

UNDERSTANDING HYPERSONIC BOUNDARY LAYER LAMINAR TO TURBULENT TRANSITION IN WIND TUNNEL ENVIRONMENT

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Abstract

Understanding laminar to turbulent transition remains a critical challenge for the design of high-speed aerospace vehicles. Ground facilities, while essential, often fail to replicate flight conditions, particularly in terms of enthalpy and acoustic receptivity, leading to discrepancies in transition prediction. This study reviews the tools and methodologies currently used to investigate transition, from flight tests to ground-based wind tunnel experiments, and advanced numerical simulations such as Direct Numerical Simulation (DNS) and stability analysis. The impact of free-stream disturbances, noise, and facility-specific effects on transition is highlighted, alongside ongoing efforts to bridge the gap between ground and flight data. Perspectives for future work include the development of "quiet" wind tunnels, improved disturbance quantification, and the integration of experimental and numerical approaches to enhance transition understanding.

Keywords: Transition, Hypersonic, Wind-tunnel, numerical simulation, stability

1. Introduction

Understanding, modelling, and controlling laminar-turbulent transition remain one of the challenges facing engineers in the design of aerospace vehicles. This is true both for aviation, where transition is linked with a change in viscous drag, and thus maintaining laminarity could lead to increased efficiency, the defence sector, notably for high-speed vehicle design. In the hypersonic regime, for generic (*e.g.* attached boundary layer) cases, the heat flux overshoots drastically in the transition region, with a transition onset highly dependent on the flight conditions (Mach number, Altitude, atmospheric turbulence, particles), making the correct design of the thermal protection as well as the proper prediction of thrust-to-drag ratio tedious. Adding to that complexity, shocks appear in the vicinity of any geometrical discontinuities present on vehicles, such as control surfaces or inlet compression ramps. These shocks interact with the boundary layer and may cause the apparition of a recirculation downstream from the interaction, leading to drastic changes in both heat flux and pressure distribution. This separation could be strongly impacted by transition, leading to unsteady pressure loads and heat flux distribution that may be strong enough to cause mission failure (loss of integrity of the thermal protection system, loss of manoeuvrability, or efficiency of the propulsion system).

Even if flight tests have become more affordable, engineers still strongly rely on ground tests in the design loops of the vehicles. However, ground facilities lack the ability to duplicate the flight environment, as they often fail to reproduce the enthalpy and the receptivity that could be encountered in flight. In this context, the challenge becomes not only understanding and modeling the transition process in ground facilities but also being able to transpose it to flight conditions.

This study aims to review the current tools and methodologies used at ONERA for investigating hypersonic boundary layer transition, with a focus on bridging the gap between ground-based experiments and flight conditions.

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2. Tools to study transition

2.1 Flight tests

Given the difficulty in accurately reproducing the flight conditions on the ground, access to true flight transition data requires conducting flight tests. However, most of the flight tests conducted today are linked with prototypes or system validation, for which the inherent complexity prohibits the addition of dedicated sensors to study the physics of laminar turbulent transition.

One solution to gather data to study transition is to conduct dedicated highly instrumented scientific flight tests [1, 2, 3, 4]. However, boundary layer instabilities leading to transition are notably hard to measure, for instance, the second Mack mode [5], which is often cited as the dominant instability at Mach number greater than 4.5 will manifest itself as a very high frequency wall pressure fluctuation, in the 100 KHz to MHz range, requiring specific high-bandwidth pressure sensors [6] and acquisition systems to handle the very large amount of data created. Additionally, the brief duration of hypersonic flight tests, often just a few seconds near the end of the trajectory, poses a significant challenge. Given the limited available telemetry bandwidth, transmitting the full raw data to ground stations in real time is impractical, and payload recovery is often too complex or expensive. Consequently, onboard processing is usually required to reduce the data volume, extracting only the most critical features (eg most energetic fourrier transform coefficient) for transmission. As such, flight tests require the development of dedicated instrumentation and onboard post processing.

Once a payload is designed to capture data about boundary layer transition, one also needs to obtain in-situ measurements of the flight environment. This includes not only the mean atmospheric conditions along the trajectory (pressure, temperature, mean wind velocity) but also the unsteady, high-frequency fluctuations that can influence transition. For instance, atmospheric turbulence and aerosol distributions must be characterized, as they can significantly alter the receptivity of the boundary layer and trigger transition mechanisms.

