

Energy-based Aerodynamic Loss and Recovery Characteristics of Adiabatic and Heated Fuselages

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An energy-based aerodynamic analysis of the mechanical loss generation and potential energy/exergy recovery mechanisms is carried out for adiabatic and heated 2-D axisymmetric flows over fuselage-shaped axisymmetric bodies. A generality of these mechanisms is obtained from dimensional analysis by appropriately scaling the freestream Reynolds and Mach numbers, whilst varying a reference fuselage's fineness ratio. Thermo-aerodynamic implications and synergies of boundary layer heating on the loss distribution, energy and heat exergy recovery potentials are further considered for varying wall temperature ratios. The result is a clear identification of partial dynamic similarity and heat transfer effects on flow mechanisms such as shear layers, separation bubbles and shockwaves of axisymmetric flows and subsequent implications on loss distribution and energy recovery characteristics relating to boundary layer ingestion. The analysis indicates that dissipating heat from aircraft surfaces aids, circumstantially, to drag reduction of unpowered fuselage bodies and increases, relative to the adiabatic, the recoverable energy fraction available for the BLI propulsor, by strategically manipulating the loss distribution, whilst removing excess heat from the aircraft's subsystem (batteries, fuel cells). Finally, an approach to assess the feasibility of exergetic heat recuperation as a possible means of useful work extraction and improved aerodynamic performance is explicitly introduced and discussed in the paper.

Nomenclature

b	=	local width parameter, m
$\bar{b}$	=	dissipation-weighted average width parameter, m
$\mathcal{B}$	=	baroclinic power, W
c	=	fuselage chord length, m
C_D	=	dissipation coefficient

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$C_{\mathcal{D}}^*$	=	dissipation flux coefficient, m
C_h	=	heat transfer coefficient, $\text{W}/(\text{m}^2 \cdot \text{K})$
D	=	fuselage diameter, m
D	=	total (profile) drag, N
D_p	=	pressure drag, N
D_v	=	friction drag, N
$\mathcal{D}$	=	dissipation rate, W/m
dS	=	infinitesimal surface element of control volume, m^2
dV	=	infinitesimal volume element of control volume, m^3
$\dot{\mathcal{E}}_{qB}$	=	exergy rate due to conductive heat flux, W
$\dot{\mathcal{E}}$	=	$\dot{\mathcal{E}}_k + \dot{\mathcal{E}}_p + \dot{\mathcal{E}}_\tau$, mechanical energy/exergy outflow rate, W
FR	=	c/D , fineness ratio
H	=	boundary layer shape parameter
H^{**}	=	boundary layer density-flux shape parameter
$\mathcal{K}$	=	thermal conductivity, $\text{W}/(\text{m} \cdot \text{K})$
M	=	Mach number
$\hat{n}$	=	unit normal vector
p	=	static pressure, Pa
p_G	=	$p - p_\infty$, gauge static pressure, Pa
P_t	=	total pressure, Pa
Pr	=	$\mu C_p / \mathcal{K}$, Prandtl number
$\mathcal{PER}$	=	potential for energy recovery
$\mathcal{PER}_\theta$	=	potential for heat exergy recovery
$\dot{q}$	=	conductive heat flux vector, W/m^2
$\mathcal{R}_p$	=	D_p/D , pressure drag ratio
$\mathcal{R}_v$	=	D_v/D , skin friction drag ratio
Re_c	=	$\rho_\infty V_\infty c / \mu_\infty$, chords-based Reynolds number
Re_θ	=	$\rho_e V_e \theta / \mu_\infty$, mom. thickness-based Reynolds number
s, n	=	local coordinate system
S	=	control volume surface, m^2
T	=	static temperature, K
T_w	=	wall static temperature, K

u, v, w	=	absolute velocity components, m/s
$\mathcal{V}$	=	control volume, m ³
$\mathbf{V}$	=	fluid velocity, m/s
α	=	speed of sound, m/s
γ	=	specific heat ratio
δ	=	boundary layer thickness, m
δ^*	=	displacement thickness, m
δ^{**}	=	density thickness, m
Δ^*	=	displacement (mass-defect) area, m ²
Δ^{**}	=	density-flux defect area, m ²
ζ	=	shock wave identifier
η_{Carnot}	=	Carnot efficiency
Θ	=	volumetric pressure work rate, W or mom. defect area (for boundary layer analysis), m ²
Θ_r	=	reversible (isentropic) volumetric pressure work rate, W
μ	=	kinematic viscosity, kg/(m · s)
ξ	=	boundary layer identifier
ρ	=	density, kg/m ³
$\bar{\bar{\tau}}$	=	viscous stress tensor, Pa
Φ	=	viscous dissipation rate, W
Φ^*	=	$\Phi + \mathcal{B}$, total mechanical loss rate, W

Subscripts

∞	=	freestream-quantity
A	=	absolute reference frame quantity
B	=	quantity on body surface
e	=	boundary layer edge quantity
O	=	quantity on outer boundaries
TE	=	trailing edge quantity
TP	=	Trefftz plane quantity
SC	=	side cylinder quantity

Superscripts

ϑ = heated fuselage drag quantity

I. Introduction

CONVENTIONAL aerodynamic assessment of non-lifting bodies is reliant on momentum conservation to extract and decompose the aerodynamic drag components exerted on the body. The near field force integration offers a numerically and experimentally convenient approach of determining the total profile drag in terms of pressure and friction drag components but is inherently limited in that it offers no traceability and causality relation to the underlying physical farfield mechanisms [1]. A more physically intuitive breakdown of the drag sources is obtained via the farfield momentum integration of the flow-field perturbations along the outer boundary surfaces of a control volume surrounding the body. Various farfield method techniques have been developed in an attempt to decipher physical drag generation mechanisms and attribute drag contributions to physical flow mechanisms related to shear layers, vortical systems and shock waves (see detailed reviews in [2–4]). A major motivation behind the development of farfield methods lies in their potential to establish compatibility between numerical and experimental measurement techniques [5, 6] for accurate aerodynamic force extraction from limited wake plane data [3, 7, 8]. Inclusion of the 2nd of Law Thermodynamics [2, 9–11] along with appropriate Taylor expansion techniques [9] further enhances physical interpretation and allows for targeted wake measurements associated with entropy generation mechanisms inside the flow field [7]. However, a certain degree of ambiguity among the various drag constituents’ definitions (profile, wave, induced drag .etc), still exists due to the derivation assumptions (small perturbation assumption, higher order term omission, unrolled vortex) and mutual coupling among the various terms [12, 13]. Recent farfield method advancements include lamb vector-based formulations [14–17] which claim to provide means of decomposing the aerodynamic force into reversible and irreversible components [15, 17], the former yielding rigorous definitions for lift and lift-induced drag [15] and the latter measuring entropy generation due to profile losses. Apparent numerical benefits of vorticity-based formulations arise from the fact that volume and surface integrations of the vortex force components are confined within regions of high vorticity, thus removing spurious contributions within low resolution regions of the computational domain.

A rigorous comparison of the limitations and definition ambiguities of various far field momentum methods are discussed by Méheut and Bailly[3] and Wu et al. [18] while Drela [19] further addresses the challenges and limitations of farfield momentum methods in identifying the aerodynamic force components (thrust, drag) when the aerodynamic coupling of the airframe and the propulsion system preclude the possibility of isolating the propulsive domain from the rest of the flow field. Such an example is the concept of boundary layer ingestion (BLI) whereby the airframe’s boundary layer is ingested and re-accelerated by a propulsion system positioned along the aft part of the fuselage. For these configurations, the resulting force field is a corollary of mutual aero-propulsive interaction that prevents individual force components from being attributed to thrust or drag. Thus, the absence of causality of near-field force integration

is only partially addressed by farfield momentum decompositions, not only because the distinction of aerodynamic force components is circumstantially ambiguous, but because the drag decomposition does not quantify the evolution of volumetric loss generation inside the flow field. This is because the farfield aerodynamic force measurement is constant and independent of position (theoretically only but not in CFD due to numerical dissipation) while the affiliated dissipative losses lag behind and run to completion at an infinite distance downstream. A direct implication of the above is that momentum-based decompositions provide, for instance, estimations about the profile drag of an aerodynamic body via wake plane measurements but do not provide means of identifying and quantifying recoverable energy portions within the boundary layer/wake that could be utilized for energy recovery purposes, i.e. BLI. The above limitations are further exacerbated for emerging aircraft architectures with on board batteries, fuel cells and alternative fuels such as hydrogen due to the additional heat management challenges. For those configurations, strategically dissipating heat from airframe surfaces represents an opportunity for passive flow control and advanced thermo-aerodynamic synergies by heating the boundary layer upstream of the BLI propulsor or by considering the plausibility of heat-to-work conversion which cannot be sufficiently addressed from a momentum-point of view.

Drela [20] addressed the above challenges via an integral power balance approach allowing for a clear identification of the flowfield's power sources and sinks, irreversibilities and energy outflows relevant for energy recuperation. The resulting formulation conveniently relates the drag power of the unpowered body to the aerodynamic loss and recovery mechanisms inside the flow field which are clearly identified and quantified. Sato [21] further demonstrated the wide range of applicability of the approach by deriving an integral defect energy balance allowing for dissipation-based drag approximations of 2D aerodynamic bodies, compressibility-loss corrections and rational definitions of performance metrics. Sanders and Laskaridis [22] provided an elegant interpretation of the energy transmission mechanisms demonstrating that these manifest in the absolute reference frame and further introduced the definition of the Potential for Energy Recovery (PER) factor as a measure of the mechanical energy portion that could be harvested via BLI. The absence of heat management capabilities and the lack of a thermodynamically "strict" irreversibility definition of the power balance formulations has been addressed by Arntz et al. [23] via the formulation of an exergy balance inclusive of 1st and 2nd Law thermodynamics that explicitly quantify the hypothetical heat exergy and thus useful work that could be extracted from heated walls. Aguirre et al. [24] further elucidated aspects of mechanical and thermal exergy outflow recovery by sifting out isentropic portions that self-recover after the full expansion of the flow downstream, shedding light on aspects of energy recovery and its possible overestimation by the conventional energy/exergy methods.

Despite the significant advancements of energy and exergy-based methods, a comprehensive study on the energy loss and recovery characteristics of typical fuselage designs as well as their response to flow field conditions is yet to be addressed. For future aircraft concepts with BLI propulsion, prior knowledge of these characteristics may aid, for example, to fine-tuning the aerodynamic design for maximum energy recovery and elucidate uncertainties regarding dynamic similarity mismatches for experimental testing. The limited experimental tests on BLI configurations [25–31]

were obtained at low Reynolds (Re) and subsonic Mach (M) numbers which do not correspond to the viscous flow structure developed over the fuselage at flight conditions. Thus, it remains unclear if the energy recovery potential of the boundary layer which largely determines the claimed benefits, may be "extrapolated" from wind-tunnel models to full-scale configurations. The landmark work of Uranga et al. [27–29] on the D8, whilst demonstrating, for the first time, the applicability of a complete energy-based experimental analysis to BLI configurations, it did not resolve the effect of mismatching similarity parameters, such as Re and M numbers on the energy characteristics of the ingested boundary layer flow. The authors [28] argued that whilst the wind-tunnel model's kinetic energy defect varies with Re , the energy recovery portion and thereby BLI benefit, is expected to be comparable with that of the full-scale configuration. This uncertainty is further exacerbated by the omission of a thermo-compressible-related energy term, the contribution of which is otherwise vaguely incorporated as a correction to dissipation for compressible flows [28, 32], based on previous results by Sato [21]. The origins of the latter loss mechanism and its response to various compressible flow scenarios remains ambiguous, especially when strong heat transfer effects are present. Such flows are of particular interest for flight vehicles with on-board heat sources such as batteries and fuel cells because heat dissipation from airframe surfaces may aid to both heat load management and potentially aerodynamic performance improvements. Previous experimental studies on wall heated (and cooled) flows over unpowered bodies [33–35] have reported circumstantial aerodynamic benefits in terms of drag reduction for limited flow conditions, but a systematic energy-based analysis of the wall heating effect on the loss and recovery characteristics, inclusive of exergetic aspects for assessing heat work potential is yet to be addressed.

In response to the preceding challenges, this paper builds upon mechanical energy and exergy-based formulations to address causality aspects between drag and mechanical loss/recovery characteristics relevant for non lifting adiabatic and heated axisymmetric flows over a fuselage. Appropriate flow field dimensional analysis, based on the principle of dynamic and geometric similarity [36, 37], allows for rigorous aerodynamic mapping of the relevant loss and recovery mechanisms and their spatial evolution characteristics as a function of Re and M numbers as well as fuselage fineness ratio (FR). The developed analysis sheds light on possible suppression or amplification of flow mechanisms related to shear layers, separation bubbles and shockwaves due to partial similarity which may be relevant for wind tunnel experiments and maps the energy recovery characteristics relevant for BLI propulsion. Next, the effect of boundary layer heating is explored as a possible means of passive flow control for mechanical energy loss and recovery manipulation purposes, relevant for emerging aircraft concepts with BLI propulsion and on-board heat sources. Finally, heat recuperation aspects are explicitly addressed via the definition of the potential for heat exergy recovery [38] that quantifies the heat to work conversion potential of configurations with non-adiabatic wall surfaces.

II. Integral Mechanical Energy Analysis

A. Control Volume Definition

A 2D cutaway view of the control volume (CV) enclosing an unpowered aerodynamic body and a farfield domain is illustrated in Fig. 1. The CV is assumed to be continuous with its nearfield portion lying on the body surface, S_B , and its outer portion enclosing an arbitrary farfield domain, S_O . It is advantageous, however, to define S_O as a rectangular (for 2D) or cylindrical (for 2D axisymmetric) domain with its outer boundary control surfaces partitioned into a side-cylinder, S_{SC} , and a Trefftz plane, S_{TP} , oriented parallel and perpendicular to the freestream velocity, respectively. For numerical convenience, the inlet plane (not shown) and the side cylinder portions are positioned in the freestream allowing all the flowfield perturbations and loss generation sources (shear layers and shock waves) to be traced via a Trefftz plane analysis. The mechanical energy analysis is carried out in the absolute reference frame (ARF) whereby the CV is fixed on the aerodynamic body flying through quiescent atmosphere at a constant velocity of $\mathbf{V}_B$ [22]. Relative reference frame (RRF) quantities, relevant for CFD analysis, are conveniently correlated to the ARF and vice versa via the absolute velocity vector, $\mathbf{V}_A$ defined as:

$$\mathbf{V}_A = \mathbf{V} - \mathbf{V}_\infty \quad (1)$$

where $\mathbf{V}$, $\mathbf{V}_\infty$ the local and the freestream fluid velocities in the RRF, respectively. The above yield $\mathbf{V}_A = 0$ for the upstream undisturbed flow and a flight velocity equal and opposite to the RRF freestream, $\mathbf{V}_B = -\mathbf{V}_\infty$.

B. Formulation Overview

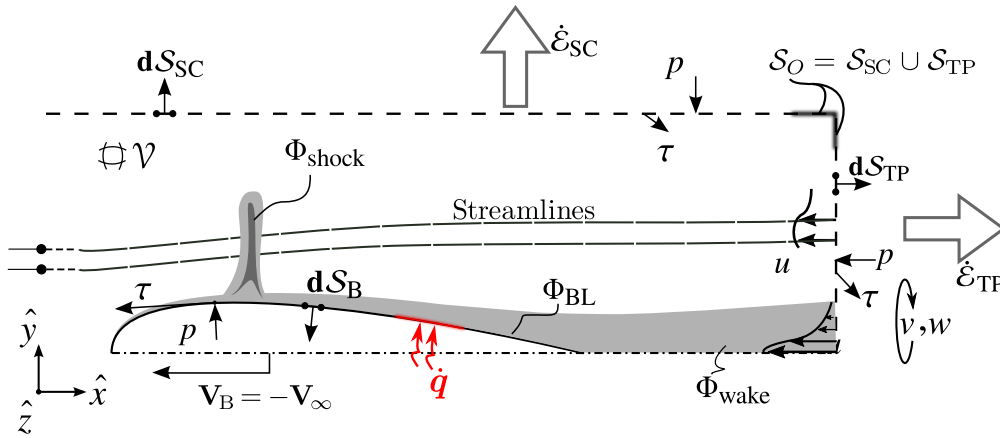


Fig. 1 Control volume definitions for an unpowered body with annotated work pathways and relevant flow features.

