Computations of Laminar-Transitional-Turbulent Flows over Supersonic Aircraft: Skin Friction Prediction and Reduction

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In the framework of supersonic research, optimising supersonic transport aircraft performances requires a good evaluation of the total drag, one of the main components being therefore the friction drag which accounts for approximately 35% of the total drag. The paper provides an overview of investigations related to the problem of friction drag estimation at supersonic speeds.

Attention is mainly focused on two topics: skin friction reduction by laminar-turbulent transition control and turbulent skin friction drag prediction at very large values of Reynolds numbers.

- **Laminar-Turbulent Transition Control**

The development of a new, economically viable, supersonic aircraft first requires to delay the onset of transition as much as possible because transition separates the laminar flow region with low drag from the turbulent region where drag dramatically increases. Giving the large sweep angles and the large chord Reynolds numbers of supersonic aircraft, techniques combining wing shaping and suction in the leading edge region seem to have the greatest potential.

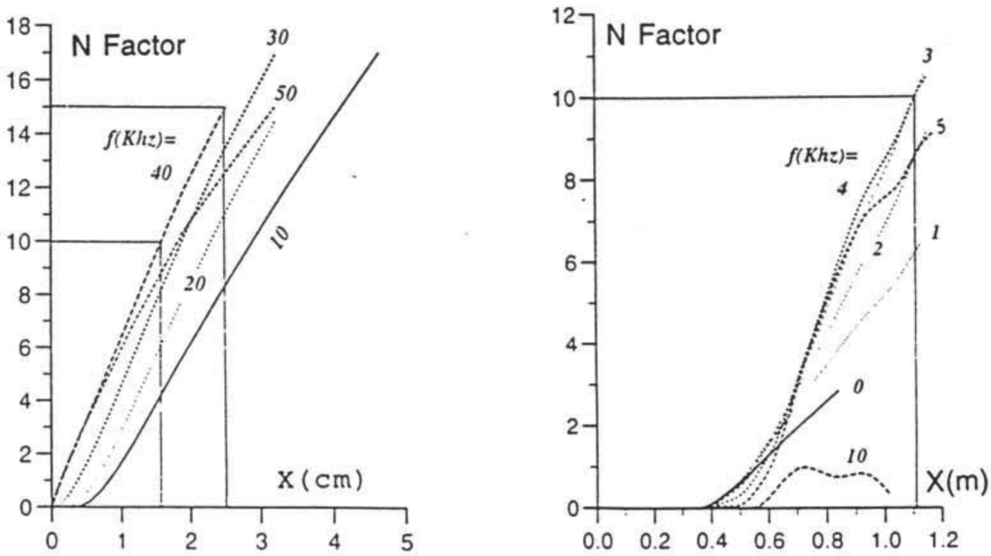
The importance of laminar-transition and its effect on skin friction in subsonic vehicle drag has been investigated for many years and is well known. This is no longer true for supersonic vehicles due to the lack of numerical and experimental results. Therefore, a preliminary study has been conducted at ONERA in order to numerically investigate the effect of suction on laminar-turbulent transition for realistic configurations.

It is well known that transition is triggered by the amplification of unstable disturbances which originate in the leading edge region, propagate in the laminar boundary layer and finally break down into turbulence. Three-dimensional, compressible linear stability theory has been used for the purpose of transition prediction. The computations provide the unstable frequency range, the wavenumber vector direction of the disturbances as well as their growth rates. Depending on the wavenumber direction, distinction can be made between streamwise and crossflow instabilities. The local growth rates are then integrated in the streamwise direction, and it is assumed that transition occurs as soon as the total growth rate of the most amplified disturbances reaches a critical level denoted as N (e^N method). In this study, it has been assumed that the critical N factor lies between 10 and 15, for free-flight conditions.

The e^N method has been applied to the wing of a Mirage 2000 aircraft in flight conditions (Mach number=1.5, altitude=12.2 km, geometric sweep angle of the wing=58°). The computations were carried out at a spanwise location corresponding to 37% of the span. In "natural" conditions, i.e. without laminar flow control, transition occurs very close to the attachment line and it is generated by crossflow disturbances, i.e. by disturbances with a wave vector nearly perpendicular to the external streamline. It can be seen in figure 1 that the unstable frequency range is rather high, of the order of 30-40 kHz.

In order to delay the onset of transition, several means have been considered : reduction of leading edge radius, wall cooling and suction in the vicinity of the leading edge [1].

