ETW – HIGH QUALITY TEST PERFORMANCE IN CRYOGENIC ENVIRONMENT

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ABSTRACT

The capability of the European Transonic Windtunnel ETW to test simultaneously at cryogenic temperatures down to 110 K and at pressure levels up to 450 kPa allows to achieve Reynolds numbers appropriate to full scale flight from low speed up to transonic Mach numbers. This paper briefly summarises the achievements in flow quality and tunnel control to generate the prerequisites for the provision of high data quality over the full operating range. Especially, low levels in unsteady flow quantities can be demonstrated. Based on an excellent data repeatability the smallest increments in aerodynamic quantities can be measured with high confidence, leading to a relaxation on model surface quality requirements. The implementation of a unique anti-vibration system yields gains in model safety and extended pitch ranges. An extraordinary thermal stability of the new half model balance offers unique perspectives for this test technology in ETW. Improvements in the IR-imaging technique and on-model deformation measurements provide attractive tools for the analysis of the complex flow behaviour as well as the determination of aerolelastic and Reynolds number effects. Some typical results are presented for full and half models. Finally, productivity aspects are considered on the basis of recently performed complex test scenarios.

INTRODUCTION

The original motivation to build facilities for flight Reynolds number simulation of aircraft models was based on significant differences between wind-tunnel and flight, often leading to costly design changes after the first flight. Nowadays, with the availability of advanced CFD methods, some Re effects are better understood but so called indirect Re effects are still difficult to predict.

For convenience, 'direct' effects can be considered as related to a given fixed pressure distribution, like the variation in skin friction due to Re effects. They may follow well defined scaling laws. 'Indirect' effects shall cover all effects causing changes of the pressure distribution, primarily driven by the displacement effects of the boundary layer on the outer non-viscous flow, resulting in modifications of pitching moment, shock wave strength or spanwise wing loading. The Reynolds number dependency of the involved viscous forces complicates any prediction especially when the boundary layer is close to separation. An illustration of the outlined scenario is given in figure 1, taken from [1].

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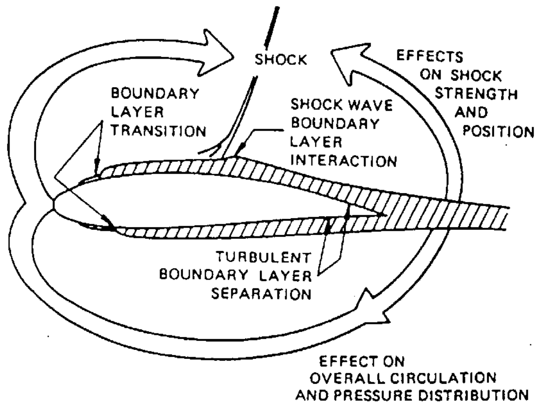


→ **direct Reynolds Number Effect**
⇒ **indirect Reynolds Number Effect**

dominant Re-Number Effect		
characteristic	direct	indirect
lift & pitching moment		x
viscous drag	x	
wave drag		x
drag divergence		x
boundary layer separation	x	
buffet boundary	x	x

Figure 1 : direct & indirect Reynolds Number Effects

Striving for a capability to analyse such effects by experimental investigations on a wind-tunnel model, some challenging prerequisites have to be fulfilled to form a suitable platform for making accurate and reliable scientific statements :

a) an appropriate facility allowing the independent variation of single aerodynamic parameters and covering a wide Re-number range, ideally up to flight conditions,
b) extraordinary steady and dynamic flow quality to generate conditions comparable to flight
c) a high level of tunnel set point stability and repeatability.

THE FACILITY

The ETW facility is a high Reynolds number transonic wind tunnel of Eifel type, using nitrogen as the test gas. It can be operated in solid wall or partially slotted test section mode. For half model testing, side wall slots will be opened, whereas venting areas in the floor and ceiling are provided for full model configuration. The Mach number range is 0.15 to 1.35. By pressurisation, the total pressure can be varied between about 115 kPa to 450 kPa , while temperature is feasible over a range from 110 K to 310 K. Further details can be found in [2] and [3].

This available ability independently control velocity, temperature and pressure provides unique capability for separating true Reynolds number effects from any pseudo Re-effects, induced by the model supporting system or the flow around the model.