This leads to complex designs and expensive flight test campaigns, even for relatively simple cases. One example of such flight tests is the BOLT programme [3], which collected data about both hypersonic transition [2] and turbulence, as well as in situ measurements of the flight conditions [7] by deploying balloon-borne sensors and lidar systems to capture real-time data on atmospheric turbulence and particulate concentrations. These in-situ observations provide the context necessary to interpret transition data accurately and ensure that ground-based experiments and numerical simulations can replicate flight-like conditions.

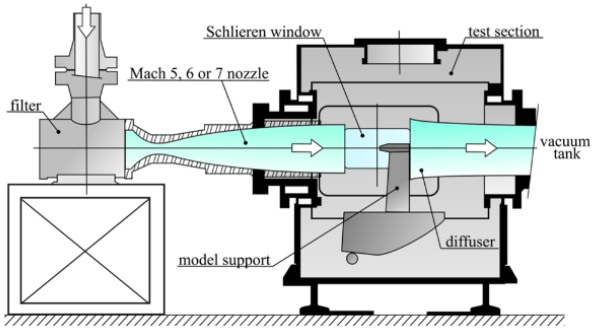


Figure 1 – Schematic of the R2Ch hypersonic facility at ONERA Meudon.

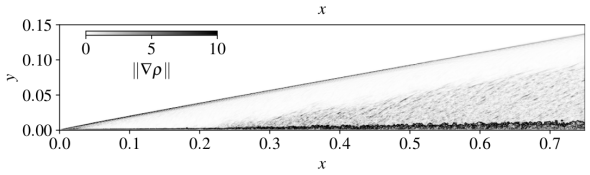


Figure 2 – Illustration of the acoustic emission of Mach 6 transitional and turbulent boundary layer computed using direct numerical simulation [8].

2.2 Wind tunnel testing

Given the complexity of flight tests, we still rely heavily on ground facilities to gather experimental data. These facilities allow for controlled experiments with repeatable conditions. While measuring transitional instabilities still remains a challenge, many metrologies can be deployed during ground test campaigns. For instance, infra-red imaging or Temperature Sensitive paint provides high-resolution heat-flux maps to study the impact of transition (*i.e.* a sharp increase in heat flux). On the other hand, high-frequency pressure transducers or off wall high frequency measurement such as FLDI, are used to investigate the linear and non-linear mechanisms driving transition, by tracking the spatial growth of instabilities and their interactions. Some examples of typical measurements conducted in conventional facilities include wall heat flux showing the impact of laminar-turbulent transition (shown in Figure 3), as well as wall pressure spectra (shown in Figure 4) from a sensor on the body, highlighting the modal instabilities at the origin of transition. In these spectra, the large peak around 135 kHz at relatively low Reynolds numbers is associated with the second Mack mode, a dominant instability in hypersonic boundary layers. As the Reynolds number increases, this peak increases in amplitude (sign of further amplification of the mode) and shifts toward higher frequencies, consistent with the thinning of the boundary layer. At higher Reynolds numbers, the non-linear breakdown of these instabilities leads to a complete filling of the spectrum, marking the transition to turbulence. This non-linear phase is also correlated with a sharp increase in wall heat flux, as observed in infrared thermography measurements (Figure 3 and 4).

However, one key limitation of most ground facilities is their high level of free-stream acoustic noise. In hypersonic regimes, turbulent boundary layers emit intense acoustic waves in the form of shocklets, as illustrated by the DNS of a transitional flat plate presented in Figure 2. For conventional wind tunnels, such as the R2Ch facility shown in Figure 1, the boundary layer developing along the nozzle walls generates acoustic perturbations that propagate into the test section. These perturbations contaminate the free-stream with fluctuations typically two orders of magnitude higher than in flight, altering the receptivity of the model's boundary layer. One important point to mention here is that the high level of acoustic noise in conventional tunnels mostly leads to earlier (*i.e.*, at lower Reynolds numbers) transition but does not drastically change the phenomenology of transition, for example, it is not strong enough to bypass the modal phase of transition. In very specific cases, however, the noise may lead to a completely different transition scenario compared to flight conditions as the wind tunnel noise may bias the seeding towards different instabilities at different frequency.

