LAMINAR FLOW EXPERIMENTS WITH A LARGE HALF MODEL IN TRANSONIC FLOW

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Abstract

A 1 : 2 scale half model of the VFW 614-ATTAS experimental aircraft with a laminar glove was tested in Europe's largest transonic wind tunnel, the ONERA-S1MA at Modane, France. With these experiments, a series of tests concerning laminar flow was completed, starting with flight tests with the ATTAS, continuing with an 1 : 13.5 scale pilot model in the TWG Göttingen, leading to low speed tests with the large model in the DNW, Netherlands and finally to high speed tests in the S1MA. This test was carried out in a cooperation of Deutsche Airbus, the DLR and Technical University of Berlin.

The aim of these experiments was to give a proper correlation between flight test and wind tunnel data for laminar wing flow, and to acquire further data with the wind tunnel model for conditions which cannot be flown with an aircraft.

Many different sensors have been installed on the model, such as hot films, piezo foils, strain gauges and so forth. For detecting transition locations the infrared technique was used. This technique has been an useful tool by determining the quality of each run, as there sometimes has been some contamination of the wing's leading edge and the glove, causing unacceptable turbulent wedges. Transition positions up to x/c = 60 % were reached. Furthermore, three kinds of instability have been inspected carefully, Crossflow Instability, Tollmien-Schlichting Instability and Attachment Line Instability. The flight test pressure distributions have been matched, some tests at nearly the same Reynolds number.

1. Introduction

In a rough assumption, the benefits, i.e. the friction drag reduction of an airfoil with a laminar boundary layer will reduce the profile drag by some 50 % and therefore lead to a total aircraft's drag reduction of up to 15 % /1/. So introducing a laminar airfoil to transonic aircraft will be quite a high step in aerodynamics, comparable to the introduction of the transonic airfoil in the early 70's.

In this report, some of the results of a national research program on laminar flow will be presented, mainly the wind tunnel experiments with a model of this aircraft's wing. Fig. 1 shows the German Flying Test Bed ATTAS (Advanced Technologies Testing Aircraft System) with a laminar glove covering one third of the right hand wing. On a swept wing, there are three types of transition, Fig. 2, the mechanisms depending on Reynolds number and geometry /2/ /3/.

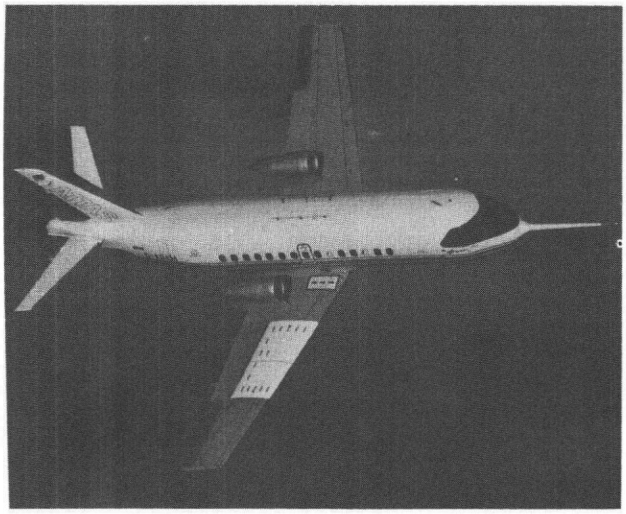


Fig. 1: ATTAS Flying Test Bed with Laminar Glove

In case of a pure 2-dimensional flow the Tollmien-Schlichting Instability (TSI) occurs. This means that the amplitude of 2D-waves will be amplified until transition, which occurs along a 2D-line normal to the flow. It can be delayed mainly by a negative pressure gradient, i.e. an accelerated flow. However, the flow cannot be accelerated along the whole airfoil, so at transonic speeds transition will take place at latest at the shock position. Having a pressure gradient in spanwise direction because of a wing's sweep leads to Cross Flow Instability (CFI). Here small vortices with their core in streamwise direction roll up until transition, which occurs along a sawtooth-like line. Finally, at high Reynolds numbers on a swept wing transition can also take place along the leading edge, the so called Attachement Line Transition (ALT). This is worst case as it leads to a fully turbulent wing.