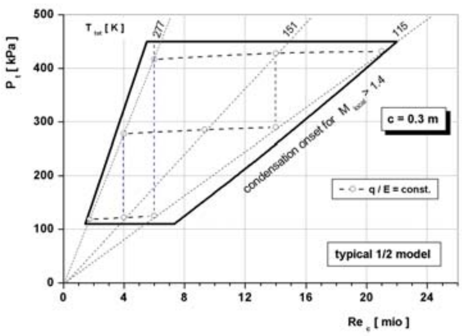
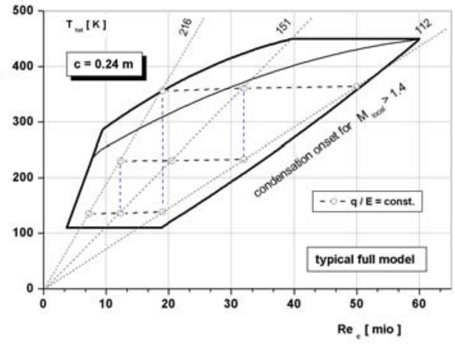


Figure 2 : Operating Envelope for Ma = 0.2

Additionally, effects of wing distortion may be investigated by a variation of dynamic head at constant Reynolds number. Figure 2 (for a typical low speed, high – lift, half-model configuration) and figure 3 (for a large transport aircraft model) illustrate an eventual test scenario targeting for a coverage of the Reynolds number range up to flight conditions including a con-sideration of pure Reynolds as well as aeroelastic effects. The indicated test conditions may obviously not fulfil all requests of an aerodynamicist but the sketched



scenario simply represents the capabilities of ETW.

Figure 3 : Operating Envelope for Ma = 0.9

Tunnel time required to perform tests is dominated by the number of selected temperature levels, as the mini-misation of temperature gradients across the balance is a mandatory prerequisite to achieve highest data quality. Nevertheless, assuming a variation of Mach number in the order of eight per indicated condition still allows to complete the presented scenario in one day for a given model configuration.

FLOW QUALITY IN ETW

When striving for a correct simulation of flight conditions in a wind tunnel , it is obvious that beside the accurate establishment of the two representative aerodynamic quantities, Mach number and Reynolds number, the flow field around the test-object needs to be in all respects comparable to the atmospheric situation of a full scale aircraft.

In general, wind tunnel flow irregularities may be divided into four main categories, namely spatial non-uniformities, swirl, low frequency unsteadiness and turbulence. The first two subjects are usually reported in the form of tunnel calibration. Extraordinary comprehensive experimental investigations have been performed during two major phases undertaken during the mid 1990's. Homogeneities of total pressure of better than 0.15% total head and temperature (less than 0.25 K) could be documented at flight conditions by flow surveys in the model volume. Further details and results can be found in [2] and [3].

As dynamic flow effects may influence boundary layer transition, separation phenomena and shock boundary layer interactions, an assessment of the fluctuations of velocity, sound and temperature was considered to be mandatory. For the fulfilment of this task, a challenging one for a tunnel which is operated at transonic speeds in a pressurised , cryogenic environment, hot-wires, hot-films, microphones, piezo-foils and KULITE pressure transducer have been committed. The central flow field as well as areas near the walls have been covered by measurements including wall mounted arrangements.

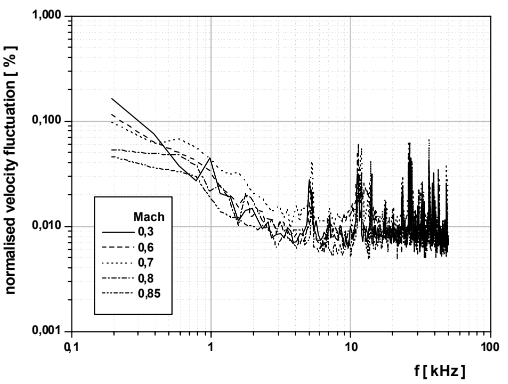


Figure 4 : normalised velocity fluctuations

In figure 4 a spectrum of the normalised velocity fluctuations is presented. The reading of a strut mounted conical hot-film probe exhibits a drop from 0.1 % down to 0.01% at about 3 kHz. Frequencies up to 20 kHz were analysed. The support generated wake could be found as the origin of the peak at about 5 kHz and the resulting harmonics.