If leading edge radius is reduced by an order of magnitude, the downstream movement of transition is very small (a few centimeters). Wall cooling also has a negligeable effect on transition location, because it cannot modify substantially the inflectional instability of the crossflow velocity component. It turns out that suction is by far the most efficient tool for laminar flow control, but its efficiency strongly depends on the chordwise distribution of suction velocity. With a constant suction velocity equal to 0.1 percent of the freestream velocity and applied from 0 to 20 percent chord, transition takes place at a chordwise position close to 0.3 m with N=10, and close to 1 m with N=15. This uncertainty illustrates the need for fundamental experiments aimed at calibrating the e^N method in supersonic flow. Considering a linearly decreasing suction velocity (with the same total mass flow rate) leads to more interesting results : with N=10, transition occurs at about 1 m from the leading edge (figure 2). These numerical results demonstrate that a uniform suction distribution performs less well than intense suction near the attachment line, with decreasing suction velocity farther downstream. Another observation is that the most unstable frequencies are now much lower than in the reference case. This can be explained by the fact that the unstable frequency range decreases as the boundary layer thickness increases.



Mirage 2000 Wing

Fig. 1. "Natural" Transition Fig. 2. Effect of suction.

The conclusion is that suction seems to be a promising tool for supersonic wing laminarization, a result which has to be checked experimentally. Another remark is that the previous results have been obtained by using a linear stability theory. Investigations are currently performed at ONERA in order to model nonlinear phenomena by using Parabolized Stability Equations (PSE).

- **Turbulent skin friction Drag Prediction**

Once maximum flow has been controlled and maintained over maximum of the wing and of the upstream part of the fuselage, through either artificial active or passive means, the turbulent viscous flow can thus develop and therefore be evaluated; as a consequence, the friction forces could be deduced. The use of many empirical methods, correlating friction to local Reynolds number, pointed out discrepancies between the estimated total drag from such tools and the measured one during flight tests. One has to keep in mind that a 5% variation in local skin friction coefficient equates to over 1 tonne of payload, for Concorde-type aircraft [2].

So, in terms of aircraft performances, one must be able to estimate turbulent skin friction drag at very high Reynolds and Mach numbers that cannot be obtained in wind tunnel tests. This is the main reason why turbulent friction measurements were performed by Aerospatiale on an Air France Concorde in July 1990. Of course, with an aircraft in airline service, tests had to be rather simple, in order to be easily installed, but accurate enough to validate the measurements. Small direct skin friction balances are difficult not only to use, but also to construct or to incorporate in existing geometries; on the other hand, Preston tubes (Pitot tubes fixed to the surface) continue to be widely used. As they are easy to install on an aircraft, that was worthwile examining their performance during flight tests [3].

The instrumentation included Preston tubes, static pressure sensors and surface temperature probes placed at six carefully chosen inspection doors on the wings and fuselage (figure 3). The aircraft flew at stabilised levels of a few minutes for transonic and supersonic Mach numbers (varying between 0.5 and 2.1), allowing friction measurements at local Reynolds numbers covering a decade from 47 up to 488 millions.

Using different types of Preston tubes calibrations, established for either incompressible or compressible flows, the skin friction coefficient could then be estimated for every Pitot tube location, at each flight level.

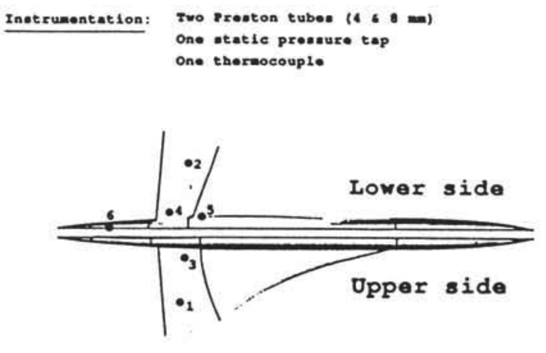


Fig. 3. Preston Tube locations

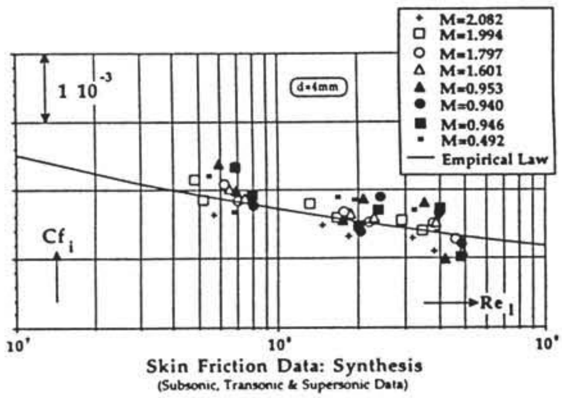


Fig. 4. Skin friction Data: synthesis.

Preston tube measurements have been very consistent: indeed, a fairly nice repetitivity was found between the two flights at comparable Mach numbers. The surface temperature probes allowed to verify that the aircraft surface reached the adiabatic thermal equilibrium, justifying the use of calibration laws without any heat flux correcting factor. One could be very confident in those data, though measurements performed under flight conditions are very often ticklish.