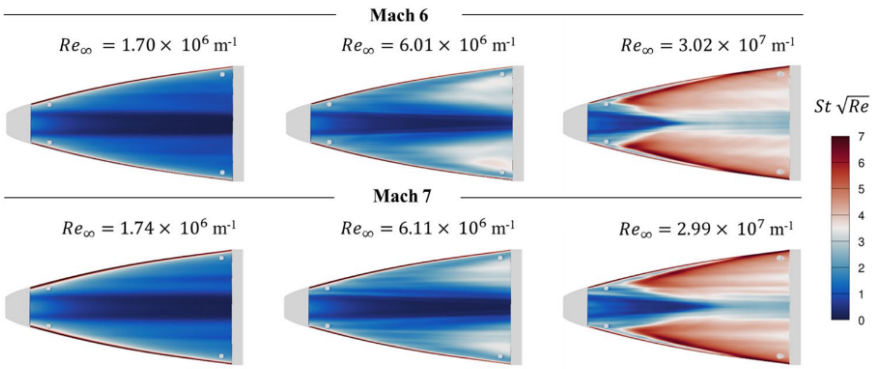
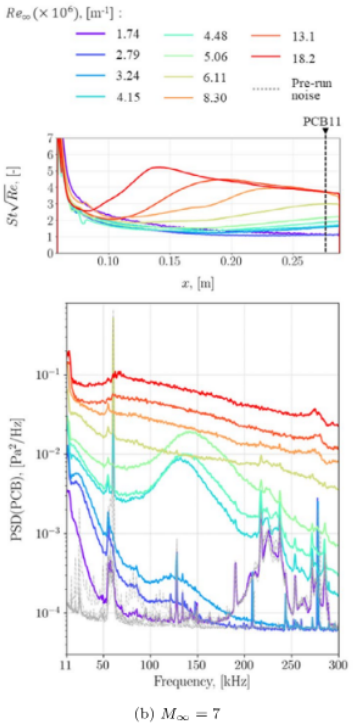


Figure 3 – Wall heat flux maps for the BOLT forebody measured in the R2Ch facility using IR thermography at different Reynolds and Mach number (fully laminar on the left, early transition on the right) [9].

One solution to that problem is to run so called "quiet" wind tunnel [10]. These facilities are designed to maintain a laminar boundary layer along the nozzle walls, thereby eliminating the acoustic emissions that contaminate the free-stream in conventional tunnels. As a result, they provide test conditions that are closer to flight, particularly in terms of receptivity and transition onset. In such environments, the transition N-factors are significantly closer to those observed in flight, offering a more accurate representation of the natural transition process. This is particularly interesting for



(b) $M_\infty = 7$

Figure 4 – Slice of the previously presented wall-heat-flux (top) and wall pressure power spectral density (bottom) on the BOLT geometry in R2Ch for different Reynolds number [9], the pressure spectra show the amplification of the second Mack mode in the 100 to 200kHz range.

highly 3D cases and high-Reynolds-number laminar flows, where complex flowfields can develop in the absence of external disturbances. However, quiet tunnels are often limited in Reynolds and Mach number ranges, which restricts their ability to fully replicate the high-enthalpy and high-speed conditions encountered in actual flight. Despite these limitations, they offer a valuable benchmark for studying transition mechanisms in a low-noise environment, free from the acoustic disturbances that can alter the transition process.

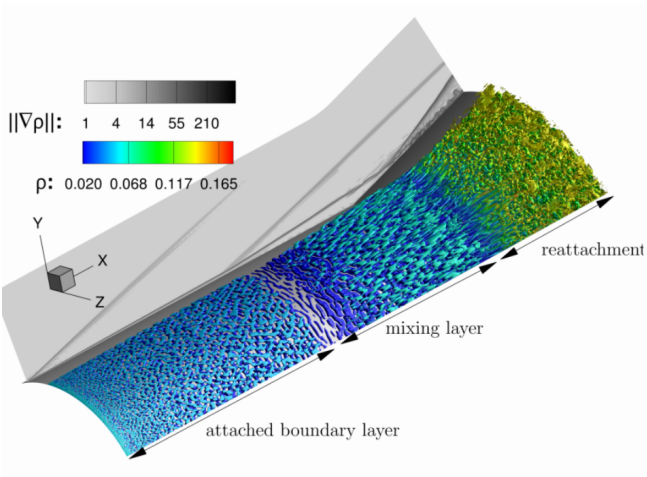


Figure 5 – Illustration of a direct numerical simulation conducted on a hollow cylinder flare at Mach 5 for a transitional Reynolds number [11]

2.3 Numerical tools

Post-processing experimental data can provide valuable insights into the nature of the instabilities driving transition and their breakdown into turbulence. However, experiments alone offer only sparse