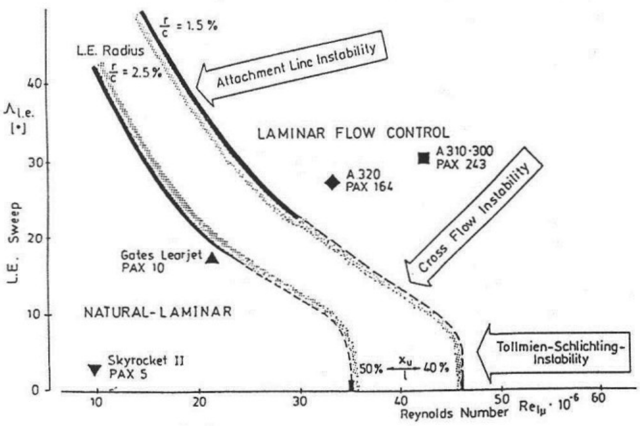


Fig. 2: Boundaries for Swept Wing Natural Laminarization

For regional aircraft, which fly at moderate transonic speed a low sweep wing up to about 20° may be built. In that case, one has to cope with TSI and CFI, so a Natural Laminar Flow (NLF) - wing just by geometry might be possible. Todays larger transonic aircraft like the Airbus have sweep angles of about 28° where CFI and ALT dominate the transition process. To avoid these two calls for a leading edge suction device for about the first 10 % chord, after that NLF will be possible again. That concept is called Hybrid Laminar Flow Control, HLFC.

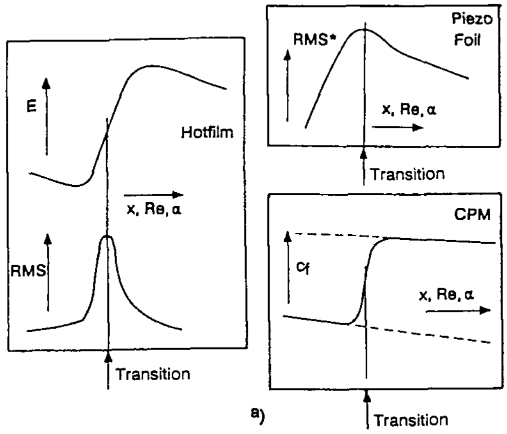
In a combined effort, industry, research institutes and universities worked on NLF in the first phase of the Transonic Laminar Flow (TLF) program. Two main experiments took place during the first phase of the TLF program:

- flight tests with the DFVLR-ATTAS, (based on a VFW 614 aircraft), carrying the above mentioned laminar glove /4/.
- tests with a 1 : 2 scale half model of this wing in the DNW (Deutsch-Niederländischer Windkanal) in Netherlands at low speeds and in the ONERA-S1MA (Soufflerie 1 at Modane/Avrieux) in France up to transonic speeds.

2. Experimental Concept

In the literature there is still a lack of sufficient high quality data about CFI and ALT, especially for flight tests and correlation to wind tunnel experiments. So the laminar glove was not designed for performance, but for studying the three transition phenomena seperately, which can be done by a certain pressure distribution philosophy. Purpose of the wind tunnel tests was the validation of laminar testing techniques in a wind tunnel, the comparison to flight test data, further information of the glove's behaviour at nonflyable conditions and gaining experience from such test in different wind tunnels for later measurements of an aircraft model with a laminar wing.

Many fundamental experiments were carried out before and during these tests to gain knowledge about certain sensors and model behaviour: Special measurement technique campaigns took place in the Deutsche Airbus-LSWT in Bremen (Low Speed Wind Tunnel) to compare the results of e.g. hot films, piezo foils etc. during boundary layer transition. For this comparison the transition "point" had to be defined first. Such a definition applied to the various measuring methods is given in Fig. 3 together with a comparison of results from different sensors in low speed flow. During these tests a special device KST (Kleiner Sensor Träger) carrying many sensors was developed for boundary layer measurements, which has been in use on the large half model, too. Furthermore, a 1 : 13 scale pilot model was tested in the TWG (DLR Transsonischer Windkanal Göttingen). Purpose of this test was the proof of concept of influencing the pressure distribution by remote controlled flaps as it was planned to "adjust" a pressure distribution on the model given from the flight tests without knowledge of incidence, exact yaw angle etc.



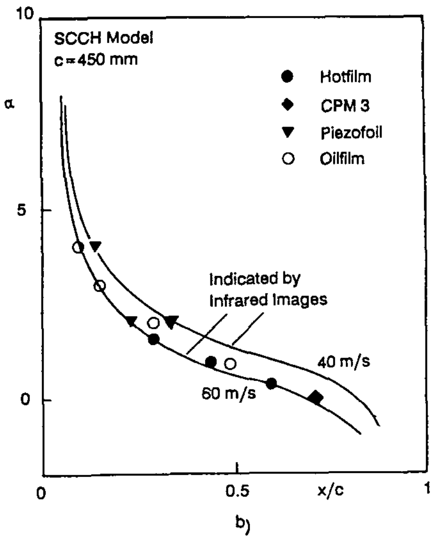


Fig. 3: Transition Determination by Various Measurement Techniques: a) Definition of Transition Point for Various Sensors, b) Comparison in Wind Tunnel Test

The experimental techniques used in the flight test were static pressure distribution, infrared images (IR) and hot films /5/. The ATTAS glove was built with glass fibre sandwich technology around the VFW 614's profile between two flap fairings from x/c = 5 % at the lower to x/c = 54 % at the upper side. Heating elements were installed right under the glove for improvement of the infrared pictures at level flight. A special window for the long infrared wavelenght was inserted. For pressure measurements a PSI-system has been used, the modules laying in the wing nose in a heated box. A special probe giving cp = +1/0/-1 as calibration pressures had to be built and installed at the fuselage.