Following the derivation by Sanders and Laskaridis [22] and with reference to Fig. 1, the ARF mechanical energy balance applied over an unpowered aerodynamic body flying through still atmosphere at a velocity of $\mathbf{V}_B = -\mathbf{V}_\infty$ yields

a drag power decomposition in terms of mechanical power outflows and sinks:

$$DV_\infty = \dot{\mathcal{E}} + \Theta + \Phi \quad (2)$$

DV_∞ is the (nearfield) drag power imparted to the flow via the integrated skin friction and pressure wall forces acting at a flying velocity of $\mathbf{V}_B = -\mathbf{V}_\infty$. Its dependence on the flying velocity implies that the power trace of the aerodynamic drag is physically related to the wall surfaces resembling a mechanical power transmission interface via the no-slip condition that entrains the wall-adjacent fluid flow at finite velocity, $\mathbf{V}_B$.

$\dot{\mathcal{E}}$ measures the net mechanical energy outflow rate leaving the outer domain boundaries ($\dot{\mathcal{E}}_{SC} + \dot{\mathcal{E}}_{TP}$) and accounts for the total kinetic energy rate ($\dot{\mathcal{E}}_k$), pressure work rate ($\dot{\mathcal{E}}_p$) and viscous boundary work rate ($\dot{\mathcal{E}}_\tau$) contributions deposited outside the CV boundaries:

$$\dot{\mathcal{E}} = \dot{\mathcal{E}}_{SC} + \dot{\mathcal{E}}_{TP} = \underbrace{\oint \left(\frac{\rho}{2} \mathbf{V}_A^2 \right) \mathbf{V} \cdot \hat{n} dS_O}_{\dot{\mathcal{E}}_k} + \underbrace{\oint p_G \mathbf{V}_A \cdot \hat{n} dS_O}_{\dot{\mathcal{E}}_p} + \underbrace{\oint -\bar{\bar{\tau}} \cdot \mathbf{V}_A \cdot \hat{n} dS_O}_{\dot{\mathcal{E}}_\tau} \quad (3)$$

where ρ the fluid density, $p_G = p - p_\infty$ the static gauge pressure and $\bar{\bar{\tau}}$ the shear stress tensor.

With reference to the CV definition of Section II.A, the net mechanical energy outflow definition (3) reduces to $\dot{\mathcal{E}} = \dot{\mathcal{E}}_{TP}$ and measures mechanical power contributions associated with flow phenomena such as wakes, shockwaves and vortices. For unpowered bodies, this term represents a measure of the drag power portion that is in useful form and could be strategically recovered via recuperation devices such as BLI and wing tip propulsion. For a fully expanded flow, the mechanical energy outflow of an unpowered body may be recast non-dimensionally in terms of a Potential for Energy Recovery (PER) factor [22] that provides a measure of the aerodynamic benefits associated with the energy recovery process.

Φ is the viscous dissipation rate that governs the process of shearing-induced irreversible conversion of mechanical energy into heat due to fluid deformation and is the driving loss mechanism inside viscous regions such as boundary layers (Φ_{BL}) wake shear layers (Φ_{wake}) and shock waves (Φ_{shock}). It is defined as:

$$\Phi = \iiint (\bar{\bar{\tau}} \cdot \nabla) \cdot \mathbf{V} dV = \Phi_{BL} + \Phi_{wake} + \Phi_{shock} \quad (4)$$

Θ is associated with compressibility (via the dilatation term, $\nabla \cdot \mathbf{V}$) and measures the volumetric pressure power of the fluid compressing and expanding against the atmosphere:

$$\Theta = - \iiint p_G \nabla \cdot \mathbf{V} dV = \Theta_r + \mathcal{B} \quad (5)$$

This term is closely related to compressibility (via the dilatation term $\nabla \cdot \mathbf{V}$) and accounts for the " $P dV$ " work [20] associated with fluid compression/expansion. Significant local contributions span regions of strong acceleration (fuselage nose) $\Theta < 0$ or deceleration (fuselage aft cone) $\Theta > 0$. These however do not directly represent localised losses, instead, they mostly account for isentropic expansion/compression of the fluid and self-cancel when the flow fully expands to freestream conditions downstream of the body. Thus, Θ may be further decomposed into an isentropic (Θ_r) and a baroclinic ($\mathcal{B}$) portion, given by the right-hand side equality of Eq. (5). The former measures the reversible portion of Θ that naturally self-recovers (i.e goes to zero) when the flow fully expands downstream of the configuration. The latter governs the compressibility losses related to non-isentropic processes (heat addition) due to viscous self-heating and heat sources acting over regions of strong fluid compression and expansion as well as shock waves (see discussions in [39]). In the scenario where the Trefftz plane of Fig. 1 is positioned sufficiently far from the trailing edge of the body at freestream pressure, then $\Theta_r = 0$ [28, 39] and Eq. (5) provides an exact measurement of the total baroclinic power imparted to the flow. For the purposes of this study, qualitative means will be adopted to attribute numerically specific proportions of the global effect to particular decomposed flow phenomena, such as boundary layers and shock waves, following defect arguments from [39, 40] which are briefly detailed in Appendix A.

C. Mechanical Loss Characterisation

1. Shear Layer Dissipation

The dissipative shear layer characteristics are more conveniently expressed in terms of the local dissipation coefficient (C_D) [20], defined as:

$$C_D \equiv \frac{\mathcal{D}}{\rho_e V_e^3} = \frac{\int_0^\delta (\bar{\tau} \cdot \nabla) \cdot \mathbf{V} \, dn}{\rho_e V_e^3} \quad (6)$$

where $\rho_e V_e^3$ the local area-specific kinetic energy edge flow of the boundary layer and $\mathcal{D}$ the dissipation rate (per unit length) integrated across the boundary layer height, δ . The dissipation coefficient is evaluated along the viscous flow path (s -direction) defined by the local $s - n$ cartesian coordinate system that follows the curvature of the body, as shown in Fig. 2 (left). For axisymmetric flows, the viscous dissipation rate along the flow path is uniquely related to the local distribution of C_D as:

$$\Phi_{BL} + \Phi_{wake} = \int_{BL} \rho_e V_e^3 C_D b \, ds + \int_{wake} \rho_e V_e^3 C_D b \, ds \quad (7)$$

where b is the lateral width parameter that accounts, relative to a 2D flow, for the lateral divergence of the shear layer crossing a finite cross-sectional area S_δ with surface element $dS_\delta = b \, dn^*$ (see Fig. 2 (Right)). It is defined as [41]:

$$b = 2\pi \left(R + n \sqrt{1 - (dR/ds)^2} \right) \quad (8)$$

*For 2D flows, the corresponding area (per unit length) simply equals the boundary layer thickness and $b = 1$

where R is the radial distance of the wall from the camberline of the axisymmetric body. The dependence of Φ on both the C_D and b implies that for axisymmetric shear layers, the rate of change of viscous dissipation $d\Phi/ds$ is not governed by the local dissipation characteristics exclusively, as in a 2D shear layer. The reason is that, in contrast with 2D flowfields, the axisymmetric shear layer profile is spreading over a finite perimeter (a finite width for the equivalent unfolded flow). To account for that, a dissipation flux coefficient may be introduced that measures the dissipation flux rate across the entire cross-sectional area $\mathcal{S}_\delta$.

$$C_D^* = \frac{1}{\rho_e V_e^3} \iint (\bar{\tau} \cdot \nabla) \cdot \mathbf{V} d\mathcal{S}_\delta \simeq C_D \bar{b} \quad (9)$$

The right-hand side C_D^* approximation assumes, at each position s , a constant dissipation-weighted average value, $\bar{b}$, across the shear layer height [41], such that $b(s, n) = \bar{b}(s) \simeq \bar{b}(s)$. Thereby, the approximation relates the dissipation coefficient value to the corresponding cross-sectional dissipation flux coefficient C_D^* via the averaged lateral width parameter $\bar{b}$. The above definitions are sketched in Fig. 3 and indicate that the streamwise rate of change of viscous dissipation, $d\Phi/ds$ of axisymmetric boundary layers is governed by the dissipation flux coefficient:

$$\Phi_{BL} + \Phi_{wake} = \int_{BL} \rho_e V_e^3 C_D^* ds + \int_{wake} \rho_e V_e^3 C_D^* ds \quad (10)$$

By definition, C_D and C_D^* are strictly positive scalars and therefore independent of the nearfield surface orientation. Thus, the local evolution of C_D for 2D and C_D^* for axisymmetric flows can be meaningfully traced along the entire flow path including partial airframe surfaces and wake shear layers to establish causality between dissipation accumulation and local viscous layer characteristics. The latter are then conveniently analysed in terms of relevant boundary layer parameters for axisymmetric flows, as discussed by Drela [41]. The V_e^3 factor underlines that severe dissipative loss generation manifests along regions of excessive acceleration (nose) while large C_D values spanning over deceleration regions (aft cone region) are partially offset by a corresponding reduction in V_e^3 .

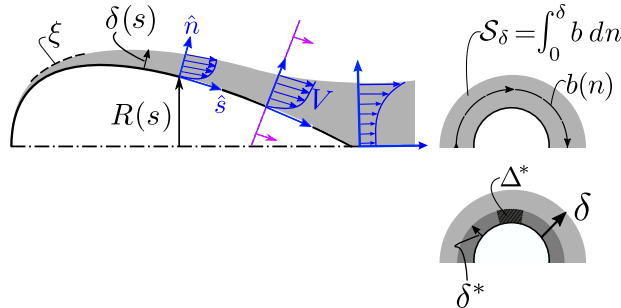


Fig. 2 (Left) 2D axisymmetric body cutaway with local coordinate system definitions and annotated boundary layer features. (Right) Backview of the cross-sectional area (corresponding to the pink dashed line) shows the transverse area, $\mathcal{S}_\delta$ and the local width parameter, b (Bottom right) Backview with defect area and corresponding 2D thickness definitions.

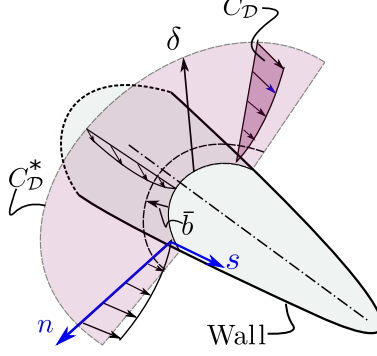


Fig. 3 Aft region of axisymmetric body in perspective with annotated C_D and C_D^* definitions corresponding to the shear layer profile and shear layer cross-sectional area S_δ , respectively.

D. Drag Power and Mechanical Loss Generation

A more intuitive form of the energy balance (2) is obtained if the Trefftz plane of the CV of Fig. 1 is positioned infinitely far downstream such that $\dot{\mathcal{E}}_{TP} = 0$ to (asymptotically) enclose all the mechanical losses of the flowfield. For this CV, the energy balance yields a unique decomposition of the aerodynamic drag power as the sum of the individual mechanical loss components inside the flowfield:

$$(D_p + D_v)V_\infty = \Phi + \mathcal{B} = \Phi^* = \Phi_{BL}^* + \Phi_{wake}^* + \Phi_{shock}^* = \Phi_{BL} + \Phi_{wake} + \Phi_{shock} + \Theta_\infty \quad (11)$$

The shear layer dissipation characteristics (Φ_{BL} , Φ_{wake}) are then conveniently analysed via the dissipation flux coefficient relation (10). The second right-hand side equality of (11) provides a numerically advantageous way to obtain the total baroclinic power by substituting its approximate estimation from Eq. (A.1) assuming fully-expanded flow in the wake. Eq. (11) underlines the major advantage of energy-based approaches over momentum-based build-ups in establishing causality between the aerodynamic force generation and its mechanical loss manifestation in the farfield. For example, two bodies flying at the same speed $-\mathbf{V}_\infty$, may experience the same drag but for different loss combinations. From an energy recovery perspective, strategic manipulation of the loss distribution may, for example, aid to the pre-shaping of a fuselage for optimal energy recovery via BLI.

E. Potential for Energy Recovery

A further implication of the mechanical loss distribution is the concept of the potential for energy recovery, initially introduced in [22] and modified in [42] to account for the baroclinic power contribution. It quantifies the residual mechanical energy portion at a given Trefftz plane position along the wake and is defined as the ratio of the downstream

residual wake mechanical losses , $\Phi_{\text{wake,r}}^*$, to the total mechanical power loss (drag power):

$$\mathcal{PER} = \frac{\Phi_{\text{wake,r}}^*}{DV_\infty} = 1 - \frac{\Phi_{\text{TP}}^*}{DV_\infty} \cong 1 - \frac{\Phi_{\text{TP}} + \Theta_\infty}{DV_\infty} \quad (12)$$

where the second right-hand side equality has been recast in terms of the mechanical losses upstream of the Trefftz plane, Φ_{TP}^* . The $\mathcal{PER}$ factor can be viewed as a measure of the maximum benefit that could hypothetically be extracted from the flow via an idealised[‡] (BLI) propulsor located at the trailing edge of the airframe or along the wake region by completely eliminating the downstream wake dissipation [38, 42]. Whereas $\mathcal{PER}$ is a conceptually useful indicator of potential BLI benefits, its maximisation does not necessarily yield an optimal fuselage design or optimal aerodynamic performance of a BLI-propelled aircraft. Instead, the total power consumption of the vehicle must be considered (for a given flight condition) and therefore the total mechanical losses (dissipative and baroclinic) of the system. The $\mathcal{PER}$ definition (12) may be viewed as a conservative upper limit due to the assumption that the hypothetical energy recovery process does not recover any baroclinic power portion affiliated with the semi-expanded wake region downstream of the trailing edge.

The exact numerical implementation of the $\mathcal{PER}$ definition within the CFD is only possible for incompressible flow ($\mathcal{B} = 0$), or for a CV that ensures fully expanded flow ($\mathcal{B} = \text{const.} = \Theta_\infty$), thereby the baroclinic losses are fully determined. In the near wake region, the viscous fluid flow is subject to the pressure field of the body so that the exact baroclinic power contribution cannot be extracted. Instead, the last right-hand approximation of Eq. (12) will be employed, based on the assumption that $\mathcal{B} \simeq \Theta_\infty$ [39].

F. Potential for Heat Exergy Recovery

The inclusion of exergetic heat management aspects in a mechanical energy analysis is discussed by Lamprakis et al. [38] via a hybrid energy balance. The proposed format unifies the heat management capabilities of an exergy analysis with the inherent intuitiveness of traceable mechanical energy inflows, outflows and work flow paths in the flowfield. For a configuration with heat recuperation capabilities, the potential for heat exergy recovery $\mathcal{PER}_\vartheta$ [38] is an exergy-based metric that quantifies the ideal heat-to-work conversion process via a Carnot engine when heat sources are present in the flowfield, for example conductive heat transfer sources due to heated walls. In the absence of volumetric heat addition/absorption, $\mathcal{PER}_\vartheta$ is defined as the ratio of the heat exergy due to conduction, $\dot{\mathcal{E}}_{q_B}$ to the drag power, $D^\vartheta V_\infty$, of the heated aerodynamic body:

$$\mathcal{PER}_\vartheta \equiv \frac{\dot{\mathcal{E}}_{q_B}}{D^\vartheta V_\infty} \quad (13)$$

where the upperscript ϑ denotes that the airframe includes non-adiabatic wall surfaces.