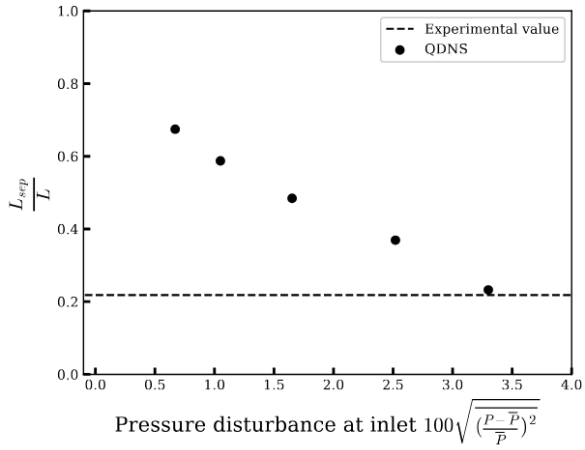


Figure 6 – Size of the mixing layer region length of figure 5 as a function of the level of perturbation injected at the inlet of the domain [11].

and localized measurements (at best, one gets surfacic data), limiting their ability to capture the full complexity of the transition process. To overcome this, one needs to complement them with numerical simulations.

One way forward is to conduct Direct Numerical Simulation, which, if it matches with the experiments, will provide access to the complete data about flowfield and thus, once post-processed correctly, a fine understanding of the transition process. An illustration of such a computation is presented in figure 5, which presents a transitional shock-boundary layer interaction caused by a compression ramp (which can be seen as a canonical control surface) at Mach 5. Apart from well known numerical challenges linked with numerical resolution and stability, the main difficulty linked with those computations is being able to reproduce the noise environment of the experiment to achieve the same transition scenario. The spectral content and amplitude of the injected disturbances must be carefully tuned to match the free-stream conditions of the wind tunnel. The impact of noise is illustrated in figure 6, which presents the size of the separated region of the computation presented in figure 5 as a function of the level of pressure disturbances injected at the inlet of the simulation. It shows that the separated region can go from almost the full computation domain to close to no separation simply by varying the noise intensity by a factor of less than ten (the ratio between flight and wind tunnel acoustic noise is close to 10) and that for a certain level of noise, one can match the experimental results. In terms of vehicle design, this result means that the efficiency of the control surface may be modulated not only by the flight Mach number or altitude but also by the free-stream perturbation environment.

However, matching the amplitude alone (or the mean flow) is not sufficient: the spectral distribution of the noise plays a critical role in accurately reproducing the transition process. For instance, as demonstrated in [13], experiments conducted in LT5 and R2Ch at similar Mach and Reynolds numbers exhibited significant differences in transition behaviour, experiments in LT5 displaying later transition at similar Reynolds number. To better understand those discrepancies, fine measurements of the free-stream acoustic fluctuations where conducted free-stream using the same acoustic probe (presented in figure 7). Despite comparable overall noise amplitudes, the R2Ch facility displayed a noise spectrum centred at much higher frequencies than LT5 (see figure 8). This difference was linked to the thinner nozzle wall boundary layer at the acoustic emission point in R2Ch, due to the significantly shorter nozzle length compared to LT5. This spectral difference led to a stronger seeding of first-mode and shear-layer instabilities in R2Ch, resulting in faster transition and a smaller recirculation region. These results highlight that even with identical noise amplitudes, the frequency content of the disturbances can drastically alter the receptivity of the boundary layer and, consequently, the transition scenario. This highlights the necessity of injecting realistic noise amplitude and spectral content in DNS if one wants to use it as a way to access the full flowfield information of a wind tunnel experiment. It is worth noting that this is also true for flight experiments, for which the free-stream fluctuation environment is even harder to measure.

Even with a DNS, understanding the physical origin of the instabilities is not trivial. In order to gain

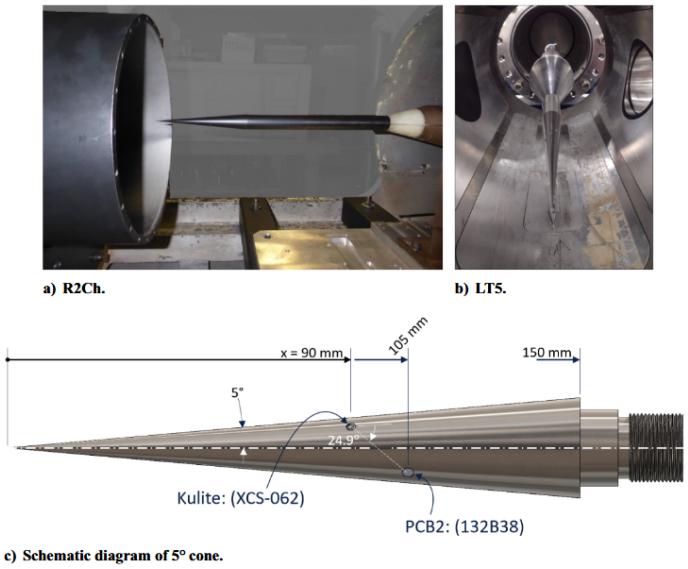


Figure 7 – Freestream acoustic probe developed in [12] and used to compare noise environment in two facilities in [13].