The wind tunnel model as shown in Fig. 4 consists of three parts, built from aluminium, the middle being the laminar glove. A TFN (Through Flow Nacelle) with a removable reduction plate can be installed for engine simulation. The sweep can be varied from 14° to 23°, 18° being basic as the VFW 614's sweep. Three remote controlled flaps are situated at x/c = 80 % to influence the pressure distribution at a given Alfa/Mach configuration. A large boundary layer plate was used. The weight of the model is about 4.5 to (10.000lbs), the wing's

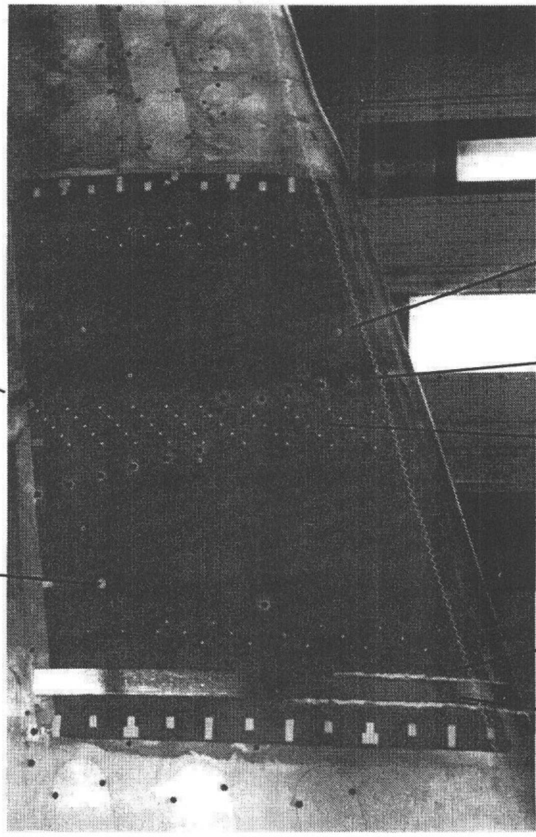


Fig. 4: Measurement Techniques on Laminar Glove

root measures 2 m and the produced lift comes up to about 8 tons. The model was designed for Mach numbers up to 0.80 and Re numbers of roughly 18 Millions. The same measurement techniques as in the flight tests have been applied as standard during the wind tunnel tests. Apart from these, many other sensors were installed on the wind tunnel model: microphones, piezo-electric sensors, strain gauges, acceleration gauges, temperature gauges, a hot wire, a boundary layer rake, CPM probes (Computational Preston Tube Method) and outside of the glove turbulence generators and liquid crystals (temperature sensitiv foils and shear sensitive liquids) /6/.

3. Results from Tests with a Large Scale Wind Tunnel Model

Fig. 5 shows the Mach/Re-number range for the wind tunnel tests and some of the flight tests. Because of the model's size, a number of test conditions in the wind tunnel experiments correspond to the same values as those for high altitude flight test. One of the main problems during all the experiments were particle impact on the leading edge. During flight tests these were insects or ice crystals, during wind tunnel tests all kinds of particels like dust, dirt, paint, grease, rust etc. were found on the model. For the first few runs the sensors were covered with tape which was necessary as shown in Fig. 6. A typical IR picture is shown in Fig. 7, featuring transonic effects (shock), particle effects (turbulent wedges) and laminar flow up to the shock location.