[‡]The propulsor does not interact with the flow-field of the body and restores, in the absence of internal losses, instantaneously freestream conditions downstream

$\dot{\mathcal{E}}_{q_B}$ measures the heat exergy associated with non adiabatic wall surfaces with finite temperature difference relative to the environment, T_∞ [23]. This term is strictly positive regardless of the heat transfer direction (cooling or heating) and measures the ideal heat work potential of a Carnot heat engine.

$$\dot{\mathcal{E}}_{q_B} = \iint - \left(1 - \frac{T_\infty}{T}\right) \dot{q} \cdot \hat{n} dS_B \quad (14)$$

where T the static temperature and $\dot{q} = -\mathcal{K}\nabla T$ the conductive heat flux which is proportional to the local temperature gradient via the thermal conductivity, $\mathcal{K}$, according to Fourier's Law. By definition, $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$ is always positive regardless of the direction of heat transfer [38] thereby monotonically accumulating along the airframe and attaining its maximum value when all the flowfield heat sources are enclosed by the CV. Thus, the $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$ definition (13) does not require the inclusion of the entire body surfaces, instead, it can be measured along partial wall surfaces with heat transfer that for example are of interest for vehicles with on board fuel cells, batteries and heat exchangers. To further clarify, Fig. 4 illustrates the idealised heat exergy recovery process for a decoupled fuselage-propulsor configuration with on board Carnot engine. The hypothetical Carnot engine utilizes the heated wall surfaces with temperature $T_w > T_\infty$ and the surroundings with freestream temperature, T_∞ , as a hot and cold reservoir, respectively (and vice versa for cooled walls). The extracted heat exergy $\dot{\mathcal{E}}_{q_B} = \mathcal{P}\mathcal{E}\mathcal{R}_\theta D^\theta V_\infty$ is subsequently transmitted, for example via a battery, to the propulsion system to reduce its power consumption in propelling the fuselage at a velocity of $-\mathbf{V}_\infty$.

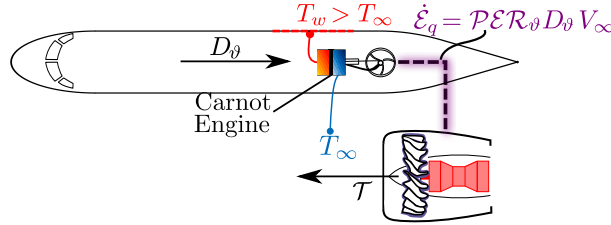


Fig. 4 Ideal heat-to-work conversion process from a hypothetical Carnot engine. The heat exergy is transmitted to the propulsion system to reduce its power consumption.

III. Case Description

A. Mechanical Energy Similarity

The aerodynamic characteristics of non-lifting flows over 2D-Axisymmetric fuselage designs are assessed in terms of their mechanical energy characteristics, including loss generation and recuperation mechanisms, the latter being relevant to aero-propulsive technologies incorporating BLI propulsion. A certain degree of generality and scalability may be established by carrying out an analysis based on "dynamically-similar" aerodynamic flows over "geometrically-similar" fuselage designs [36]. In the context of a mechanical energy analysis, the governing flow scaling parameters are obtained

by non-dimensionalising the steady state mechanical energy equation's variables as:

$$\tilde{[]} = \frac{[]}{[]_{\infty}} \quad (15)$$

where $\tilde{[]}$ represents any non-dimensional flow variable, $[]_{\infty}$ any freestream related variable and along with ideal gas law relations, the following dimensionless differential form of the mechanical energy equation is obtained:

$$\tilde{\nabla} \cdot \left(\frac{1}{2} \tilde{\rho} \tilde{\mathbf{V}} \tilde{V}^2 \right) = -\frac{1}{\gamma M_{\infty}^2} \left(\tilde{\nabla} \cdot (\tilde{p} \tilde{\mathbf{V}}) - \tilde{p} \tilde{\nabla} \cdot \tilde{\mathbf{V}} \right) + \frac{1}{Re_{\infty}} \left(\tilde{\nabla} \cdot (\tilde{\bar{\tau}} \cdot \tilde{\mathbf{V}}) - (\tilde{\bar{\tau}} \cdot \tilde{\nabla}) \cdot \tilde{\mathbf{V}} \right) \quad (16)$$

where $\tilde{\nabla} = L \nabla(\cdot)$, γ the specific heat ratio, $Re_{\infty} = \rho_{\infty} V_{\infty} L / \mu_{\infty}$ the freestream Reynolds number for a given characteristic length L , $M_{\infty} = V_{\infty} / \alpha_{\infty}$ the freestream Mach number and α_{∞} the speed of sound. The mechanical similarity [37] is therefore imposed by the Re_{∞} and M_{∞} numbers alone for viscous and pressure terms, respectively (assuming the specific heat ratio γ constant), while the freestream Prandtl number, Pr_{∞} [§] is not explicitly affecting the scaling of the mechanical energy terms. For low subsonic flows, the incompressibility assumption further drops the M_{∞} dependence and the implicit effect of varying the freestream Prandtl number [43]. For typical adiabatic flows, the Prandtl number is nearly invariant with values close to unity, so that mismatches in this parameter have only minor effects to the flowfield [36]. Even if significant heat transfer effects are present due to volumetric heating or heat conduction, these will manifest themselves as local changes in the mechanical response of the fluid which are traceable via the terms of the mechanical energy equation. The "geometric similarity" is approximated by scaling a reference fuselage design and using the fineness ratio as the driving parameter defined by the ratio of the fuselage chord length c to the fuselage diameter D . Thus, the overall shape of each of the six fuselage designs $FR = 4, 6, 8, 10, 12, 14$, shown in Fig. 5, is a predetermined scaled version of a $FR = 10$ (blue curve) reference design of fixed chord length, c with scaled nose and aft cone curvatures. Wall heating is examined for the same fuselage geometries by imposing uniform wall temperature ratios, T_w / T_{∞} , where T_w the wall-temperature and $T_{\infty} = 300$ K the freestream temperature which is held constant throughout.

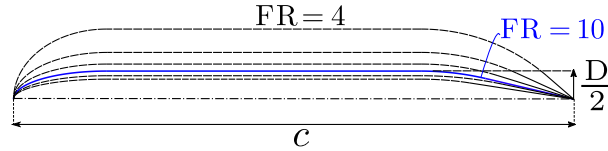


Fig. 5 Scaled fuselage geometries for varying fineness ratios based on a $FR=10$ reference design (blue curve).

B. CFD Setup

Figure 6 summarizes the parametrisation of the 2D - axisymmetric CFD domain and the cylindrical control volume along with annotated geometric features relevant for the scaling of the fuselage. The fuselage is constituted by an

[§]Dynamic similarity requires the matching of Pr_{∞} . In that sense, the flow is only partially similar"

elliptical nose of length l_{nose} , a cylindrical midbody and a tail cone section of length l_{tail} which consists of a circular arc segment of radius r_{tail} tangential to the midbody and a right-angled cone portion intersecting the fuselage centerline at an angle β . Pressure far field conditions are imposed at the front and upper outer portions of the CFD domain and a static pressure outlet condition at the vertical rear (Trefftz) plane. The frontal and upper pressure farfield CFD domain portions are positioned equidistant from the fuselage at $d_{\text{farfield}} = 30 \times c$, while the pressure outlet boundary at $d_{\text{outlet}} = 80 \times c$. The cylindrical control volume is meshed as a C-type structured grid using the ANSYS ICEM CFD package and sized such that all the wake and shock-related mechanical energy outflows exit only from the the Trefftz plane. Thereafter, numerical solutions of 2-D axisymmetric RANS studies are obtained using the commercial CFD package ANSYS FLUENT and post-processed in MATLAB. Turbulence is modelled using the two-equation eddy-viscosity $\kappa - \omega$ SST model coupled with second order spatial discretization and the Green-Gauss Cell-Based gradient scheme. Mesh refinement has been applied along the fuselage surfaces to ensure the conformity of the mesh with the $y^+ < 1$ requirement of the turbulence model, resulting in $0.6 - 1 \times 10^6$ grid elements, depending on the geometry and flow conditions. A wall-normal growth ratio of 1.05 has been imposed along the fuselage and the wake region to ensure high spatial resolution of the viscous losses and boundary layer characteristics. The domain size independence studies, overall mesh refinement and grid convergence strategy have been adopted from Reference [22] and are omitted herein. For completeness, mesh details of an FR = 8 fuselage operating at $Re_c = 1 \times 10^6$ are provided in Fig. 7 a) and b) for the nose and aft cone - near wake region, respectively. The fluid flow is modelled as an ideal gas with piecewise-polynomial C_p variation and temperature dependent viscosity according to Sutherland's law. The thermal conductivity is held constant for flows with adiabatic walls and varies according to kinetic-theory when wall heating is imposed. The full matrix of CFD cases is summarised in Table 1 as obtained by independently varying M_∞ , Re_c , T_w/T_∞ and FR for a total of 882 combinations.

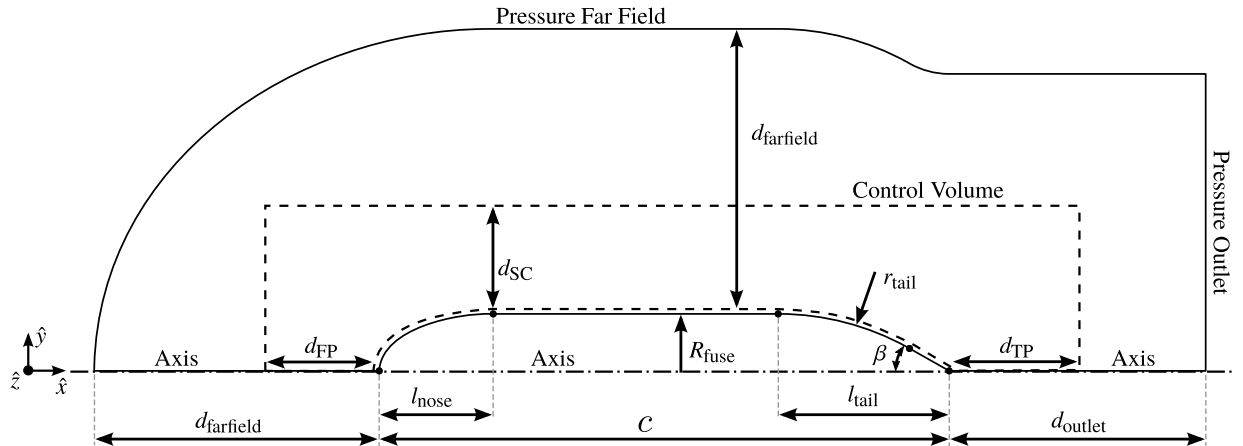


Fig. 6 2D axisymmetric CFD domain cutaway with annotated geometric features, boundary conditions and cylindrical control volume definition, adapted from [22].

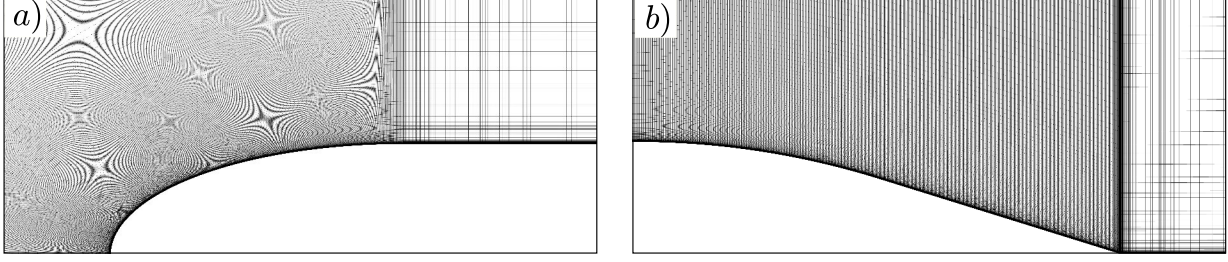


Fig. 7 Structured mesh sample of an $FR = 8$ fuselage at $Re_c = 1 \times 10^7$ with a total of $\simeq 1.4 \times 10^6$ cells: a) Nose section. b) Aft body and near wake region.

Table 1 Parametric combinations resulting in 126 adiabatic and 756 heated cases.

Parametric combinations	
M_∞	0.4, 0.6, 0.8
FR	4, 6, 8, 10, 12, 14
T_w/T_∞	Adiabatic, 1.5, 2.0, 2.5, 3.0, 3.5, 4.0
Re_c	5×10^5 , 1×10^6 , 5×10^6 , 1×10^7 , 5×10^7 , 1×10^8 , 5×10^8

C. CFD Post-processing of Shear Layer Quantities and Shock Waves

1. Shear Layer Quantities

For axisymmetric viscous flows, the corresponding shear layer quantities are integrated over the local shear layer transverse area S_δ , bound by the local radius of the axisymmetric body $R(s)$ and the boundary layer thickness $\delta(s)$ as shown in Fig. 2. The local cartesian s-n coordinate system is defined to follow the curvature of the body and the symmetry axis along the wake region. The boundary layer thickness distribution, $\delta(s)$, is identified from the total pressure losses, similar to Aguirre et al. [24] and is numerically approximated hereafter as the iso-surface corresponding to a threshold value of ξ that approaches unity:

$$\xi = \frac{P_t}{P_{t_\infty}} \rightarrow 1 \quad (17)$$

where P_t the isovalue of the total pressure for a given ξ threshold and P_{t_∞} the freestream total pressure. The ξ threshold is set to a value close to unity and is fine-tuned to preclude flowfield regions with marginal numerical dissipation, such as the near leading edge stagnation region. For all the subsequent CFD studies, a threshold value of $\xi \simeq 0.9996$ has been found to yield the most consistent balance between high BL resolution and suppression of spurious effects. By definition, the identifier excludes a portion of the upper boundary layer that depends on the selected threshold (here $0.9996 < \xi < 1$), however the dissipation associated with it is marginal and less than 1% for the examined dataset [39]. Examples of shear layer extraction using the ξ identifier for adiabatic and heated fuselages of $FR = 4$ and $FR = 10$ are shown in Fig. 8 a) and b), respectively. A threshold value of $\xi = 0.9996$ is set for the extraction of the viscous domain with the respective iso-surface (blue line) denoting the boundary layer thickness δ . Therefore, the various shear layer edge properties that are required for the following shear layer parameters are readily obtained from the reported CFD

data that correspond to the $\xi = 0.9996$ iso-line. A spurious region of marginal dissipation is observed for the adiabatic (upper half), which however does not affect the calculations. If required, a slightly lower ξ threshold value may be chosen to smooth out the spurious regions.

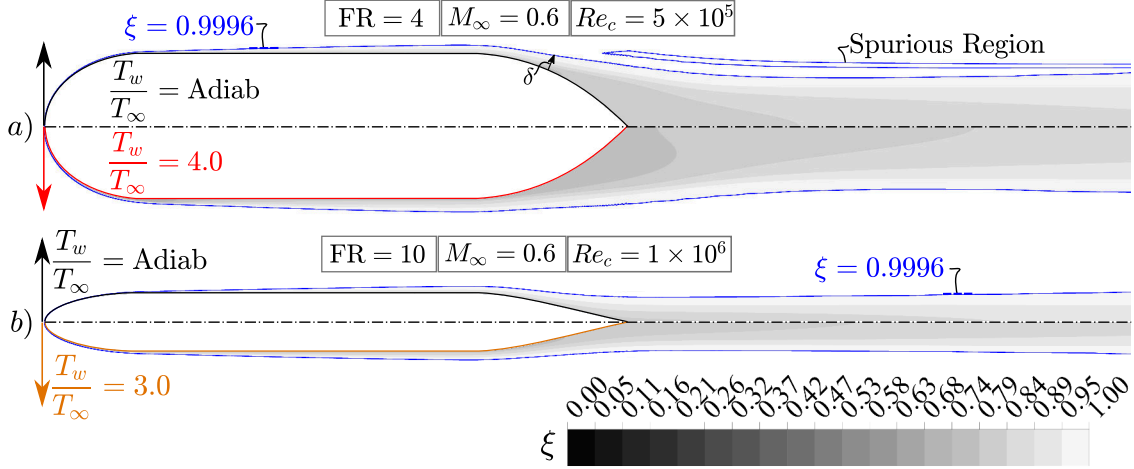


Fig. 8 Shear layer extraction using the ξ identifier with a threshold value of $\xi = 0.9996$ for adiabatic (upper half) and heated fuselages (lower half) and operating conditions: a) $FR = 4$, $M_\infty = 0.6$, $Re_c = 5 \times 10^5$ and $T_w/T_\infty = \text{adiab.}, 4.0$. b) $FR = 10$, $M_\infty = 0.6$, $Re_c = 1 \times 10^6$ and $T_w/T_\infty = \text{adiab.}, 4.0$.