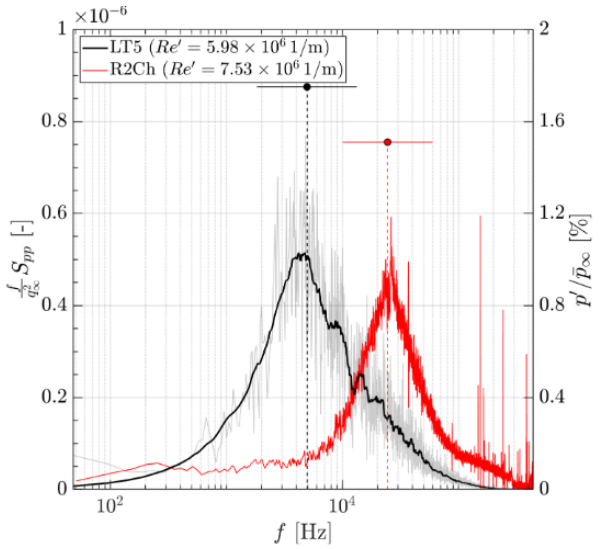
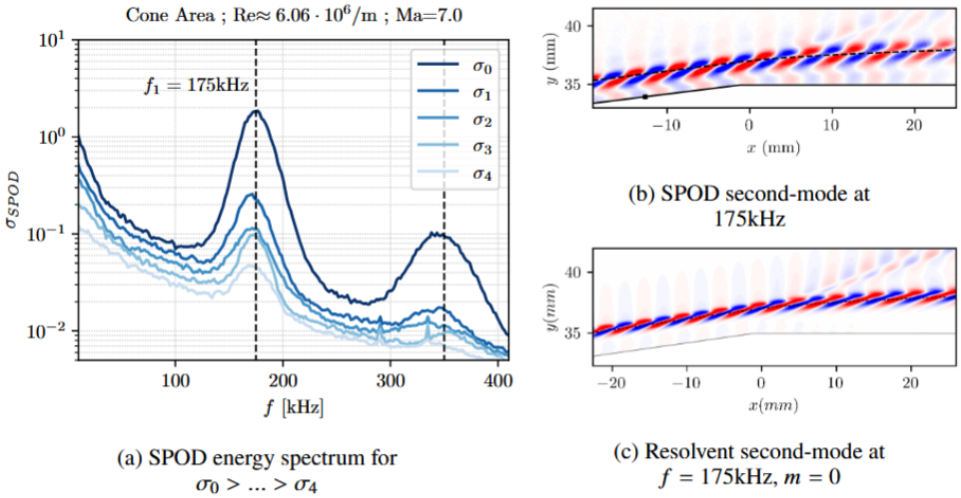


Figure 8 – Measured free-stream acoustic sprectra from LT5 and R2Ch, reprinted from [14]

more insight, stability analyses are often conducted, they allow to study the behaviour of the linearised Navier-Stokes operator and thus understand the linear behaviour of perturbations, which dominate the physics for small amplitudes and are therefore at the origin of the instabilities. Combining DNS and stability analysis allows to understand what comes from linear mechanisms and what is created by the nonlinearities, providing valuable physical insight into the transition process observed experimentally.

One interesting point is that a direct comparison between stability analysis and other approaches is possible. For instance, linear predictions can be validated against DNS results, as demonstrated in [11], where the growth of instabilities in a hypersonic compression ramp flow was shown to align with the most amplified modes from stability analysis. Similarly, experimental data can be post-processed to extract coherent structures and compare them with theoretical predictions. In [15], Spectral Proper Orthogonal Decomposition (SPOD) was used to isolate dominant flow structures from high-speed schlieren images. The SPOD modes, representing the most energetic fluctuations at specific frequencies, were found to match the optimal responses from Resolvent analysis, confirming the linear origin of the observed instabilities, such as the second Mack mode or trapped acoustic



(a) SPOD energy spectrum for $\sigma_0 > ... > \sigma_4$

(b) SPOD second-mode at 175kHz

(c) Resolvent second-mode at $f = 175$kHz, $m = 0$

Figure 9 – Schlieren imaging SPOD spectrum (a) and leading mode (b) compared to stability analysis results (c) for Mach 7 flow over a 5 degree cone followed by a cylinder [15]

waves in separated regions.