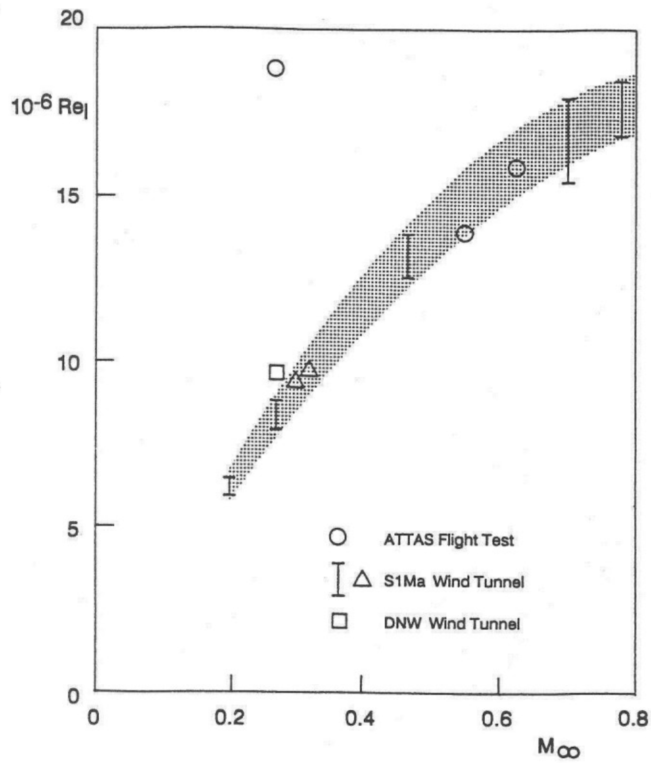


Fig. 5: Ma/Re Number Envelope during Wind Tunnel and Flight Test

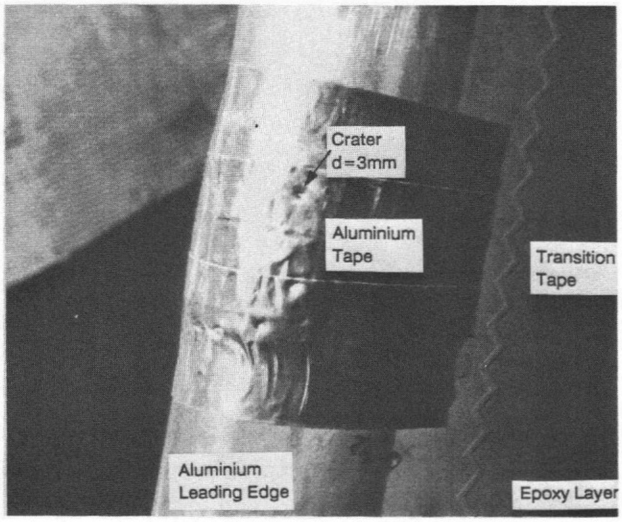


Fig. 6: Particle Impact on Aluminium Tape Covered Leading Edge During Preliminary Wind Tunnel Runs

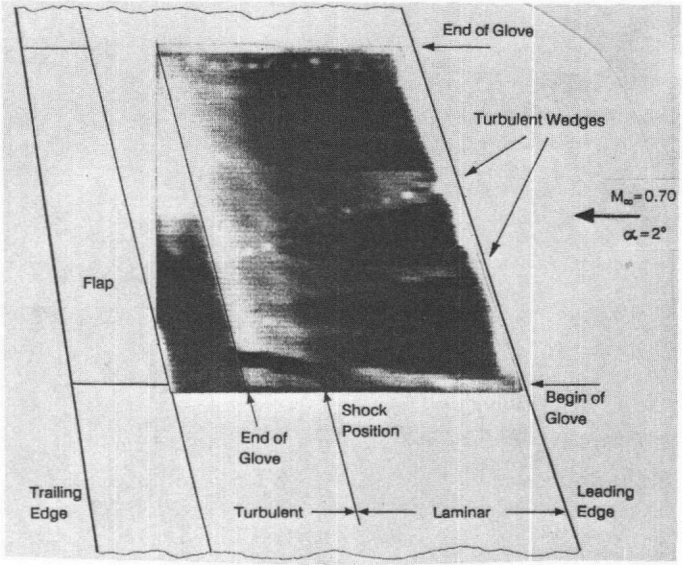


Fig. 7: Infrared Image of the Laminar Glove at Transonic Speed

3.1 Measurements on Different Types of Transition

The different sensors used during the TLF tests as shown in Fig. 4 react differently on the physical effects during transition:
Temperature sensitive liquid crystals as well as the IR technique depend on a temperature difference between wall and outer flow and react on lateral thermal conductivity leading to different wall temperatures (at high speed recovery factor effects are added, too). Hotfilms or, more general, near wall mounted hot wires rely on and measure the longitudinal conductivity for the transportation of their induced heat. Flush wall mounted pressure transducers including microphones sense the normal forces which are regular patterns as well as noise with an increasing chaotic structure. The large scale flow structures as found in the finally turbulent boundary layer state can lead to model movements which are felt by strain gages. The piezo-electric elements in general sense all changes like thermal, shear and normal forces and mechanical bending. Finally, Preston tubes as for CPM react on the increasing total pressure in the boundary layer and hence the augmented velocity gradient in the near wall region.

In Fig. 8, typical pressure distributions together with IR images and hot film signals are shown for TSI and CFI. Having a strong pressure gradient in chordwise direction with a following pressure increase will lead to TSI at that point, while different chordwise gradients along span will lead to a pressure gradient in spanwise direction and, therefore, to CFI. The hot film signals are typical for many time resolving sensors: it starts at a low amplitude level, then first turbulent spots appear, which merge together until only some