Integral shear layer quantities are obtained at each position s from their axisymmetric definitions [41] while the affiliated local edge quantities are interpolated from the ξ iso-surface. The following defect areas are defined and used hereafter:

Mass defect area:

$$\Delta^* = \iint \left(1 - \frac{\rho V}{\rho_e V_e} \right) dS_\delta \quad (18)$$

Momentum defect area:

$$\Theta = \iint \left(1 - \frac{V}{V_e} \right) \frac{\rho V}{\rho_e V_e} dS_\delta \quad (19)$$

Density-flux defect area:

$$\Delta^{**} = \iint \left(1 - \frac{\rho}{\rho_e} \right) \frac{V}{V_e} dS_\delta \quad (20)$$

The corresponding boundary layer shape parameter H and density-flux shape parameter H^{**} are equivalent to their 2D definitions:

$$H = \frac{\Delta^*}{\Theta} \quad (21)$$

$$H^{**} = \frac{\Delta^{**}}{\Theta} \quad (22)$$

For further purposes, it is convenient to relate the preceding defect area definitions (18) - (20) to their corresponding thickness. The bottom right sketch of Fig. 2 shows an example definition of a defect area and its affiliated thickness

at an arbitrary position along the body. The relation between the two is obtained by expressing the transverse area element dS_δ as a function of the local width parameter $b(n)$ definition (8). For example, the viscous displacement, δ^* , corresponding to a given mass defect area Δ^* is numerically obtained as:

$$\delta^* = \iint \left(1 - \frac{\rho}{\rho_e}\right) \frac{V}{bV_e} dS_\delta \quad (23)$$

2. Shock Waves

The shock wave regions are numerically identified using the shock identifier of Lovely and Haines [44] for steady flows. The identifier ζ traces cells with local Mach number normal to the maximum pressure gradient direction equal to unity:

$$\zeta = \frac{\mathbf{V} \cdot \nabla p}{\alpha |\nabla p|} \quad (24)$$

In practice, a threshold value of $\zeta \simeq 0.95$ has been found to provide adequate treatment for the inclusion of the numerical shock domain, in agreement with previous observations by [22, 45], whereas detailed discussions on the identification of boundary layer and shock wave regions are given in Reference [46].

IV. Characteristics of Adiabatic Axisymmetric Fuselages

A. Drag Power Characteristics

The normalised nearfield pressure $\mathcal{R}_p = D_p/D$ and friction $\mathcal{R}_v = D_v/D$ drag ratios (power) variations at fixed $M_\infty = 0.4$ are shown in Fig. 9 a) as a function of the FR for varying Reynolds number. The normalised components of pressure drag and friction drag decrease and increase, respectively, by a proportional amount with an increase in Re_c and FR. For this Mach regime, the local bulk flow acceleration is insufficient to trigger shock formation so that the two aerodynamic force components are governed by the boundary layers' characteristics exclusively. The proportionality of the the local friction force to the wall velocity gradients results in a direct Re_c dependence via the effect of the latter on the boundary layer shape parameter H . Higher Re_c flows favour lower H values (and ultimately higher D_v) as shown in Fig. 11 a) and thus allow the boundary layer to sustain stronger adverse pressure gradients at the aft cone region prior to separation ($H \cong 4$) at the expense of increased friction force and viscous dissipation [41]. The above have an indirect implication on the pressure drag via the inverse relation of Re_c to the viscous displacement, δ^* , of the flow. From the von Karman integral momentum equation, the local pressure defect evolution of 2D and axisymmetric shear layers is governed by the $-\rho_e V_e \delta^* \frac{dV_e}{ds}$ term, indicating conceptually that pressure drag is generated at airframe regions with large viscous displacement under adverse pressure gradients ($dV_e/ds < 0$) [43]. These conditions are mainly present at the aft region of airfoils and the aft cone region of axisymmetric bodies yielding an equivalent inviscid body (EIB) that diverges locally from the real by a wall-normal distance (approximately) equal to the mass defect, δ^* and accounts for

the viscous deflection of the inviscid bulk flow away from the wall [41]. Fig. 10 a) shows a $FR = 8$ fuselage aft cone at a fixed $M_\infty = 0.4$ with varying Re_c EIBs superpositioned, indicating that higher Re_c flows suppress the viscous displacement effect and the affiliated pressure drag penalty. The abrupt upward shift of the H evolution in Fig. 11 c) along the aft cone region for blunter bodies signifies substantial thickening of the boundary layer (for $FR > 6$), local flow reversal (for $FR = 6$) and separation bubble formation (for $FR = 4$) in response to the progressively stronger adverse pressure gradients imposed by the aft cone curvature. From the above, the aft cone region of a progressively smaller FR fuselage shape implies higher D_p contributions due to stronger adverse pressure gradients acting on larger δ^* values, the latter being further magnified, relative to a 2D flow, by the viscous flow path enclosing a progressively reduced perimeter along the aft cone region (due to conical shape converging).

The effect of compressibility on the nearfield drag components is shown in Fig. 9 b) in terms of the viscous (blue) and pressure (orange) drag ratios for $M_\infty = 0.6, 0.8$, normalized by the corresponding $M_\infty = 0.4$ values. The trends indicate a favourable D_v reduction effect with larger benefits for lower Re_c and FR values and a subsequent proportional penalty on the D_p component. In the absence of shocks, fluid expansion and compression domains span locally over regions of strong curvature and stagnation points that induce locally strong coupled thermo-compressible effects. These are associated with subsequent density variations inside the viscous layers, the latter being described by the density flux thickness shape parameter, H^{**} , as shown in Fig.11 b). The local distribution of H^{**} along the fuselage indicates a stream-wise invariant step increase along the cylindrical midbody and an amplification effect on the local fluctuation over highly expansive and compressive regions close to the nose and aft cone. The amplification effect is more pronounced at increased M_∞ and lower FR shapes due to the simultaneous presence of stronger flow compressibility and curvature-induced pressure gradients as shown in Fig.11 b) and c), respectively. The implication of the higher H^{**} values is the compressible thickening of the EIB via the larger viscous displacement, δ^* as shown in Fig. 10 b) with a subsequent D_p penalty alongside a corresponding increase in the shape parameter distribution of H (see Fig. 11b)) resulting in a local decrease of the wall normal velocity gradient. Finally, the effect of increasing Re_c is only weakly-coupled to compressibility via the self-heating of the shear flow due to viscous dissipation, resulting in marginally larger H^{**} values.

B. Mechanical Energy Loss Generation

1. Dissipation

The accumulation of dissipation along the fuselage and the wake are shown in Fig. 12 a) for different fineness ratios and fixed Re_c and M_∞ numbers. In the absence of shocks, the mechanical losses manifest themselves primarily as viscous dissipation inside boundary and shear layers. For all fuselage fineness ratios considered, dissipative loss generation is more pronounced near the leading edge of the fuselage where the flow rapidly accelerates, whereas the opposite behaviour is observed close to the aft cone region due to flow deceleration that results in a subtler $d\Phi/dx$ slope.

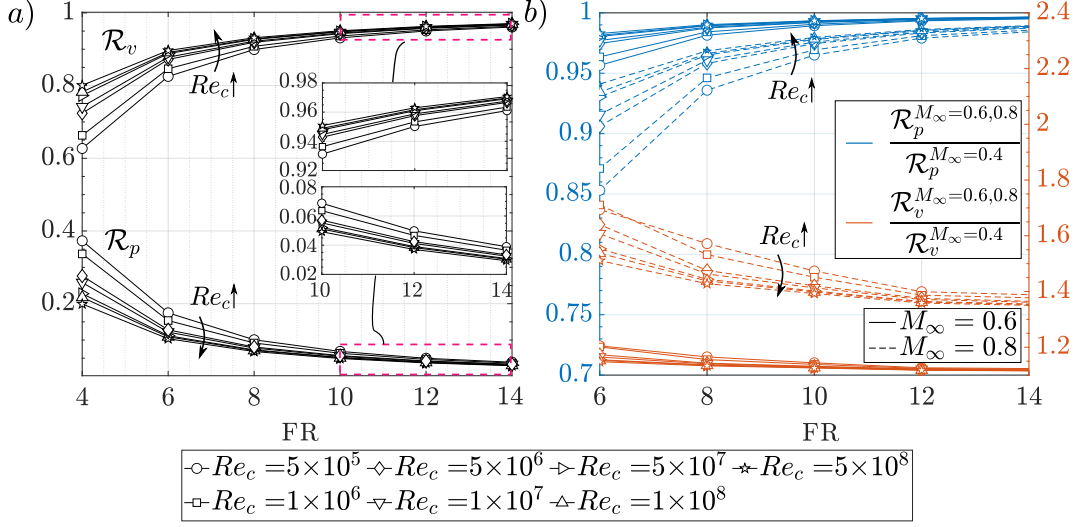


Fig. 9 a) Normalised nearfield pressure and friction drag ratios at $M_\infty = 0.4$ as a function of FR for varying Re_c . b) Ratios of normalised nearfield pressure and friction drag components of $M_\infty = 0.6, 0.8$ as a function of FR.

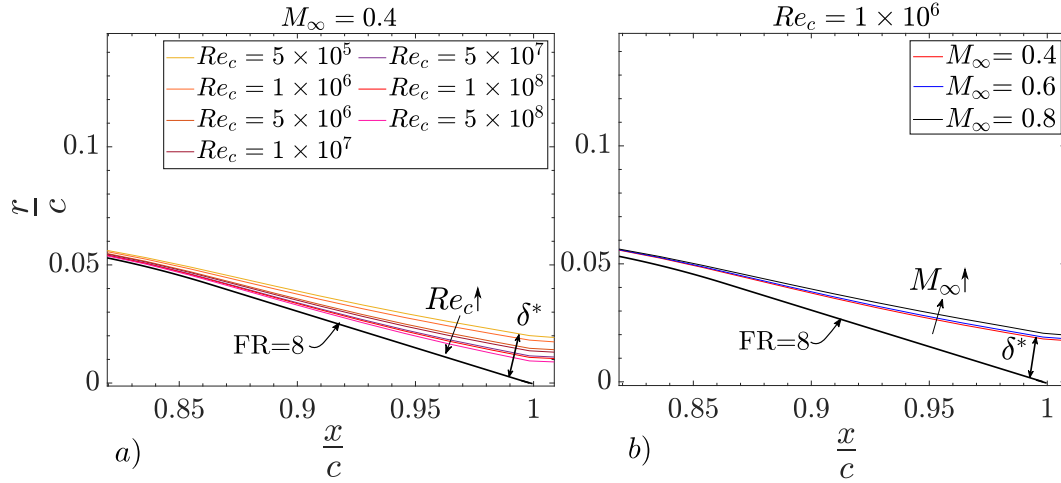


Fig. 10 a) Viscous displacement evolution resulting in an equivalent inviscid body as a function of varying Re_c number for $FR=8$ and $M_\infty = 0.4$. b) M_∞ effect on viscous displacement evolution for fixed $Re_c = 1 \times 10^6$ and $FR=8$.

It is then convenient to trace the evolution of dissipation losses[¶] in terms of the local distribution of the dissipation coefficient as shown in Fig. 12 b). Equations (9) and (10) indicate that C_D^* controls the rate of change of the accumulated viscous dissipation evolution, $d\Phi/dx$ for fixed $\rho_e V_e^3$ conditions, while the V_e^3 exponent acts as dissipation amplifier or damper over accelerative and decelerative regions, respectively.

The 2D dissipation coefficient C_D remains relatively constant for weak flow acceleration/deceleration [20] such that changes in C_D^* primarily indicate large variations in b and therefore wetted area. This is indicated in Fig.12 b) by the

[¶]the numerical calculation of Φ is obtained by its volume integral definition (4) while the dissipation relation (10) is used as a conceptual aid to interpret the flow physics.

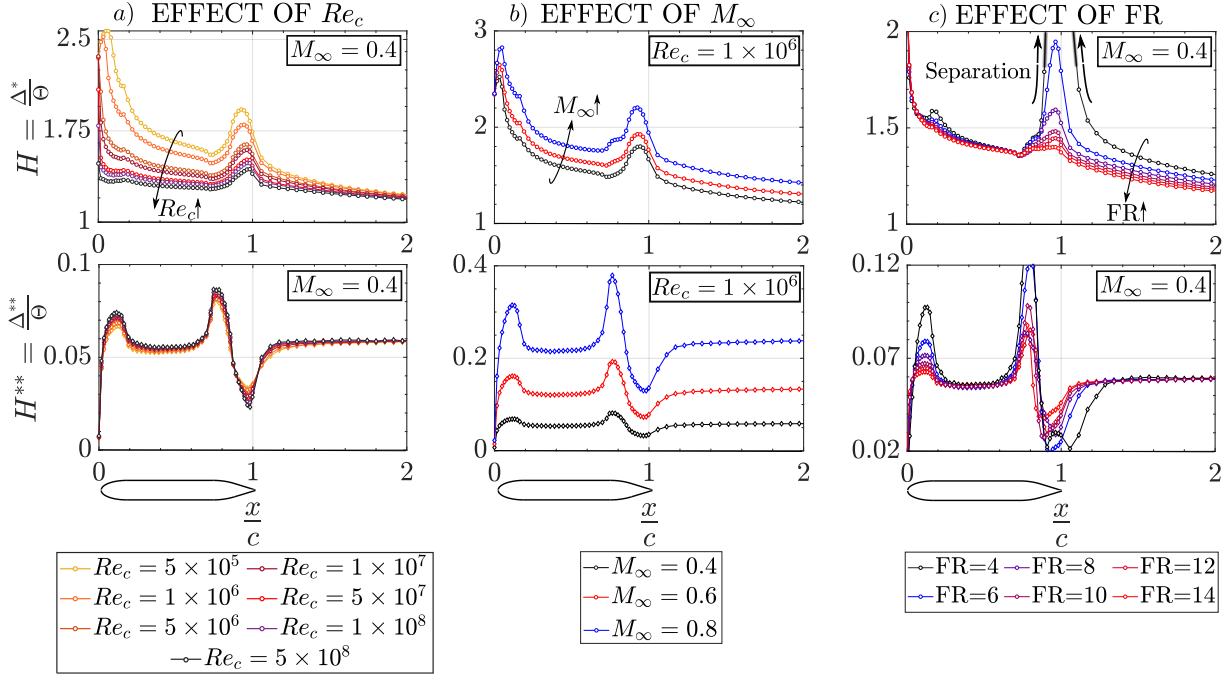


Fig. 11 Evolution of the boundary layer's shape parameter (H) and density flux parameter (H^{**}) along the body and the wake of the fuselage for varying: a) Re_c . b) M_∞ . c) FR.

progressively steeper rate of change of C_D^* over the nose ($x/c < 0.15$) and aft cone ($0.8 < x/c < 1$) regions, which are amplified for lower FR shapes, due to substantial cross-sectional area variations (captured by the b parameter) and the induction of stronger favourable and adverse pressure gradients, respectively. The implication is that a locally steep rise in dC_D^*/dx entails excess dissipation due to acceleration and wetted area increase that manifests as a proportional increase in $d\Phi/dx$ and vice versa, as shown in Fig.12 a). Within the wake, the gradual viscous decay of the residual mechanical flux perturbation energy leads to a monotonic reduction of $d\Phi/dx$ and C_D^* in Fig. 12 a) and b), respectively. For FR = 4, the abrupt upward shift of C_D^* along the aft cone (highlighted in yellow) indicates the formation of a separation bubble (see Fig. 14 a) as an example). The bubble forms an effective obstacle that increases locally b and forces the bulk boundary layer flow to to accelerate, as sketched in Fig. 13. Hall et al. [32] further address separation in terms of C_D as a mechanism that shifts localised dissipation away from the wall, that is, the separation bubble is merely an implicit loss mechanism that induces dissipation by thickening the effective shape of the body.