Beyond that, combining stability analysis with DNS or experiments enables a clear separation of linear and non-linear effects, which is critical for understanding the entire transition scenario. In [11], this integrated approach allowed to understand the transition scenario for a hypersonic compression ramp flows. The initial amplification of oblique modes (first Mack mode), is perfectly captured by linear stability theory, as these perturbations grow exponentially in the attached boundary layer due to the non-normal nature of the Navier-Stokes operator. However, as the flow evolves, the interaction of these oblique modes through triadic coupling generates streamwise streaks, a purely non-linear phenomenon that cannot be predicted by linear theory alone. These streaks arise from the non-linear saturation of the oblique waves and their subsequent breakdown. Once formed, the streaks themselves can be amplified linearly by mechanisms such as the lift-up effect or Görtler instability in curved regions. This secondary linear growth, though initiated by non-linear interactions, is still governed by the linearised dynamics of the base flow. This decomposition allows for a comprehensive understanding of the transition process : linear mechanisms select and amplify free-stream disturbances, those linear instabilities then undergo non-linear interactions to create new structures, which then may be further linearly amplified until final breakdown.

Another example of use of global stability to better understand the transition process is the work of [15] which combined Resolvent analysis and experimental measurements on the CCF12 geometry. The Resolvent operator perfectly captured the linear mechanism of the second Mack mode on the cone, with optimal responses matching the SPOD modes extracted from high-speed schlieren data. This agreement confirmed that the initial amplification of the second mode is purely linear and dominated by the convective instability of the attached boundary layer. However, Bispectral Mode Decomposition (BMD) of the high speed schlieren images identified triadic interactions between the second Mack mode, itself and its super-harmonics (see figure 10 and 11). These interactions, invisible to linear analysis, explain the progressive energy cascade from high-frequency second-mode to its superhamonics, ultimately driving the final breakdown to turbulence.

2.4 Comment on similarity

Traditionally, hypersonic wind tunnel experiments have relied on Mach and Reynolds number similarity to replicate flight conditions. While these parameters are critical for matching the overall flow topology and dynamic pressure, they often fail to capture the full physics of transition. As highlighted throughout this paper, free-stream disturbances play a major role in altering the transition scenario, but they are not the only missing parameter. Another key similarity criterion is the wall temperature, which strongly influences not only the stability of hypersonic boundary layers but also their development and thickness. As demonstrated in [16], a lower wall temperature can destabilize the second

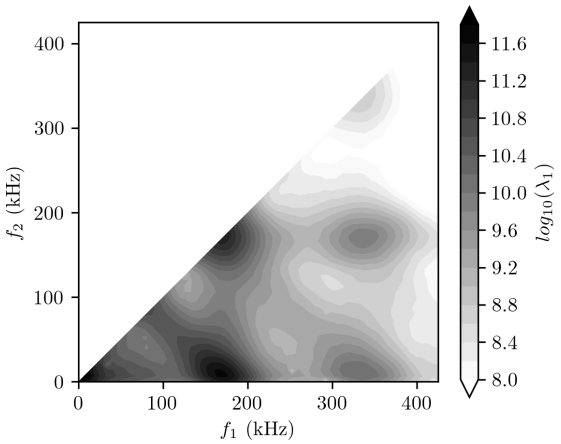


Figure 10 – Interaction map from the BMD decomposition of the high-speed Schlieren imaging of the flow around a 5° cone at Mach 7 in the R2Ch facility, showing the second mode self-interaction to create superharmonics, reprinted from [15]

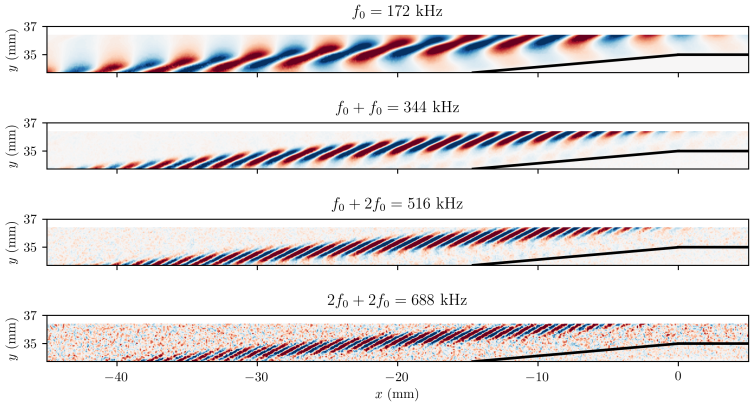


Figure 11 – Second mode fundamental wave and superharmonics extracted from the BMD of high speed Schlieren imaging, reprinted from [15]