Corresponding rms-values in figure 5 demonstrate levels of 0.12% to 0.25% after integration up to a frequency of 20 kHz. The visible trend of decreasing levels with increasing Mach numbers is in good agreement with results achieved with other applied techniques. Corresponding normalised pressure fluctuations show the well known maximum to occur at a Mach number of 0.75. A further increase in flow speed will cause a growth of local supersonic areas suppressing the upstream propagation of acoustic disturbances with a resulting reduction of the noise level in the test section again. The indicated rms-values reflect an integration over a frequency range up to 12kHz.

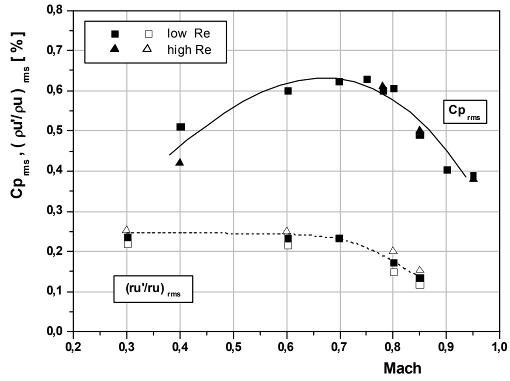


Figure 5 : pressure - & velocity fluctuation level

Another indicator for flow quality may be given by the so called e^n – method[4]. Based on stability considerations of waves, an empirical value of $n = 10$ is allocated for the boundary layer of a laminar wing airfoil in flight. From literature, a proposal to correlate the n- factor to turbulence level is available, where $n = 8.15$ relates to a turbulence level of 0.1%. For ETW n – factors between 7 and 10 have been derived with a majority around a value of 8.

CONTROLLABILITY & REPEATABILITY

In ETW, simultaneous direct control of stagnation pressure, total temperature and Mach number is feasible by combined setting of the blow off system, the injected liquid nitrogen flow and the compressor speed. The relevant aerodynamic quantities are measured by highly accurate flow reference systems installed on tunnel centreline level, to avoid hydrostatic corrections to be applied. For the pressure measurements, a combined unit with an absolute and two more accurate differential transducers are operated to achieve the specified accuracy and resolution across the full tunnel envelope.

Further improvement to Mach number stability during a polar can be achieved by activating the second throat. This technique allows the Mach number to be kept constant within better than +/- 0.0005 when pitching the model continuously with a rate of 0.25 degrees per second.
Typical deviations in Mach number are presented in figure 6, documenting the above stated quality. The

indicated test conditions refer to Reynolds numbers covering the test capability from an ambient non-pressurised facility up to full scale flight condition. It should be emphasised that operating at pressure levels below 125 kPa is outside of the original ETW design specification.

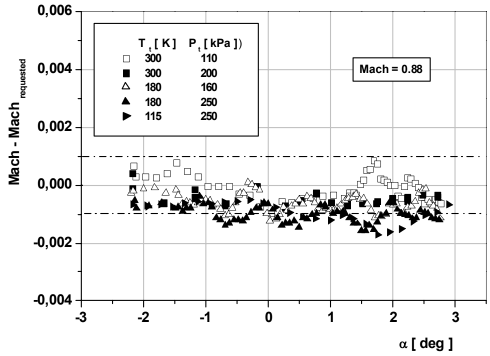


Figure 6 : Mach number variation during a polar

The present wall interference assessment of ETW has been based on comparative measurements testing the same model at identical aerodynamic conditions in a solid and slotted wall test section. As quite small corrections have been worked out, a 2nd entry was performed with a limited number of test points, to increase the confidence in these essential results. After a period of more than one year high standards of model assembly quality were required especially on the attachment of the transition band which was applied for the low Reynolds number cases.

The differences in balance measured drag by the two entries is shown in figure 7. A superb long term repeatability within the expected tight goal of +/- 1 drag count can be quoted. For Mach numbers above 0.8 the model was well within the drag rise area and, additionally, test section choking occurred for the solid wall configuration.