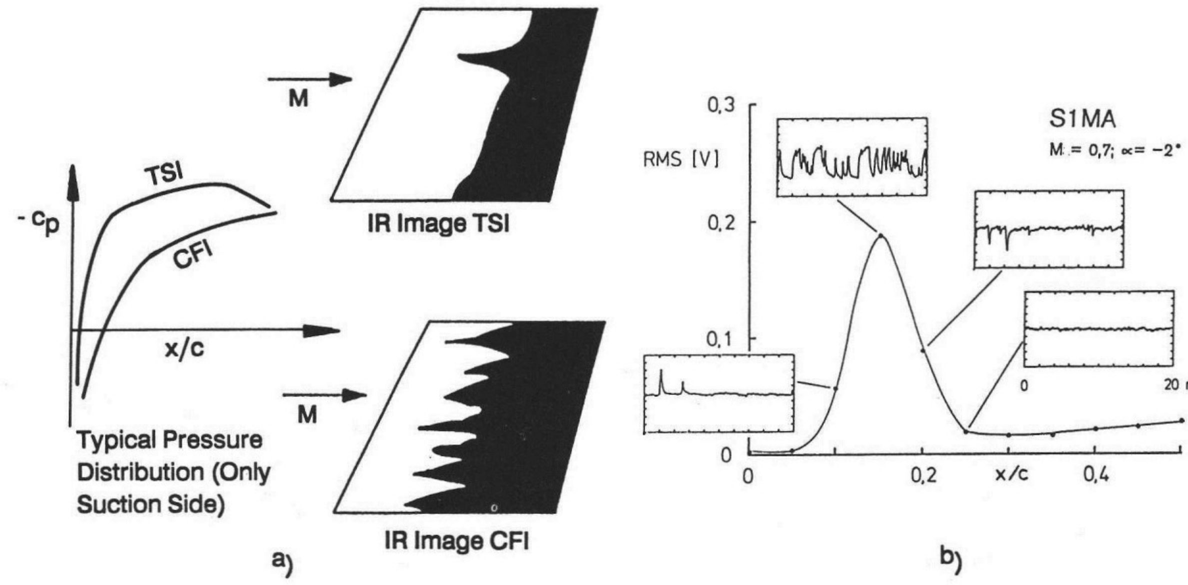


Fig. 8: Indication of Transition: a) Static Pressure Distribution leading to TSI or CFI, b) Hotfilm Signals Along Chord

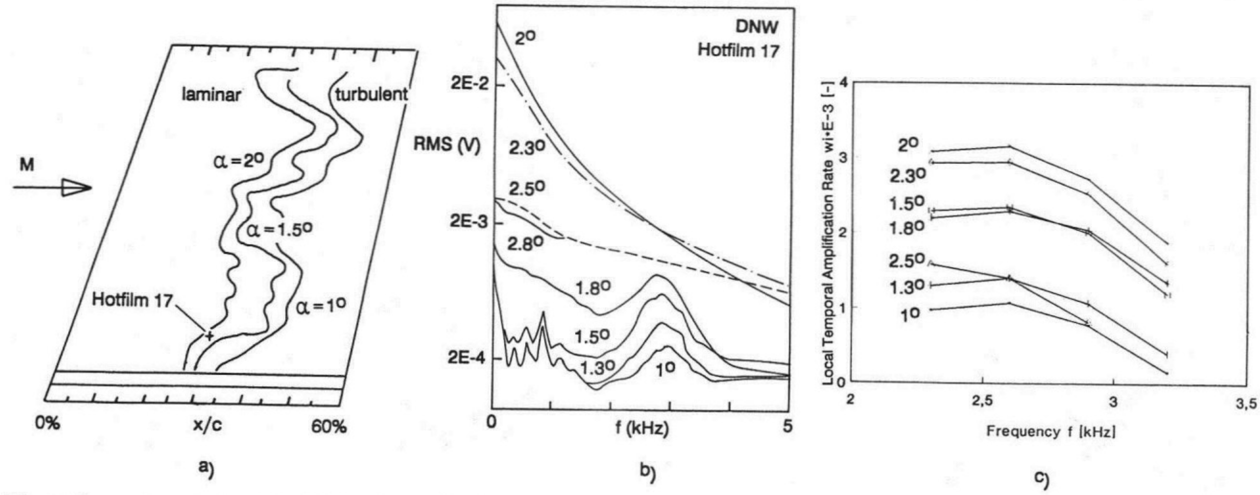


Fig. 9: Comparison of Transition Determination by Experiment and Calculation: a) Infrared Image, b) Hotfilm Measurement, c) Stability Analysis (SALLY)

"laminar leftovers" remain which finally dissapear. At last i.e. in a turbulent boundary layer the mean level of such a signal is increased compared to the laminar one but by far not as high as during transition itself /7/. That means that without knowing the state of a boundary layer or the history of such a signal it is often difficult to distinguish between a laminar and turbulent time signal.