Mack mode while suppressing the first mode in attached boundary layers. Moreover, their study showed that flight-like instability mechanisms can be reproduced in conventional wind tunnels by using strongly cooled models, thus matching the wall temperature effects observed in flight (see figure 12). This shift in instability dominance may drastically alter the transition scenario. Thus, achieving full similarity, in Mach, Reynolds, and wall temperature, may be necessary to accurately predict transition onset and mechanisms during ground tests. However, even with these adjustments, viscosity jumps across shock waves will still differ between wind tunnels and flight conditions, as shown in [16]. This discrepancy arises because the temperature rise across the shock—and thus the viscosity change—depends on the stagnation temperature and gas properties (e.g., Sutherland's law). In wind tunnels, the lower stagnation temperatures lead to smaller viscosity jumps compared to flight, altering the boundary-layer development and shock-boundary-layer interactions, which in turn affects the receptivity and stability characteristics. The situation becomes even more complex at higher enthalpies, where real-gas effects and chemical reactions further deviate from ideal-gas assumptions, making ground-based replication of flight conditions increasingly difficult. In such cases, direct duplication of the entire flight environment may be the only viable option.

3. Pespectives and ongoing work

In this context, future work should go in the direction of bridging the gap between both numerical and experimental simulations and ground and flight tests. This includes understanding and quantifying

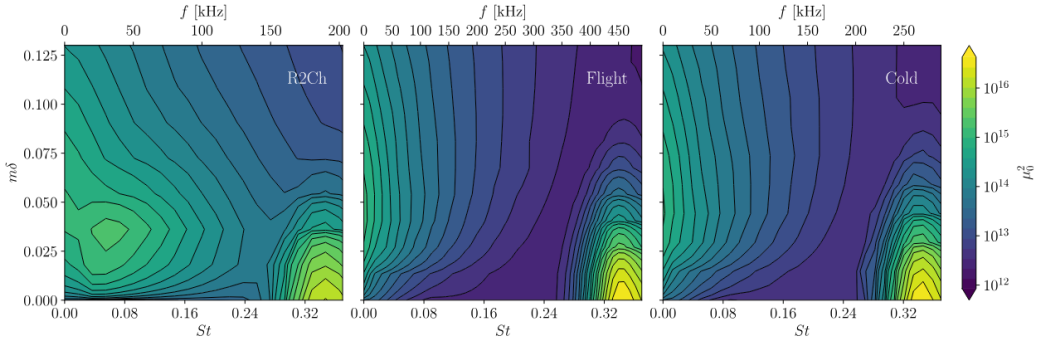


Figure 12 – Amplification maps in the frequency-azimuthal wavenumber domain showing the different between the stability of a 5 degree cone boundary layer at Mach 6 and the same Reynolds number for typical wind tunnel conditions (left), typical flight conditions (middle), reproduction of wall-to-recovery temperature ratio in the wind tunnel (right), reprinted from [16]

the perturbations causing transition both in flight and in wind tunnel tests. Both wind tunnel and flight environment have to be finely characterised to numerically recreate the environmental effects. This can be done both experimentally, by doing free-stream fluctuations measurements [17, 18], or numerically by conducting simulations of the noise emission of the turbulent boundary layer on the nozzle wall [19, 20], or combining both experimental and numerical techniques by using data assimilation [21]. Ideally, one could even create a simple reduced order model of the free-stream perturbations that is injectable in high-fidelity simulations. Naturally, this work on free-stream perturbations also leads to preliminary studies towards the hypothetical development of a quiet wind tunnel that produces a test environment much closer to flight; this is part of the European SILENT project [22] which aims to develop a quiet hypersonic wind tunnel to better replicate flight conditions (see figure 13). In particular, stability based control techniques can be integrated into nozzle design loops to further enhance their effectiveness [23]. These methods provide a robust framework for mitigating instabilities and ensuring laminar flow over extended regions. DNS could also be used to ensure that the designed nozzle will be quiet.