Superimposed in Fig. 12 a) and b) are C_D^* plots for $M_\infty = 0.6$ and $M_\infty = 0.8$ which correspond to the pink and blue dashed lines, respectively. Higher M_∞ numbers result in reduced local C_D^* values along the airframe surfaces with a subsequent beneficial reduction in the accumulated boundary layer dissipation fraction (see subplot of Fig.12 a)). This is partially due to the slight increase of the boundary layer shape parameter distribution shown in Fig. 11 b) and partially due to the denominator's additional compressibility-related drag power, described in Section IV.A. Higher M_∞ flows give rise to "compressibility-induced separation", that is, for a given aerodynamic body at fixed Re_c , the

formation and characteristics of the separation bubble are dependent on compressibility. This is demonstrated in Fig. 14 a) in terms of juxtaposed streamlines along the aft body portion of an FR = 4 fuselage operating at fixed $Re_c = 1 \times 10^6$, but different M_∞ numbers. The higher $M_\infty = 0.6$ scenario triggers the earlier onset of the separation bubble due to the compressible thickening of the EIB that manifests, ultimately, as a more extensive separation region relative to the $M_\infty = 0.4$ scenario. A more extensive recirculation region would then correspond to a C_D^* spike of higher magnitude in Fig.12 a), whereas a flow with weak flow reversal yields a subtler loss penalty. The latter scenario is shown in Fig. 14 b) in terms of Mach number contours that correspond to the $M_\infty = 0.4$ (dashed brown) and $M_\infty = 0.8$ (dashed blue) curves of the FR = 6 fuselage in Fig.12 b). In this scenario, the higher $M_\infty = 0.8$ scenario yields local flow reversal in the near vicinity of the trailing edge which results to a subtle C_D^* peak (highlighted in blue). An interesting implication is that the boundary layer's tolerance to separation can be enhanced by introducing excess dissipation to the flow (lower H values). An example is heat conduction (wall cooling) from the airframe's walls to partially negate the undesirable compressible displacement of the flow along the aft cone region.

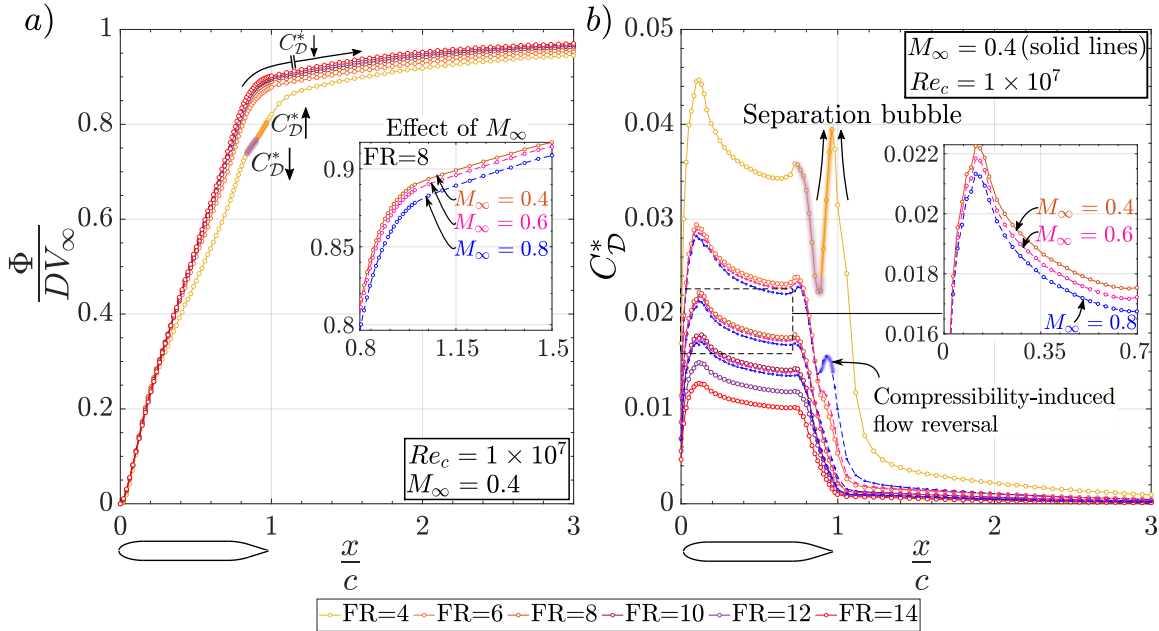


Fig. 12 a) Normalised dissipation evolution along the fuselage and the wake at fixed $Re_c = 1 \times 10^7$ and $M_\infty = 0.4$. Subgraph shows plots of the FR = 8 fuselage for varying M_∞ number. b) C_D^* evolution along the body for varying FR at fixed $Re_c = 1 \times 10^7$ and $M_\infty = 0.4$ (solid lines). C_D^* of varying M_∞ numbers are superimposed (dashed lines) and zoomed as a subplot.

2. Baroclinic Power

The compressibility losses manifest, for a fully expanded domain, as a residual mechanical loss component, i.e. $\Theta_\infty > 0$ regardless of operating conditions and shape characteristics, as shown in Fig. 15 a) and b). In the absence of shocks, compressibility yields a loss less than 2% for the entire dataset of flow conditions and fuselage bluntness tested.

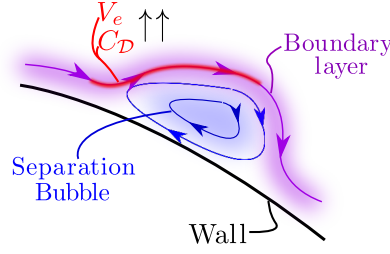


Fig. 13 Boundary layer accelerating (red region) over a separation bubble with a subsequent penalty in C_D and C_D^* values.

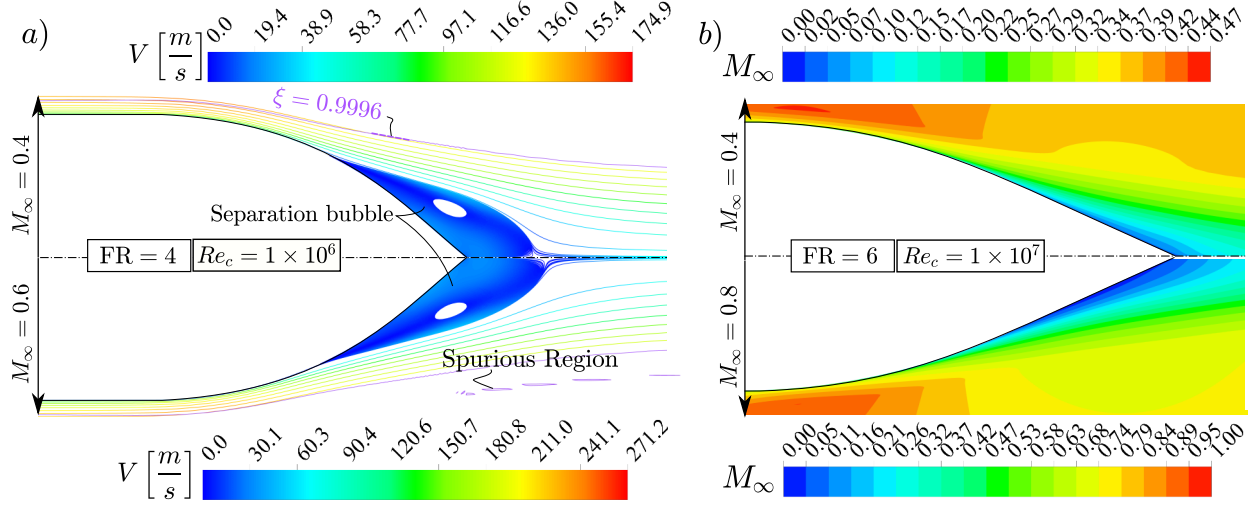


Fig. 14 Effect of Mach number on the aft cone boundary layer characteristics: a) Streamlines for juxtaposed $M_\infty = 0.4$ (upper) and $M_\infty = 0.6$ flows at fixed $FR = 4$ and $Re_c = 1 \times 10^6$ (lower). b) Mach contours for juxtaposed $M_\infty = 0.4$ (upper) and $M_\infty = 0.8$ flows at fixed $FR = 6$ and $Re_c = 1 \times 10^7$ (lower).

For the lower subsonic regime ($M_\infty = 0.4$), compressibility-induced losses are a second order effect and can be omitted so that the mechanical losses are measured by viscous dissipation, exclusively. The freestream Mach number is the driving non dimensional parameter for the scaling of the compressibility losses as indicated by the upward shift of the loss curves in Fig. 15 a) due to the step increase in H^{**} (and thus density defect), as shown in Fig. 11 b). The density defect variation is further amplified over the nose and aft cone regions of blunter bodies due to the stronger expansive and compressive regions that form in response to the steeper curvature changes of the body.

For subsonic flows, the compressibility losses are related to the viscous self-heating of the boundary layer, meaning that higher Re_c flows are more dissipative and deposit larger amount of heat along the body, thus enhancing the thermo-compressible interference along viscous expansion/compression zones. For transonic flows, the shock formation dominates the baroclinic losses due to the wave dissipation (Φ_{shock}) acting over a thin region of excessive compression, with stronger shocks being translated to larger loss contributions, as shown in Fig. 15 b) for $FR = 4$ (highlighted in purple). For the upper Re_c regime, the normal shock that forms downstream of the nose progressively strengthens, measuring $> 20\%$ of the total mechanical losses. A possible reason is that for higher Re_c number flows, the boundary

layer's momentum deficit is confined close to the wall, thereby reducing the sonic line's distance from the wall [47, 48] allowing a stronger shock to propagate across the outer portion of the boundary layer profile and vice versa. The increase of the baroclinic power with the upstream shock M number signifies its close relation to the underlying mechanisms of the entropy jump, δs , and the subsequent total pressure loss jump that scales as $\sim e^{-\delta s}$ across the shock as described by classical shock theory [49].

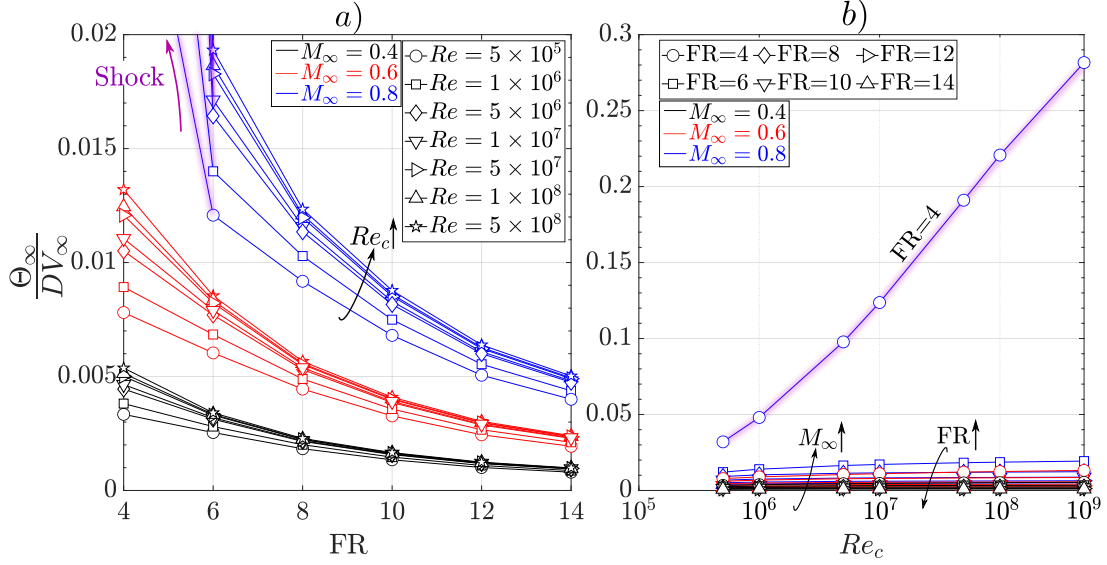


Fig. 15 a) Normalised baroclinic power against FR for varying Re_c . Data for FR=4 and $M_\infty = 0.8$ are omitted for clarity due to large shock contributions. b) Replotted normalised baroclinic power data against Re_c for varying FR.

C. Potential for Energy Recovery of Adiabatic flows

The conversion of useful mechanical flux energy, $\dot{E}$, to heat via viscous dissipation along the wake results in a monotonic downward trend of $\mathcal{PER}$ along the wake regardless of the aerodynamic body shape and operating conditions, as shown in Fig. 16 a) and b). A direct consequence is that the maximum $\mathcal{PER}$ value is attained at the trailing edge of the configuration. For fixed operating conditions, lower FR fuselage designs favour larger $\mathcal{PER}$ values by shifting progressively larger amounts of dissipation from the body to the wake region at the expense of increased pressure drag power proportions. This is because of the deceleration of the boundary layers along the aft cone region as indicated in Fig. 11 a) by the locally sharp increase of H . Similarly, higher Re_c flows favour more dissipative shear layer profiles as signified by the locally step decrease in the shape parameter H evolution in Fig. 11 a) and the subsequent step decrease of $\mathcal{PER}$ as shown in Fig. 16 b).

For BLI applications, the trailing edge $\mathcal{PER}$ value ($\mathcal{PER}_{TE}$) is the more relevant position so that a generalised non-dimensional $\mathcal{PER}_{TE}$ map may then be formed to assess the potential energy recovery benefits associated with BLI. Figure 17 shows $\mathcal{PER}_{TE}$ plotted against FR for varying Re_c at fixed $M_\infty = 0.4$ (black lines) and $M_\infty = 0.6$ (blue

dashed lines). The highest $\mathcal{PER}_{TE}$ values are attained for lower FR at higher M_∞ conditions, measuring up to 30% of the total drag power. Blunter bodies impose stronger pressure gradients over the aft cone region alongside a subsequent increase in the local evolution of H , as shown in Fig. 11 c) and thus an airframe dissipation reduction as per Fig. 12 a). The higher Mach number implicitly affects the flow-field via the compressible thickening of the flow displacement that decelerates the viscous flow at the expense of additional baroclinic losses. The latter are approximated herein as $\mathcal{B}_{TE} \cong \Theta_\infty$, thus the deviation from the exact $\mathcal{PER}_{TE}$ definition (12) may partially negate or even outweigh the seeming compressibility benefit. As expected, the strong dependence of viscous effects on the Reynolds number renders it as the governing flow parameter, with higher Re_c flows favouring airframe dissipation and thereby a subsequent reduction in $\mathcal{PER}$. For typical fuselage FR range and cruise conditions (highlighted in purple), the amount of drag power that could be harvested ideally from the flow is in the order of 8% – 10%.

D. Implications of $\mathcal{PER}$

The concept of $\mathcal{PER}$ has a few useful implications of fuselage design. The first implication is that low Re_c adiabatic viscous flows over blunter airframes offer the highest hypothetical potential for synergistic coupling with BLI propulsion. Therefore, the $\mathcal{PER}$ map of Fig. 17 indicates that fuselage designs of small, low-speed commuter aircraft or unmanned UAVs could benefit substantially from BLI propulsion. Future aircraft concepts with hydrogen-powered propulsion systems may also be good candidates for BLI propulsion since those typically feature wider fuselage designs to accommodate liquid hydrogen tanks.

A second implication is that, in the absence of shocks, the dynamic similarity requirements for a subsonic wind-tunnel fuselage model are sufficiently met by matching the Re_c alone, thereby ensuring the appropriate scaling of the governing viscous dissipation losses. Thus, the notable $\mathcal{PER}$ discrepancy due to Re_c mismatching between experimental (typically in the order of $\simeq 10^6$) and full-scale ($\simeq 10^8$) conditions calls into question the comparability argument [28] between experimental and full-scale BLI benefits.

