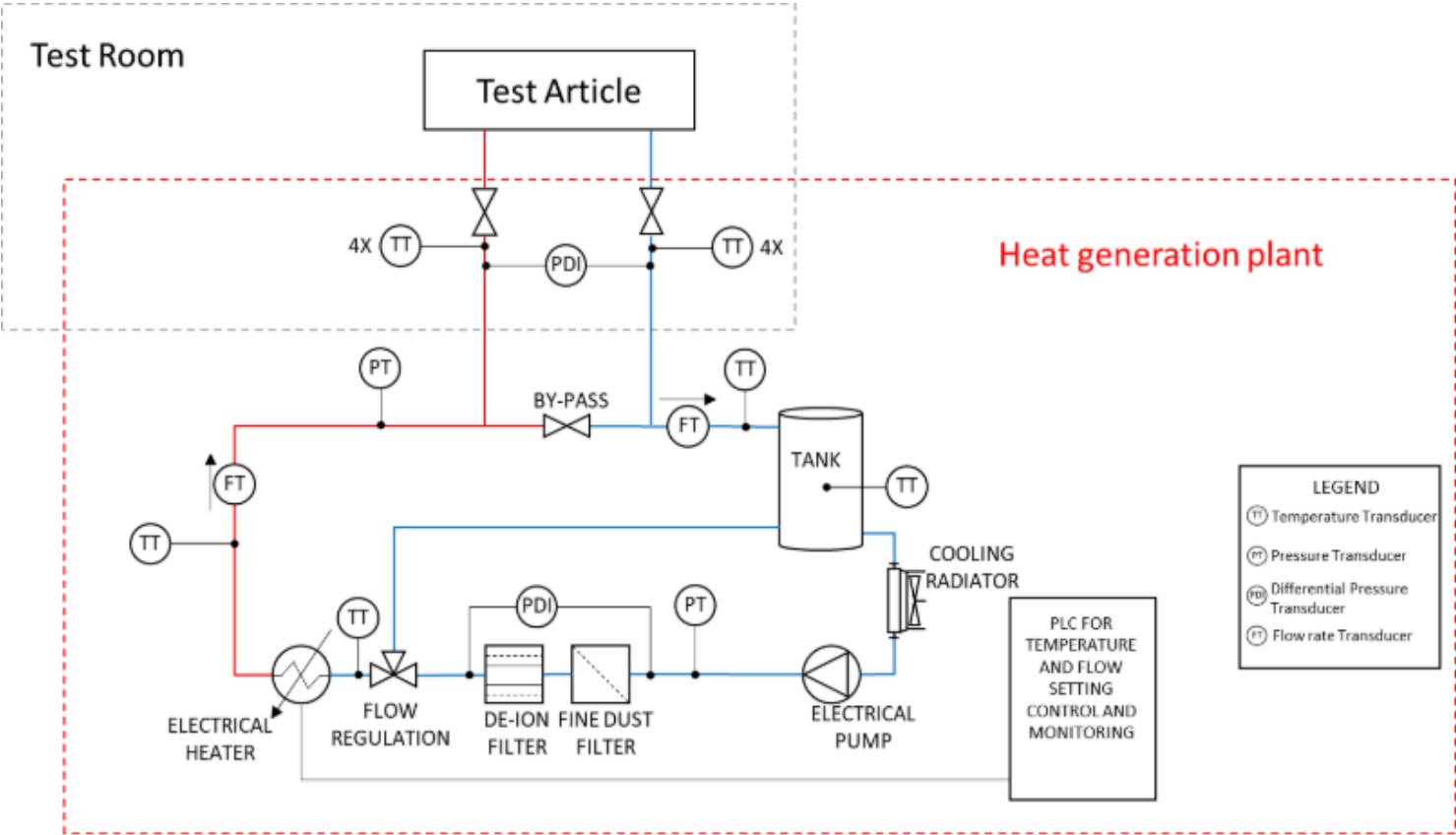


Figure 6
Layout of the main components of the heat generation system.