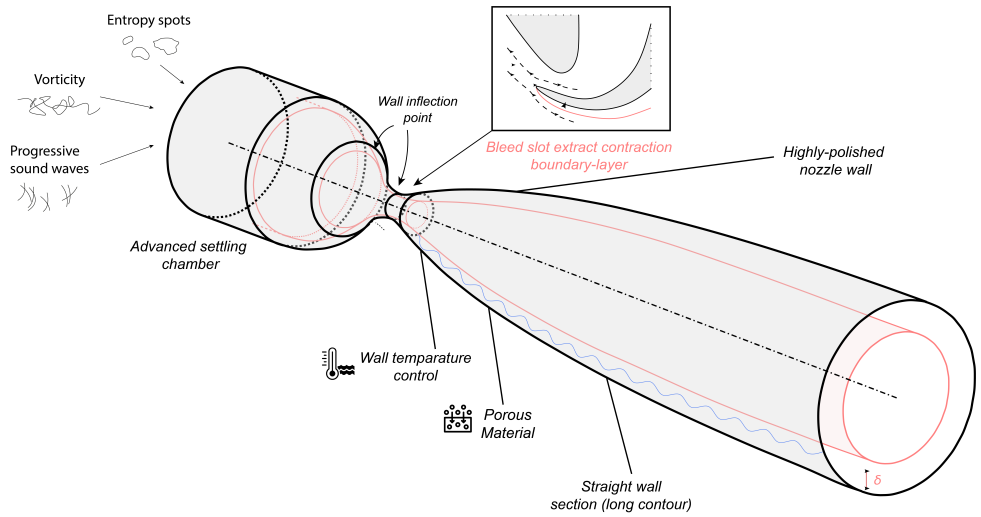


Figure 13 – Illustration of the different control techniques applied to wind tunnel nozzle flow to maintain laminarity and thus produce quiet flow. Work conducted in the scope of the EDF SILENT project, reprinted from [22].

While a quiet wind tunnel would allow recreating an environment closer to flight, the strong impact of free-stream disturbances highlighted here advocates for the need for scientific, highly instrumented flight test campaigns to assess "true" flight disturbance conditions. In this context, the FREHyA project, a collaboration between CEA and ONERA, aims to study the feasibility of a highly instru-

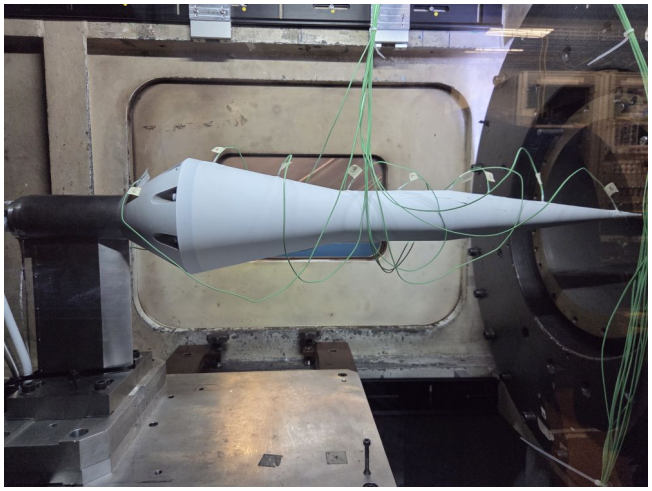


Figure 14 – Commissioning of a liquid nitrogen wall cooled model to recreate wall-to-recovery temperature ratio similar to flight, work conducted in the scope of the FREHyA project.

mented payload dedicated to the investigation of shock-boundary layer interactions (SBLI) and transition. The project also includes a comprehensive ground-based test campaign to replicate the expected flight conditions and validate the transition mechanisms observed. One key point of the ground part of this project is to investigate is the ability to recreate better similarity on ground, this led to the design and commissioning of a nitrogen cooled model (see figure 14).
Given the complexity of replicating flight conditions in ground tests, shared experimental campaigns across different facilities are also a good solution to allow researchers to isolate the impact of specific parameters (e.g., noise spectra, wall temperature) and validate numerical models against a broader range of conditions. In this context, NATO Research Task Groups (RTGs) provide an ideal framework for organizing such collaborative efforts, bringing together experts from academia, industry, and national laboratories from both sides of the Atlantic. The recently launched NATO AVT-439 task group, focused on studying hypersonic boundary-layer transition in realistic condition provides an ideal framework for organizing such efforts.
Finally, all these studies must translate toward engineering applications, which requires development and integration of transition models into automated, high-performance CFD workflows. For example, the work presented in [24] on anisotropic mesh adaptation demonstrates how automation from CAD to post-processing can drastically reduce setup time while maintaining accuracy for complex hypersonic flows. Future efforts should focus on embedding transition prediction tools (e.g., boundary layer extraction, stability analysis, receptivity models) into such automated frameworks [25].

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