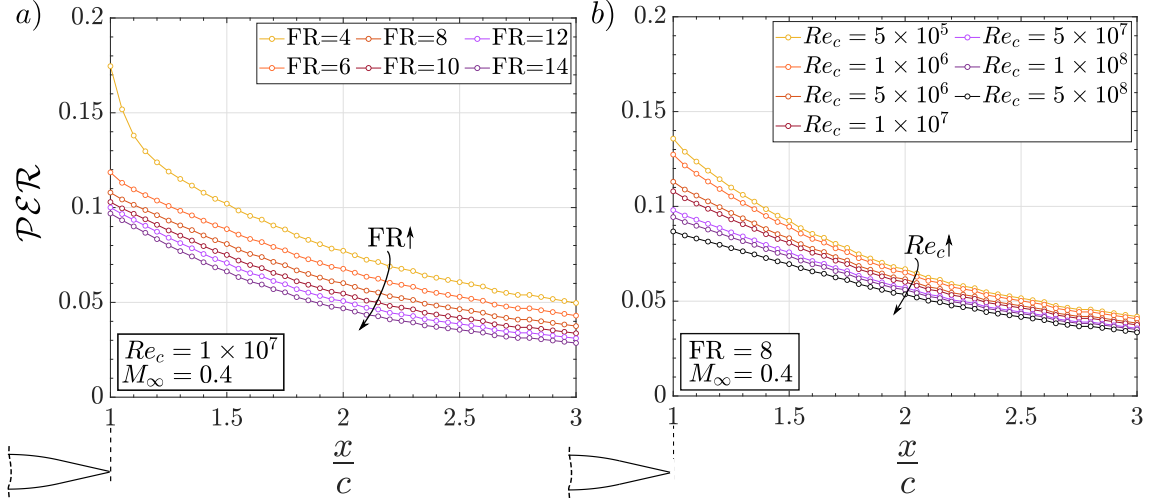


Fig. 16 $\mathcal{PER}$ evolution along the wake for: a) Varying FR at fixed $M_\infty = 0.4$ and $Re_c = 1 \times 10^7$. b) Varying Re_c at fixed $M_\infty = 0.4$ and $FR=8$.

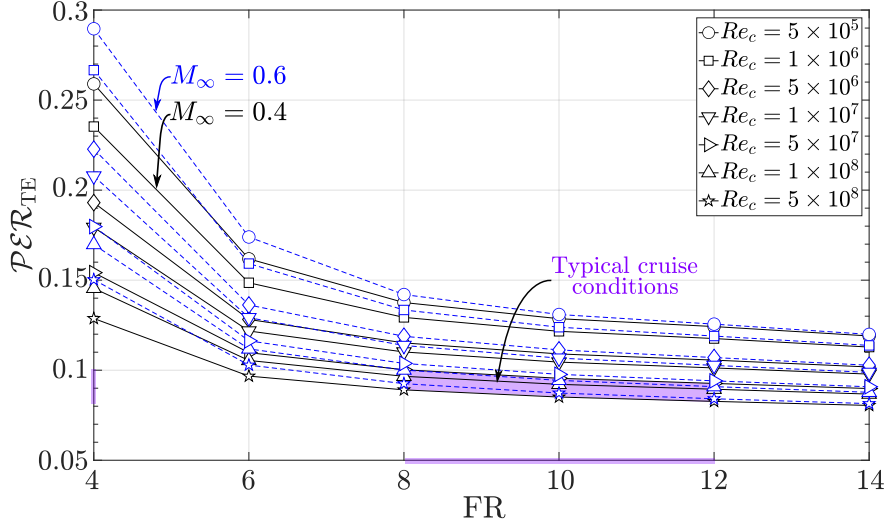


Fig. 17 $\mathcal{PER}_{TE}$ plots as a function of FR and for various Re_c . Highlighted red region corresponds to typical flight conditions and FR designs.

V. Characteristics of Heated Axisymmetric Fuselages

A. Wall Heating Effect on Nearfield Drag

Figure 18 a) shows plots of the heated fuselage nearfield drag D^θ normalised by the corresponding adiabatic D with respect to FR at fixed $M_\infty = 0.4$ for varying Re_c and wall temperature ratios T_w/T_∞ . Higher wall temperature ratios appear to tilt the drag ratio trends leading to drag penalty for blunt aerodynamic shapes ($FR \leq 6$) at low Re_c and to substantial drag reduction benefits for more streamlined aerodynamic bodies. This is explained by the opposing influence of wall heating on the near-field friction (D_v^θ) versus pressure drag (D_p^θ) components, in combination with the increasing pressure drag dominance corresponding to smaller FRs.

In the near-field, heat deposition modifies locally the boundary layer evolution characteristics by reducing the effective Reynolds number of the flow [33]. This is achieved via the large density defect imposed by the inverse scaling of density with temperature (ideal gas law) that outweighs the increase in the fluid's viscosity. In the nearfield, this manifests locally as an upward shift of the H and H^{**} values with increasing T_w/T_∞ along the fuselage as shown in Fig. 19 b) and c), respectively. The former signifies skin friction drag reduction ($D_v^\theta/D_v < 1$) as well as mitigation of the boundary layer dissipative characteristics while the latter represents a pressure drag penalty, ($D_p^\theta/D_p > 1$) due to the heat-induced viscous displacement thickening shown in Fig. 19 d). The opposing dependence of the drag components to wall heating indicates that the net heat transfer effect on the drag characteristics of aerodynamic bodies is circumstantial. The higher sensitivity of pressure drag on heat transfer effects (up to $3.3 \times D_p$ in 19 a)) implies that even aerodynamic shapes for which, the friction drag component measures the largest portion of the total profile drag, the associated heat-induced pressure drag penalty suffices to completely negate and outweigh any apparent benefits from skin friction drag reduction. For high M_∞ flows, the additional compressibility thickening of the EIF over an adverse pressure gradient zone further magnifies the overall pressure drag penalty, yielding a net drag increase as shown in Fig. 18 b).

Thus, the severity of the penalty is determined by the state of the aft cone boundary layer flow and its resistance to separation, as discussed in the previous section. The above are directly applicable to 2D lifting flows for which, heating of the boundary layer may lead, for example, to loss of lift as a result of severe viscous decambering [41] by prematurely triggering the onset of flow detachment and separation bubble formation along the aft region of the upper airfoil surface. Conversely, cooling a lifting surface may enhance the separation tolerance and thus lift characteristics of an airfoil at the expense of excess surface dissipation (opposite effect of heating), in agreement with the earlier experimental observations by Mabey [33].

B. Wall Heating Effect on Dissipation

The effect of boundary layer heating on the normalised viscous dissipation generation for varying wall temperature ratios T_w/T_∞ at fixed $FR = 8$, $Re_c = 1 \times 10^7$ and $M_\infty = 0.6$ is shown in Fig. 20 a). For fixed flow conditions, higher T_w/T_∞ values reduce the rate of change of viscous dissipation losses, $d\Phi/dx$, along airframe surfaces, thereby conveying larger portions of dissipation to the wake region. In terms of local scale quantities, this translates to an overall downward shift of the C_D^* distribution along the airframe's nose and midbody as shown in Fig. 20 b). The aft cone region is instead governed by an abrupt local rise in C_D^* (highlighted portion of the curves) with increasing T_w/T_∞ due to thermo-compressible flow reversal and separation being induced by the viscous deflection of the flow. Freestream compressibility further promotes separation, leading to a larger thermo-compressible separation zone of excessive dissipation, denoted by the (highlighted) C_D^* peak and by the steeper $d\Phi/dx$ slope of the pink dashed line for $T_w/T_\infty = 4.0$, $M_\infty = 0.8$ in Fig. 20 a) and b) respectively. This effect is illustrated in Fig. 21, for a heated $FR = 8$

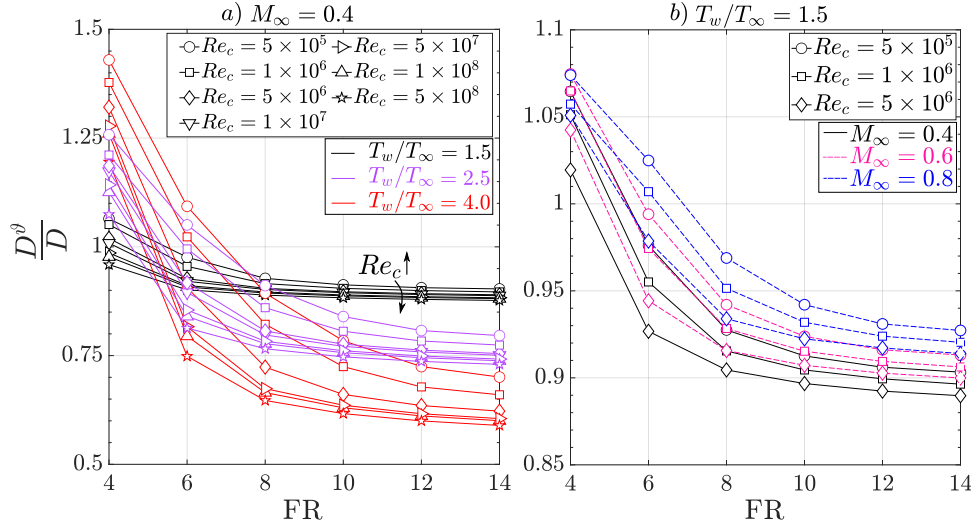


Fig. 18 Ratios of heated to adiabatic drag (D^θ/D) as a function of FR for: a) Fixed $M_\infty = 0.4$ and varying Re_c for various T_w/T_∞ ratios. b) Fixed $T_w/T_\infty = 1.5$ and varying Re_c and M_∞ numbers.

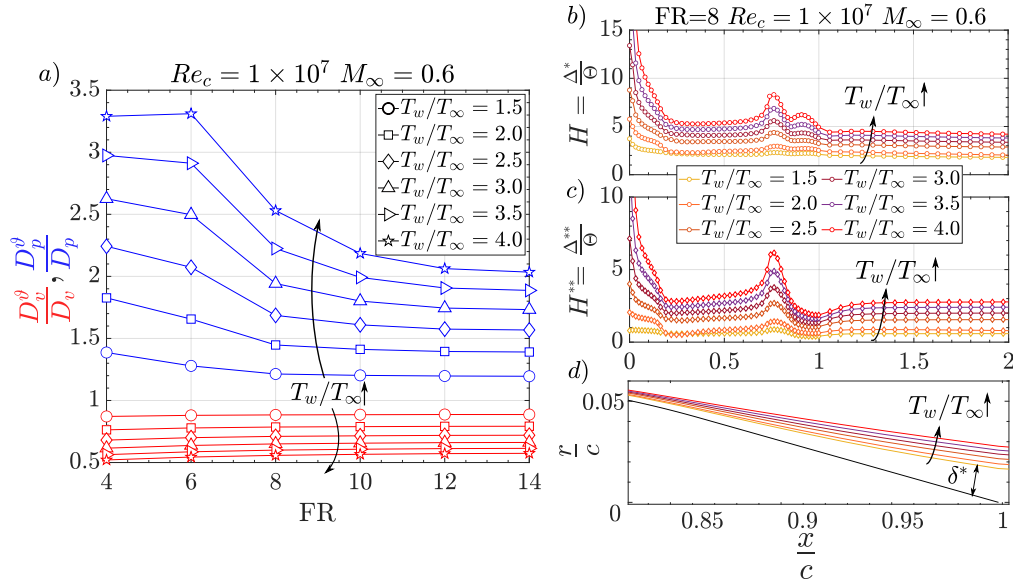


Fig. 19 For fixed conditions of $FR=8$, $Re_c = 1 \times 10^7$, $M_\infty = 0.6$ and varying T_w/T_∞ : a) Ratios of heated to adiabatic friction (D_v^θ/D_v) and pressure (D_p^θ/D_p) drag components against FR. b), c) H and H^{**} evolutions along the fuselage and the wake for varying T_w/T_∞ . d) Aft cone viscous displacement for varying T_w/T_∞ .

fuselage operating at different M_∞ numbers. For the low $M_\infty = 0.4$ regime of Fig. 21 a), the boundary layer is attached, whereas an increase in M_∞ in Fig. 21 b) triggers the formation of a separation bubble. The beneficial mitigation of airframe dissipation is also indicated by the reduction of the effective local momentum thickness-based Reynolds number, $Re_\theta = \rho_e V_e \theta / \mu_\infty$ seen by the flow as shown in Fig. 20 c). The gentler streamwise rate of change, dRe_θ/dx , for increasing T_w/T_∞ ratios underlines the slower growth of the boundary layer's dissipative characteristics in response to larger heat deposition rates.

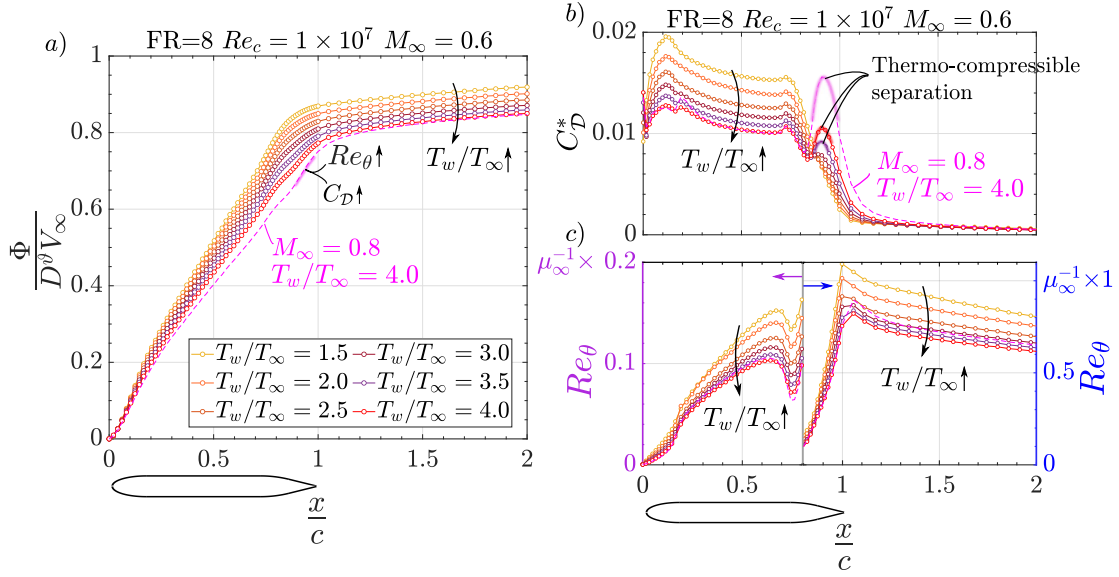


Fig. 20 Effect of varying the wall temperature on the evolution of a) Φ b) C_D^* c) Re_θ for fixed conditions of $FR=8$, $Re_c = 1 \times 10^7$ and $M_\infty = 0.6$.

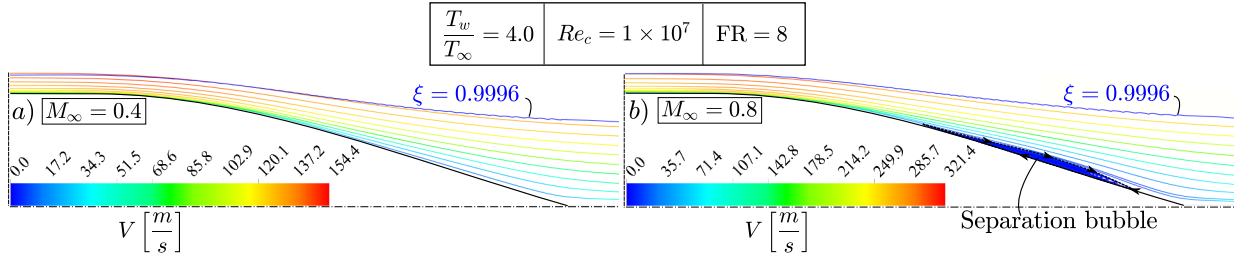


Fig. 21 Streamlines along the aft cone of a $FR = 8$ fuselage for fixed $Re_c = 1 \times 10^7$ and $T_w/T_\infty = 4.0$ and two Mach numbers: a) $M_\infty = 0.4$. b) $M_\infty = 0.8$.