One of the most vital values in laminar flow technology is the amplification rate of waves in stability analysis. In Fig. 9 experimental and theoretical results are compared in that sense for some DNW data points. From the hot film RMS (root mean square) values the Tollmien-Schlichting frequency can be read as roughly 2.8 kHz and the flow remains laminar up to about 1.8° incidence. The theoretical curves have been calculated for different frequencies and show an amplification maximum at roughly 2.7 kHz and a drop of the amplification rate just after 2° incidence. This means that experimental techniques and theoretical results agree very well. So apart from other informations using such sensors can reduce calculation time as they already give the frequency needed as an input for the stability analysis.

The ALT had to be forced by a trip wire, during fligth tests as well as during the wind tunnel tests, Fig. 10. Increasing sweep as on large transport aicrafts makes its natural appearence more likely. The diameter of the trip wire was 2.5 mm for the flight and 0.3 mm for the wind tunnel tests, choosen to meet the Pfenninger-Poll-criterion /8/, which says that the Re-number based on momentum loss thickness should exceed 100 for ALT. Because ALT starts at the leading edge and leads to a sudden transition along the whole chord it is the most dangerous type of instability. On a given profile shape and wing sweep it can be dealt with by suction. During the flight tests some hot films along the leading edge were able to detect the ALT.

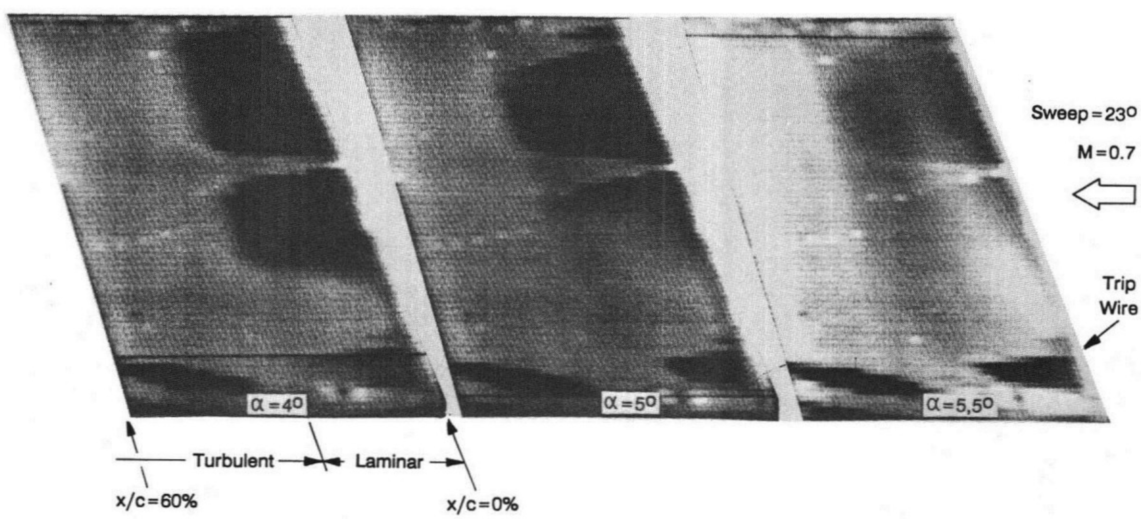


Fig. 10: Infrared Image of a Test to force Attachment Line Instability by a Trip Wire

3.2 Flow Quality for Laminar Testing in the DNW and S1MA Wind Tunnel

As mentioned above, during transition the pressure and velocity fluctuations in the boundary layer increase. On the other hand, increasing fluctuations in the free stream flow may lead to premature transition, e.g. a high noise level in a wind tunnel. Every wind tunnel has its individual noise level and noise spectrum, originating from the propulsion system, the corner flow, turbulent boundary layers, diffusors etc., measurable in an empty test section /9/. The existance of a model in the test section and the flow around it increases this noise level and any further device increases it even more, as shown in Fig.11. Apart from other features, from this figure it can be taken that the noise level of the S1MA is quite low at low velocities, even not too far away from the exceptionally high quality DNW results, which was not known before.

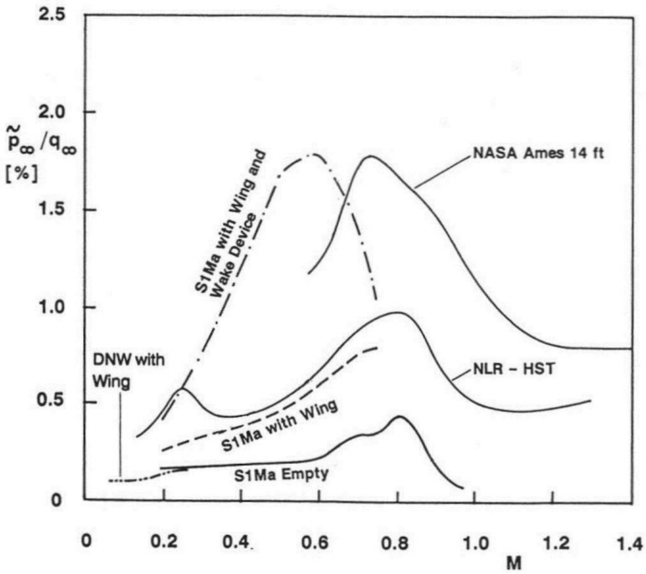


Fig. 11: Broad Band Noise Level in Different Wind Tunnels and Effect of Installations