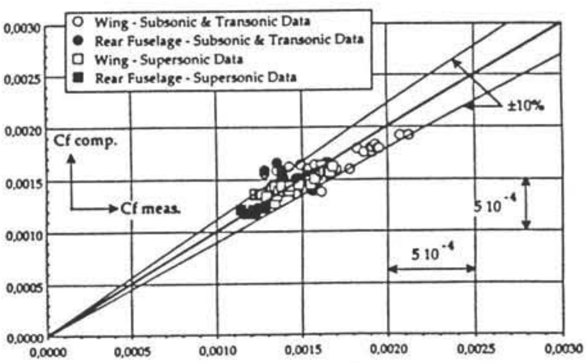
When comparing results to data provided by standard empirical friction laws, rather important discrepancies existed on the determination of the skin friction coefficient (figure 4). Nevertheless, it is useful to point out that the experimental scatter, at a given

Reynolds number, is quite comparable and even less than the one obtained by Fischer and Saltzman on the XB-70-1 aircraft [4]. These differences were mainly attributed to the lack of accounting three-dimensionality, compressibility and pressure gradient effects, when applying empirical power- or logarithmic-type laws.

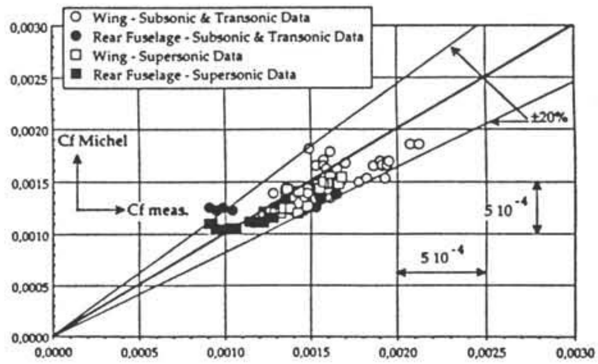
Thus, a more detailed analysis has been pursued by carrying out inviscid and viscous computations for the flight test conditions and comparing the computed to the measured local turbulent skin friction coefficient.

For the inviscid field, the method developed by ONERA and Aerospatiale, deals with the 3D Euler equations written in the conservative form, discretized using a cell-centered difference scheme and solved by an implicit upwing finite volume scheme. The 3D multi-blocked structured grids had 365,000 (513,000) nodes for supersonic (subsonic) conditions and 14,300 nodes on the aircraft surface. Comparisons between the computed pressure field and data obtained in flight from pressure taps have shown a rather satisfactory agreement. As the inviscid wall flow is imposed as external boundary condition for the viscous calculation, one could deduce that boundary layer computations have been performed under conditions very close to those of the stabilized flight levels.

The three-dimensional boundary layer equations are solved using a code developed at ONERA/DMAE. The numerical scheme integrates Prandtl equations along the local streamlines within the viscous layer, which are sub-characteristic lines. The discretization of the equations is done in the plane tangent to the surface of the aircraft, using a coordinate system which respects the metric properties of the fuselage to express the derivatives of the velocity. Then, the integration of the boundary layer equations could be performed using their local cartesian form, even in areas where the mesh is not regular. For turbulence modelling, a zero-layer approach (mixing length formulation) and a two-layer approach solving transport equations for the turbulent kinetic energy and dissipation rate were both applied.



Skin Friction Data Analysis: Computation / Flight



Skin Friction Data Analysis: Empirical Law / Flight

Skin friction Data

Fig. 5. Computations/Flight meas. Fig. 6. Flight meas./Empirical laws.

From results obtained at different transonic and supersonic aircraft Mach numbers, the agreement between the measured and computed skin-friction coefficients is excellent, especially on the wings. Indeed, no more than 10% difference has been found (figure 5), while greater discrepancy was observed when comparing to empirical laws (figure 6).

Viscous computations revealed that a great part of the upper wing could be straightly influenced by the boundary layer developing along the fuselage, though interfering with couple of Preston tube locations. Such a connection is strengthened at relative large values of the lift coefficient. Along the rear part of the fuselage, the boundary layer is rather thick and the absence of coupling between inviscid and viscous calculations as well as the opportunity to solve boundary layer equations could then be subject to discussions. Complementary work would be needed, since one could question the validity of either a two-layer method or a mixing length scheme at such high values of Reynolds numbers ($\sim$ 500 millions).

As a conclusion, the analysis of the afore-mentioned flight test data has pointed out the necessity to perform boundary layer calculations, rather than to use existing empirical laws, if one needs to determine precisely enough the friction drag coefficient of a supersonic aircraft. Because of low computational time, one could easily think to introduce such viscous calculations for industrial applications; however, those computations need the knowledge of the pressure field which, for the moment, requires greater computational time. But, such a precise knowledge of the friction drag remains important enough for the aircraft payload to include that method during the development of a new supersonic transport programme.

Reference

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