A direct implication of the above is that wall heating controls the dissipation distribution along the viscous flow path by dampening the local evolution of the boundary layers' dissipative characteristics. Thus, surface dissipation reduction via wall heating (or other means of flow control) is a priori an aerodynamic benefit to the extent that the total mechanical losses (DV_∞) are reduced in the process, as per Fig. 18 a). Strategic dissipation control (for fixed drag power) is still advantageous as a means of passively controlling separation (via wall cooling) and potentially shock formation (via heating) in transonic flows. In the context of BLI, the loss distribution can be "fine-tuned" for larger wake dissipation losses meaning that a larger proportion of the drag power is then available for recuperation via a BLI propulsor.

C. Wall Heating Effect on Baroclinic Losses

The normalised residual baroclinic losses are shown in Fig. 22 a) as a function of FR for varying wall temperature ratios, T_w/T_∞ at fixed $Re_c = 1 \times 10^7$ and $M_\infty = 0.6$ conditions. For these subsonic conditions, the residual baroclinic losses stem from the heat deposition inside compressive and expansive flow regions that favour non-isentropic thermo-

compressible energy exchange. At fixed wall temperature and flow conditions, the baroclinic losses progressively subside for streamlined aerodynamic shapes due to the mitigation of local pressure changes along the nose and the aft cone. The abrupt drop for $FR = 4$ results from the disproportionate increase of the denominator ($D^\theta V_\infty$), as per Fig. 18 due to the extensive recirculation region that forms in the aft cone and near wake region. For typical adiabatic flows, the viscous self-heating of the flow manifests in the form of a viscous-interdependent thermal boundary layer field, however, for heated walls, the thermal boundary layer forms independently as a result of heat deposition in the fluid via heat conduction and convection. The implication is that entropic regions, enclosing baroclinic losses, may extend beyond the viscous boundary layer domain, because the spatial distribution of the heat-induced thermal boundary layer may propagate further from the viscous boundary layer. This is the case along the nose region, where the early viscous boundary layer growth lags behind the heat-induced thermal boundary layer field. For the flow conditions tested however, the localized Prandtl number distribution along the nose sustains values > 0.74 , mainly due to the sharp bulk flow acceleration that confines heat transfer close to the wall. To the extent that the Prandtl number is close to unity, as is the case for typical subsonic adiabatic turbulent flows ($PR \simeq 0.85$), the viscous and thermal boundary layer thickness are of the same order of magnitude [50] thereby, the baroclinic losses and thus the total mechanical profile losses are adequately enclosed within the viscous boundary layer domain. Based on the preceding assumption for adiabatic flow alongside other approximations, Sato [21] derived a baroclinicity-corrected dissipation integral for adiabatic flow, however its applicability is not universal and its validity further subject to TSL approximations.

Figure 22 b) shows transonic baroclinic losses due to shock formation as a function of T_w/T_∞ for varying Re_c number at fixed $FR = 4$ and $M_\infty = 0.8$. A typical transonic λ shock - separation bubble structure constituted by a leading compression wave system and a rear shock leg, as described by Babinsky and Harvey [51], of varying strength is formed for almost all the flow conditions downstream of the nose (see also Fig. 23 a) as an example). For the flow scenarios examined, the thermal boundary layer field that forms in response to heat deposition acts as a passive shock mitigation mechanism by reducing the local flow Mach number explicitly due to the $\sim T^{-0.5}$ scaling of the latter and implicitly due to the heating effect on the boundary layer shape (H increases). A possible explanation is that the combination of higher temperatures and H values shift the boundary layer's sonic line further away from the wall thereby restricting partially the shock wave from propagating inside the boundary layer, in agreement with previous results by Arntz and Atinault [52] on BLI powered adiabatic and heated airfoils. For the flow conditions examined, higher T_w/T_∞ values diminish the shock intensity, leading to a decrease in baroclinic shock losses for the upper $Re_c > 5 \times 10^7$ regime, as shown in Fig. 22 b). For the intermediate $5 \times 10^6 < Re_c < 1 \times 10^7$ regime there is a minimum loss value around $T_w/T_\infty = 3.5$ while the lower $Re_c \leq 1 \times 10^6$ regime is governed by excessive baroclinic losses despite the apparent shock mitigation benefit. This duality underlines the opposing dependence of shock and separation losses on heat transfer effects. Alongside shock reduction, wall heating severely penalizes the separation resistance of the flow leading to the formation of progressively larger shock-induced recirculation zones despite the shock strength mitigation, in agreement

with previous results by Back and Cuffel [53]. That is, wall heating as a means of compressibility loss reduction is bound to be futile if the baroclinic loss penalty due to heat-induced separation outweighs the shock mitigation benefit. This trade-off is illustrated in Fig. 23 in terms of two mirrored Mach contours along the $FR = 4$ body for $T_w/T_\infty = 1.5$ (upper half) and $T_w/T_\infty = 4.0$ (lower half) at fixed $Re_c = 1 \times 10^6$ and $M_\infty = 0.8$ conditions. For $T_w/T_\infty = 1.5$, the supersonic acceleration of $M < 1.3$ over the nose terminates with a λ -shock and a weak secondary shock followed by a subsequent shock-induced separation bubble just downstream. For $T_w/T_\infty = 4.0$, the excess heat supply suppresses compressibility ($M < 1.1$) and shock formation but triggers the onset of a larger nose separation bubble due to the associated separation intolerance penalty.

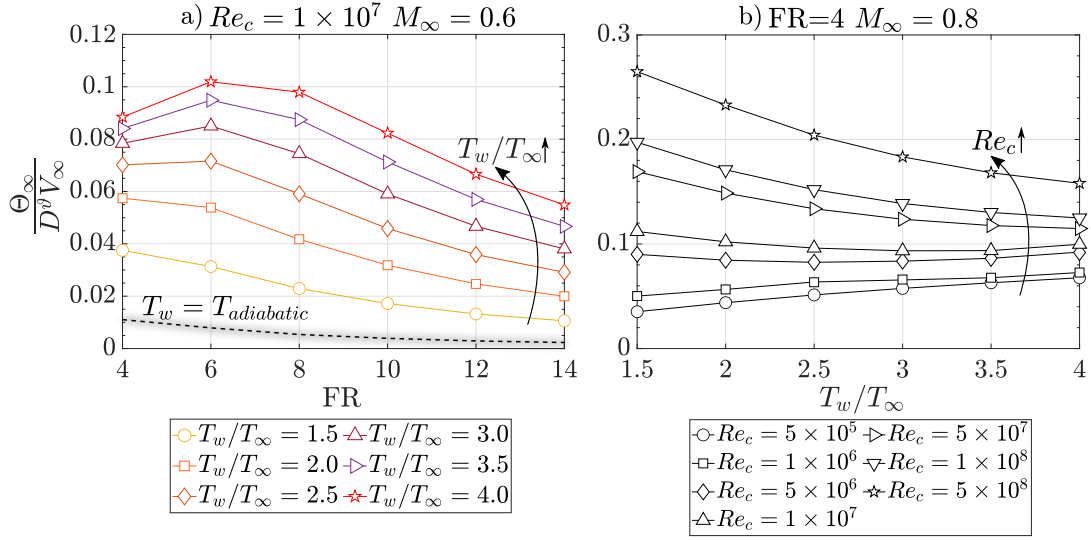


Fig. 22 Normalised residual baroclinic losses as a function of a) FR for varying T_w/T_∞ at fixed $Re_c = 1 \times 10^7$ and $M_\infty = 0.6$ conditions. b) T_w/T_∞ for varying Re_c number and fixed $FR=4$ and $M_\infty = 0.8$ with shock formation.

D. Wall Heating Effect on $\mathcal{PER}$

A further implication of wall heating, as a means of controlling the dissipation distribution, is that larger portions of the total dissipation can be strategically shifted to the wake region with a subsequent increase in $\mathcal{PER}_{TE}$ for given operating conditions and FR as shown in Fig. 24 a) and b). $\mathcal{PER}_{TE}$ increases monotonically and scales almost linearly with higher T_w/T_∞ ratios for streamlined aerodynamic bodies of $FR \geq 10$ with attached flow. For blunter bodies, the monotonicity is retained but the rate of increase $d\mathcal{PER}_{TE}/dT_w$ is partially offset by the locally disproportionate aft cone dissipation penalty induced by the progressively larger recirculation zones forming in response to heat deposition. To the extent that the trailing edge baroclinic losses are adequately predicted as $\mathcal{B}_{TE} \simeq \Theta_\infty$, the heat induced $\mathcal{PER}_{TE}$ values consistently exceed the corresponding adiabatic, underlying the possibility of favourable coupling between a heated aerodynamic body and a BLI propulsion system. Finally, the upward shift of the $\mathcal{PER}_{TE}$ trends with higher M_∞ is consistent with the adiabatic analysis and further signifies thermo-compressible interference mainly along the aft cone

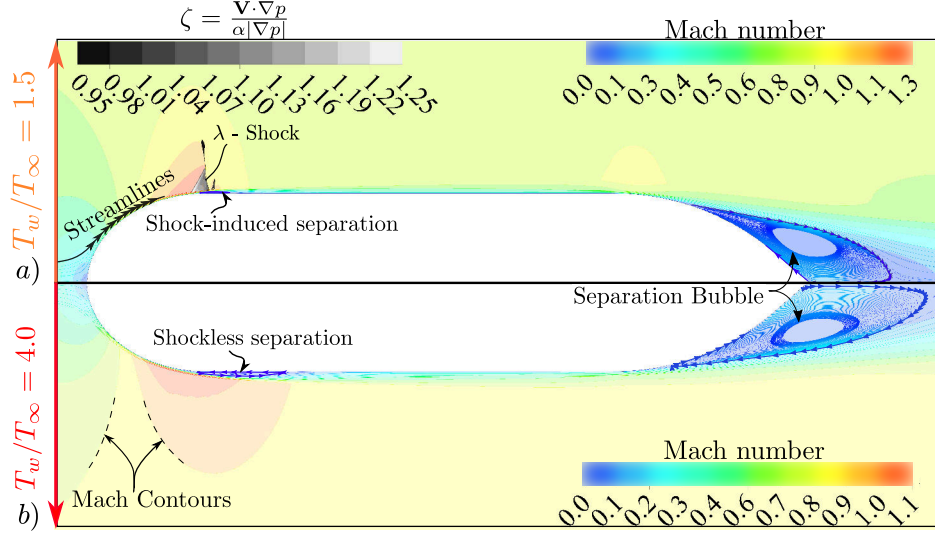


Fig. 23 Mach number contour and streamlines of an FR = 4 fuselage at $Re_c = 1 \times 10^6$ and $M_\infty = 0.8$ with two wall heating scenarios: (Upper half) $T_w/T_\infty = 1.5$ (Lower half) $T_w/T_\infty = 4.0$. Shockwave region is identified using the ζ shock identifier.

region via the compressible thickening of the EIB, yielding progressively larger $\mathcal{PER}_{TE}$ values for lower FR shapes at the expense of increased baroclinic losses.

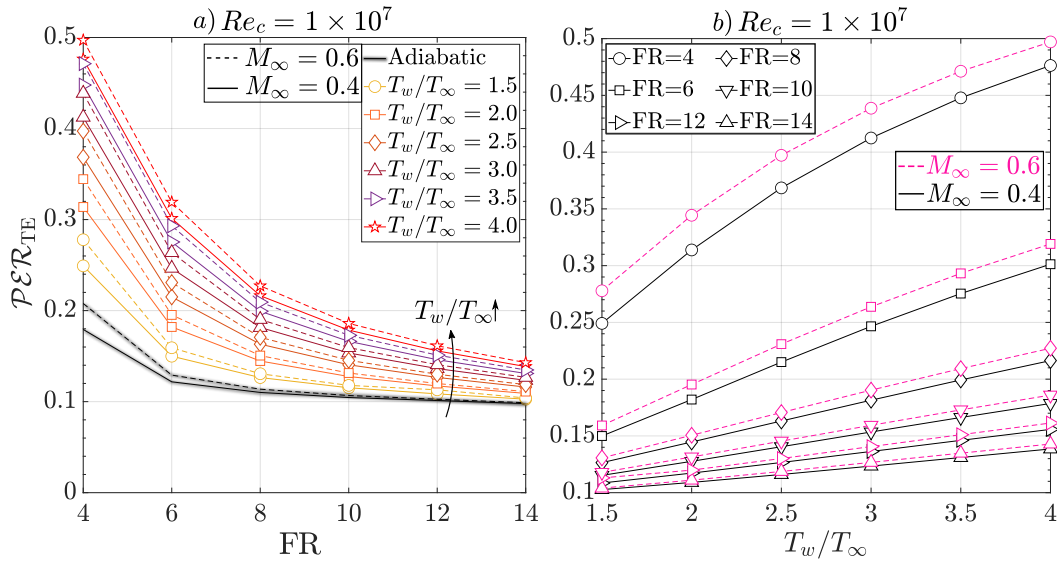


Fig. 24 For fixed $Re_c = 1 \times 10^7$: a) $\mathcal{PER}_{TE}$ plots as a function of FR for adiabatic and varying T_w/T_∞ ratios and two M_∞ numbers. b) Replotted $\mathcal{PER}_{TE}$ curves as a function of T_w/T_∞ for varying FR and two M_∞ numbers.

E. Heat Exergy Recovery

Figure 25 a) shows the effect of FR on the $\mathcal{PER}_\theta$ evolution (blue gradient lines) along the airframe and the local heat transfer coefficient, $C_h = \dot{q}/(T_w - T_\infty)$, distribution (purple gradient lines) at fixed conditions and $T_w/T_\infty = 3.0$.

For uniform wall temperature distribution, the heat transfer coefficient scales solely with the local heat flux, $\dot{q}$, so that variations along the fuselage translate directly to a streamwise rate of change of $d\mathcal{P}\mathcal{E}\mathcal{R}_\theta/dx$. This leads to faster $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$ accumulation along regions of local flow acceleration (nose) due to the increased heat flux rate that compensates for the associated improved convective heat transfer characteristics and expansive temperature reduction of the flow and vice versa. The abrupt C_h monotonicity change for $\text{FR} = 4, 6$ along the aft cone region indicates the sudden change of heat transfer characteristics due to flow separation followed by bubble formation. The overall disproportional reduction of the $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$ values for blunter bodies ($\text{FR} = 4, 6$) is only weakly related to the heat flux distribution, instead it accounts for the opposing effect of heating with FR on the drag power in the denominator as per Fig. 18 a). The dependence of $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$ on the Carnot efficiency implies that the quality of the heat-to-work conversion process is dictated by the temperature differential between the wall and the freestream flow. This manifests as a steep rise in $d\mathcal{P}\mathcal{E}\mathcal{R}_\theta/dx$ with increasing T_w/T_∞ due to the progressively larger wall heat deposition at higher heat (Carnot) efficiency as shown in Fig. 25 b) for fixed conditions and aerodynamic shape. The downward shift of the C_h curves for increasing T_w/T_∞ is explained with similar reasoning by re-expressing the C_h definition as a function of the Carnot efficiency, η_{Carnot} , and the temperature ratio $\kappa = T_w/T_\infty$ as:

$$C_h = \frac{\dot{q}}{T_\infty \cdot \kappa \cdot \eta_{\text{Carnot}}} \quad (25)$$

where $\kappa > 1$ (for heating) and $T_\infty = \text{const.}$, so that for higher wall temperature, the heat flux rate does not suffice to balance the disproportional increase of the denominator due to the correspondingly larger κ and η_{Carnot} values.