One target of these tests was a very accurate measurement of the profile drag coefficient, which can only be done by wake measurements. If the distance of the rake from the trailing edge is less than about one chord length, the static pressure must be measured, too, e.g. with 5-hole-probes. The rake used in the S1MA as shown in Fig. 12 and as mentioned in Fig. 11 was moveable in all directions, so that the influence of CFI-wedges could be measured by spanwise traversing. It consists of the rake itself, with 45 Pitot- and nine 5-hole-probes and a very large traversing mechanism, standing about 3 chord length downstream of the trailing edge. Even at that distance it caused disturbances in the whole test section which could be measured by microphones in the test section and even with most of the sensors on the model. These disturbances lead to an increase in noise level of about factor 3 and therefore an upstream shift of transition location of up to dx/c = 7 %.

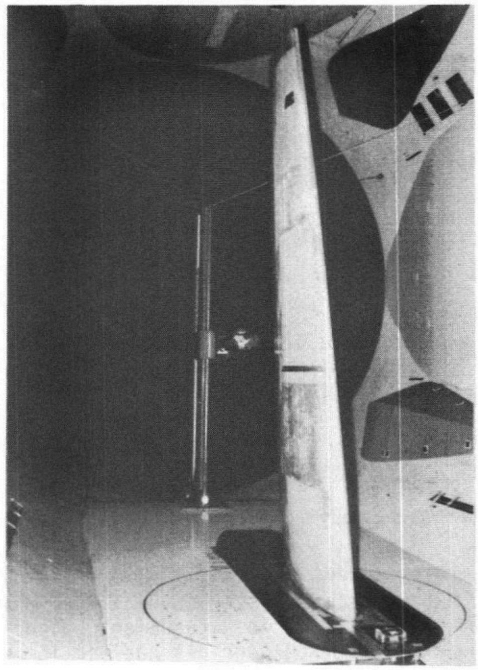


Fig. 12: Large Half Model and Wake Traversing Mechanism in the Test Section of the S1Ma

So one has to be very careful with any kind of instrumentation in a wind tunnel as well as with the flow quality when doing laminar experiments. One must be aware that the instrumentation will have some influence on the results and therefore, measuring the wind tunnel flow with time resolving sensors is vital for laminar testing, as its correlation with sensors on the wing shows whether or not the transition is caused by outer flow or just by profile or wing effects.

3.3 Comparison to Flight Test Data

One of the aims of the wind tunnel test was the direct comparison of data points with flight test data for the same pressure distribution at given Mach- and Re-numbers. For this purpose the pressure distributions from the flight tests were installed in the wind tunnel computer and for a given data point at given Mach number displayed together with the measured one from the model. In an iterative process incidence and flap deflection were adjusted until both distributions matched, Fig. 13. Now the signals from hot films, the IR images etc. could be compared and having the complete pressure distribution around the whole profile theoretical calculations could be done. This procedure has been done for 10 flight data points at different Mach numbers, from 0.27 to 0.70. Taking the required high quality data and accuracy into account an iterative procedure as described plus special model features and advanced measurement techniques were demands resulting in experienced personnel during such a test campaign. Laminar testing for applied research is very different from standard industrial wind tunnel tests.

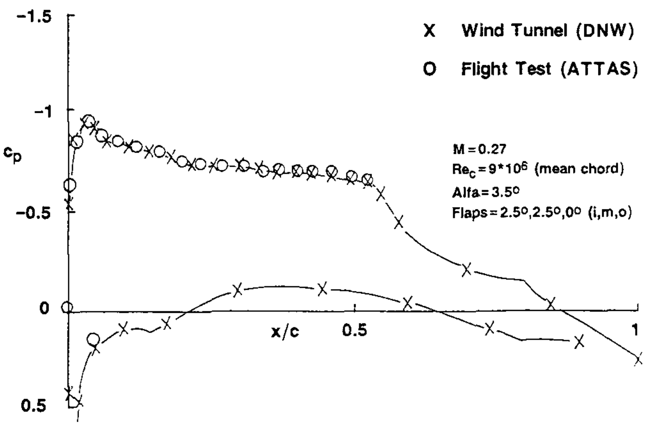


Fig. 13: Comparison of Pressure Distributions between Flight Test and Wind Tunnel Test

In theoretical stability analysis the e-power-N-method is most common /10/. Here, the amplification of the amplitude of waves is calculated for different frequencies, the logarithm of the relation of final to starting amplitude being the so called N-factor. Such curves for all frequencies lead to an envelope along chord and knowing the transition point from e.g. an IR image gives the N-factor for this data point, Fig. 14. As CFI is more sensitive for lower frequencies which occur in a wind tunnel, the correlation between wind tunnel and flight tests is better for TSI than for CFI. Using HLFC technique in wind tunnel tests may lead to a better agreement.