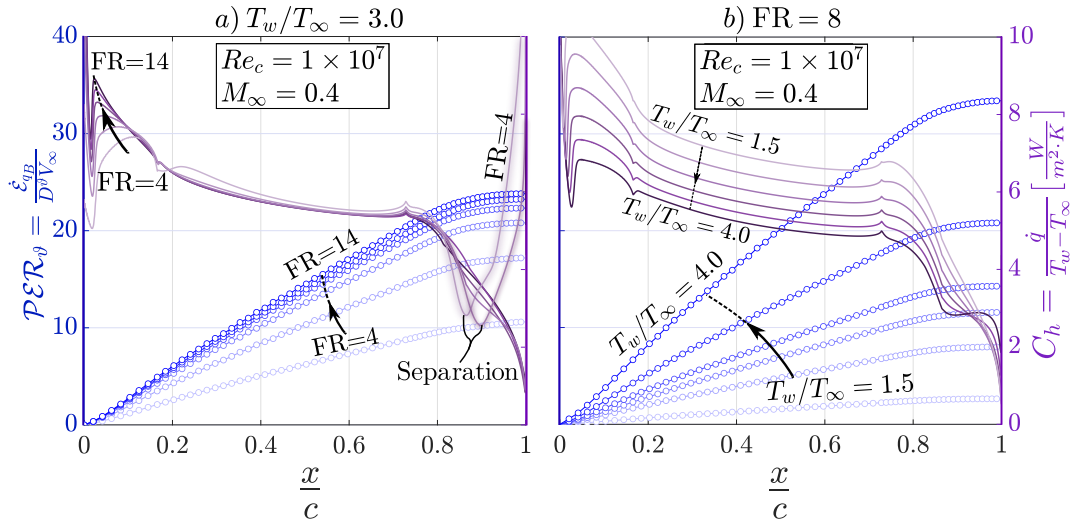


Fig. 25 Heat exergy recovery potential, $\mathcal{P}\mathcal{E}\mathcal{R}_\theta$, (blue gradient lines) and local heat transfer coefficient, C_h , (purple gradient lines) along the fuselage at fixed $Re_c = 1 \times 10^7$ and $M_\infty = 0.4$: a) For fixed wall temperature ratio of $T_w/T_\infty = 3.0$ and varying FR b) For fixed $\text{FR} = 8$ and varying wall temperature ratios T_w/T_∞ .

VI. Conclusions

A generalised, energy-based approach has been developed in this paper for analysing the aerodynamic loss and recovery characteristics of adiabatic and heated axisymmetric flows. The approach adopts a hybrid nearfield/farfield scheme that builds upon mechanical energy conservation, exergy, mechanical and geometric similarity principles as well as boundary layer analysis and enables:

- 1) Assessment of the effects of Re_c , M_∞ , FR and wall temperature on the mechanical loss generation and potential energy and exergy recovery mechanisms.
- 2) Identification of the physical origins of the baroclinic power, its approximation and inclusion within the mechanical energy equation.
- 3) The definition of a modified dissipation flux coefficient that quantifies the local dissipative contributions of shear layers and separation bubbles in axisymmetric flows.
- 4) Mapping the energy recovery characteristics of adiabatic and heated axisymmetric flows, relevant for BLI, in terms of the modified $\mathcal{PER}$ metric, as well as relevant similarity parameters.
- 5) The introduction, within the context of a mechanical energy analysis, of a heat exergy recovery metric ($\mathcal{PER}_\theta$) as well as an overall approach of quantifying the heat-to-work potential associated with non adiabatic walls.

Having applied the above scheme to an extensive test matrix, the nearfield drag power imparted from the fuselage to the flow has been correlated to the total volumetric mechanical loss generation inside the flowfield, defined as the sum of the total viscous dissipation and baroclinic power. The former addresses the irreversible conversion of viscous mechanical energy flow to heat while the latter accounts for thermo-compressible losses due to heat sources and self heating of the flow spanning across compressive/expansive regions of strong pressure gradients and shock waves. Section IV indicates that viscous dissipation is the primary loss generation mechanism, measuring $> 98\%$ of the total mechanical loss in agreement with previous observations by Sato [21], Hall et al. [32] and Uranga et al. [28], only to the extent that no shockwaves or strong heat transfer effects are present in the flowfield. For transonic flows, the underlying shock loss mechanism was observed to manifest itself as the baroclinic power which is triggered by the local shock dissipation (and thus heating) acting on a thin shock region of excessive compression. For the examined transonic flow scenarios, the shock losses measured up to 28% of the total drag power, with higher Reynolds numbers favouring the onset of stronger shock formation alongside larger boundary layer dissipation (see discussions in Sec. IV.B.2). The implication is that the effect of mismatching the full-scale Re_c in wind-tunnels compromises both the dissipative and baroclinic characteristics of the flow. For subsonic fuselage designs, mechanical similarity and dissipation losses are driven by Re_c number alone, whereas for transonic aircrafts with shock formation, a significant mismatch in Re_c of a wind-tunnel model may yield unacceptably inaccurate shock strength predictions despite the apparent M_∞ matching. For aerodynamic bodies with flow reversal and separation bubbles at fixed Re_c number, the severity of the separation bubble characteristics (excess dissipation and baroclinic losses) are controlled by M_∞ due to the affiliated compressibility-induced viscous

displacement thickening. The $\mathcal{PER}$ analysis of Sec. IV.C indicates that the airframe dissipation measures between 80 – 90% of the total dissipation based on the bluntness of the body, meaning that for fixed flow conditions, lower FR bodies shift larger portions of dissipation in the wake, favouring larger $\mathcal{PER}$ values and thereby potential BLI benefits.

Heat deposition in the boundary layer has been found to yield circumstantially improved aerodynamic characteristics by reducing the total mechanical losses and by acting as a passive flow control mechanism that decreases the local dissipative characteristics of the boundary layer and the local Mach number across the thermal boundary layer profile (see discussion in Sec. V.B and Sec. V.C). The net effect of boundary layer heating is thus airframe dissipation loss mitigation and possibly shock wave suppression at the expense of excess heat-induced baroclinic losses and reduced separation resistance of the boundary layer. In the scenario where the favourable heat transfer aerodynamic mechanisms outweigh the penalties, boundary layer heating yields a net aerodynamic benefit that manifests as a reduction in the total drag power and as an increase in $\mathcal{PER}$ relative to the adiabatic, signifying the possibility of favourable aerodynamic synergies between wall heating and BLI propulsion. Another interesting implication of non-adiabatic walls is that mismatches in similarity parameters may be partially bridged via heating or cooling of wall surfaces. For example, cooling the wall surfaces of a wind-tunnel model may replicate dissipative characteristics of higher Re_c flows and vice versa. Additionally, cooling may be employed along adverse pressure gradient zones (aft cone region) to suppress undesirable separation effects when the viscous displacement effect is magnified by the lower experimental Re_c number flow. Further applications may extend to lifting 2D flows whereby cooling may enhance the lift characteristics by delaying the onset of severe viscous decambering and thus stall due to separation at the expense of excess dissipation, while localised wall heating may aid to shock suppression at the cost of moderate baroclinic loss penalty. Finally, heat recuperation aspects are explicitly addressed by the potential for heat exergy recovery $\mathcal{PER}_\theta$ definition to conceptually assess the feasibility of synergistic heat extraction via strategic thermal management of on-board heat sources such as batteries, fuel cells and electrical systems.

A. Numerical Estimation of Baroclinic Power

The baroclinic power measures the local deviation of the compressive/expansive fluid response relative to the equivalent isentropic flow (EIF) [21] as a result of viscous self-heating inside shear layers and conductive or volumetric heat sources in the flowfield. For transonic flows, shock formation favours excessive baroclinic power contributions due to wave dissipation (and thus heat generation) in response to the abrupt change of the flow properties across it. The simultaneous presence of strong isentropic expansion and compression contributions that only self-cancel after the full expansion of the flow downstream do not allow for local quantification of baroclinic contributions. A numerically advantageous and exact way to calculate the total baroclinic power is to carry out the integration of Θ over a

fully-expanded domain, denoted with (Θ_∞) , to filter out the isentropic portion, Θ_r :

$$\mathcal{B} = \mathcal{B}_{\text{BL+wake}} + \mathcal{B}_{\text{shock}} + \mathcal{B}_{\mathcal{Q}} \cong \Theta_\infty \quad (\text{A.1})$$

where it is assumed that the baroclinic residual is comprised of indirect contributions from boundary and free shear layers $\mathcal{B}_{\text{BL+wake}}$, shock waves $\mathcal{B}_{\text{shock}}$ and heat sources (conductive, volumetric) $\mathcal{B}_{\mathcal{Q}}$. To the extent that the baroclinic power does not vary substantially along the semi-expanded near wake region ($\mathcal{B}_{\text{wake}} \approx 0$) as demonstrated in [39, 40] the validity and applicability of the right-hand side approximation of Eq. (A.1) may effectively be extended to the entire wake region.

B. Numerical Uncertainty

The numerical accuracy of the adiabatic and heated fuselage CFD solutions is reported in terms of a power imbalance between "nearfield power supply" and "power deposition" components. With reference to Eq. (11), the relative power imbalance error ϵ is obtained as follows:

$$\epsilon = 1 - \frac{\Phi^*}{DV_\infty} = 1 - \frac{\Phi + \Theta_\infty}{DV_\infty} \quad (\text{B.1})$$

Equation (B.1) compares the power difference between the nearfield drag power (power input) to its equivalent mechanical loss manifestation (Φ^*) in the flowfield.

Sample plots of ϵ for a wide range of operating conditions are provided in Figs. B.1 and B.2 for adiabatic and heated scenarios, respectively. A very good agreement for both adiabatic and heated flows is observed independent of operating conditions, with power imbalances typically less than 1.5%. The adiabatic plots of Fig. B.1 exhibit a monotonic behaviour towards higher FR and Re_c values for all three M_∞ numbers. A possible reason is that a combination of higher FR and Re_c conditions shift a larger proportion of the mechanical losses along the very highly-resolved airframe surfaces. As a result, a smaller fraction of the total mechanical loss occurs within the progressively coarser farfield region downstream of the trailing edge. Also notice that the presence of the shockwave for the $M_\infty = 0.8$ curve set (see Fig. B.1 c)) does not alter the overall characteristics of the curves, as the shock region lies within the very well-resolved near-wall regions of the nose end.

Consistently low ϵ values are also reported for the heated fuselages. However, the aforementioned monotonic behaviour is present typically for $\text{FR} \geq 10$ depending on the Re_c examined. This mixed behaviour is possibly related to the presence of stronger separation regions, the severity of which is also related to the heat supplied to the flow. This seems to justify the wider spreading among curves of different T_w/T_∞ of lower Re_c number flows in Fig. B.2 a) - c) (first column), as these flows are less tolerant to separation. Conversely, higher Re_c flow regimes more effectively suppress

extensive boundary layer separation and as such, the monotonic behaviour, observed in the adiabatic, is gradually restored for $FR \geq 8$, as shown in Fig. B.2 a) - c) (mid and last column).

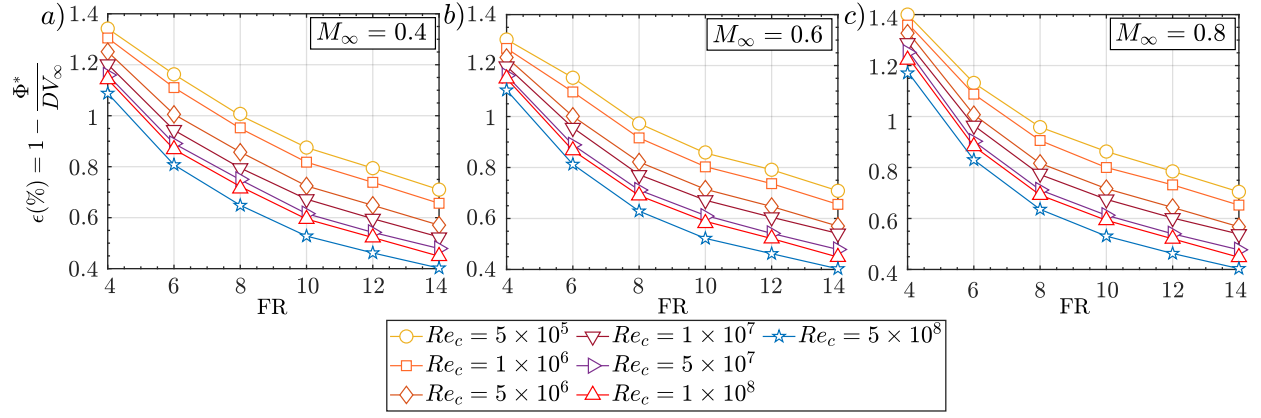


Fig. B.1 Power imbalance error (ϵ) plots of adiabatic unpowered fuselages as a function of FR, varying Re_c and three Mach numbers: a) $M_\infty = 0.4$. b) $M_\infty = 0.6$. c) $M_\infty = 0.8$.

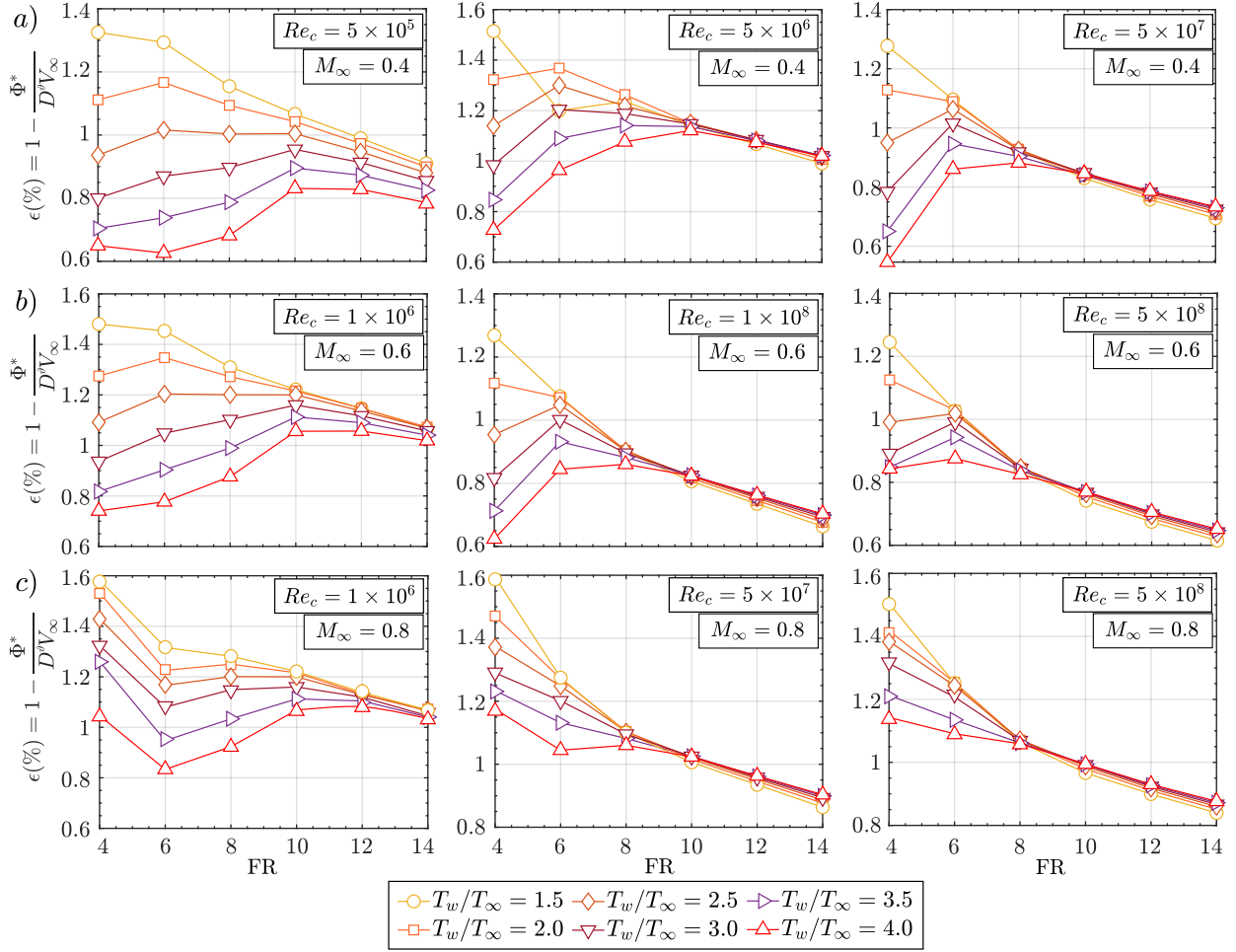


Fig. B.2 Power imbalance error (ϵ) plots of heated unpowered fuselages as a function of FR for varying T_w/T_∞ ratios and operating conditions: a) (Row) Fixed $M_\infty = 0.4$. b) $M_\infty = 0.6$. c) $M_\infty = 0.8$.

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