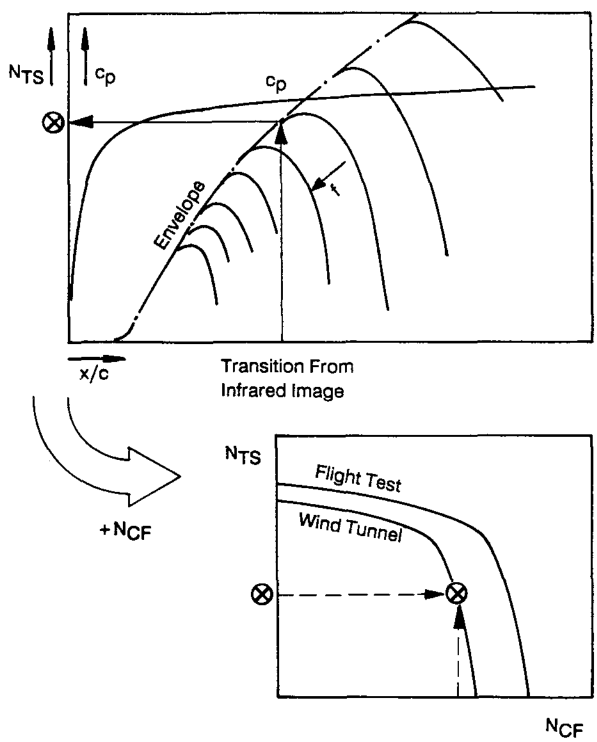


Fig. 14: Sketch of N-Factor Determination by Theoretical Analysis and Flow Visualization

4. Conlusions and Outlook

With all the techniques mentioned above, many data were acquired: some 100 MB (Mega Byte) for each of the 1500 data points. All these data can be evaluated for each single sensor, but correlated to the other sensors, too. Data evaluation is still in progress, together with comparison to theoretical results. In Fig. 15 transition locations are shown for different Mach numbers and sweep angles. At low incidence there is always CFI starting at the leading edge and coming downstream with increasing incidence. At the highest x/c-position it changes to TSI which then comes upstream again. In case of transonic speed transition occurs at the shock positions at latest. Similar evaluations have been carried out concerning engine effects or the comparison of wind tunnels etc.

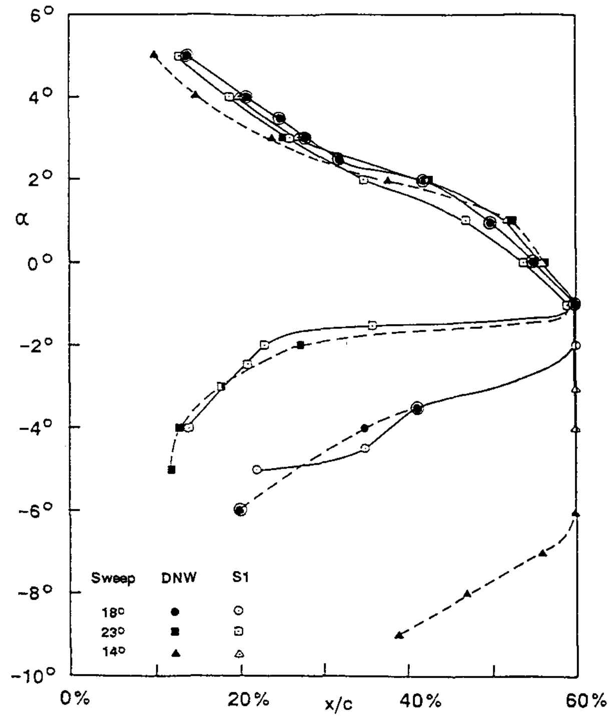


Fig. 15: Transition Location on Laminar Glove as Function of Incidence and Sweep

Summarizing, the targets of these tests have been reached very well, lots of experience has been gained and the experimental data can be used now as inputs for testing and improving theoretical stability analysis. So during the TLF program the three branches theory, design and experiment have improved in their specific as well as the combined work.

The glove design was well suited for these tests as TSI and CFI could be separated. All types of transition have been observed and parameters like flap deflection, TFN influence, sweep angle etc. were studied. One of the main questions whether or not laminar testing in a wind tunnel like the S1MA is possible was answered positively and a valuable correlation to flight test data was possible. The concept of NLF has been proved. One of the next activities in laminar testing will be to study the

HLFC concept by tests in the S1MA wind tunnel with a model very similar to the one presented in this report. These experiments are part of the European Laminar Flow Technology program supported by the CEC.

5. Acknowledgments

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