The final fully corrected drag consists of the balance output, the contribution from cavity drag as well as sting and wall interference effects. "Base drag" has been derived from pressure measurements normally taken within the cavity of a model where the sting penetrates the rear fuselage. Hence, striving for an excellent repeatability in drag also requires high quality pressure data. Results from solid wall tests presented in figure 8 partly refer to figure 7. A polynomial fit for the data recorded in the 2nd entry is indicated. Old versus new results are well within a goal

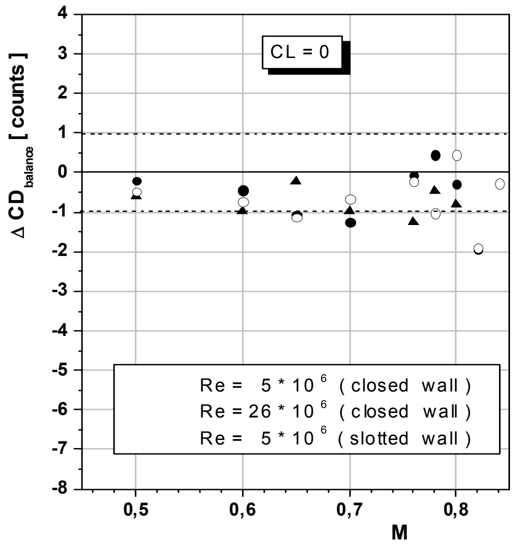


Figure 7 : long term repeat of balance measured drag

of +/- 1 drag count. The Reynolds number of 17 million was generated at two different temperatures by adapting the tunnel pressure. A perfect match is achieved confirming the visible Re trend of the cavity drag component, while the Mach number effect appears to be independent of Re. Note that 1 count is approximately equivalent to a pressure difference of 100 Pa in cavity pressure.

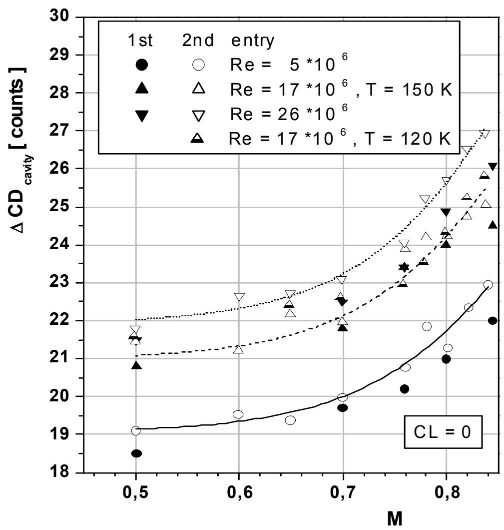


Figure 8 : Repeatability of Cavity Drag Measurements

TEST ASSEMBLY STANDARDS

It is obvious that due to the reduced boundary layer thickness, wind tunnel testing at high Reynolds numbers requires a higher standard of model surface quality than ambient testing. As a consequence, for the first generation of models to be tested in ETW, mirror polished metallic surfaces with a roughness of better than Ra = 0.1 were regularly achieved.

The introduction of non–intrusive measuring techniques at cryogenic conditions requires the application of model coatings, either for reduction of thermal conduction in the case of applying the IR-technique for the detection of boundary layer transition, or to generate a diffuse non-reflective surface for the successful operation of the **M**odel **D**eformation **M**easurement **S**ystem MDMS, which is based on the Moiré technique.

After careful thermographic experimental investigations the best compromise was found by the selection of the standard aircraft paint. White coating is the preferred option because of its increased sensitivity, since the paint contains larger particles.

From an aerodynamic point of view the question was raised on the possible effects of the paint on the flow development over wings and the resulting changes in aerodynamic characteristics. A comparison of drag for an uncoated versus a fully painted model is presented in figure 9.

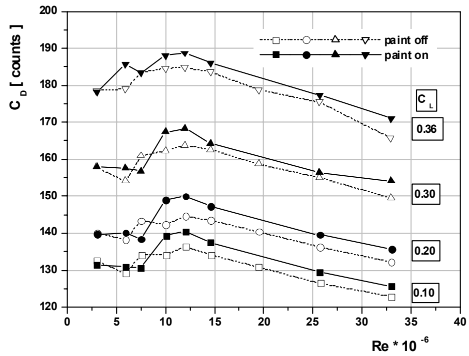


Figure 9 : Effect of full coating on drag

The proven excellent data repeatability combined with a perfect repeat of flow conditions allows to identify even the smallest differences in drag. Here, a constant shift in C_D of about 3 counts was recorded as an effect of model coating over the Reynolds number range which is dominated by the presence of turbulent flow. The relevant transition location was simultaneously monitored by IR-imaging.

As already reported [5], some cases have been experienced where disturbances to aerodynamic data quality were caused by surface imperfections mainly in the leading edge area of wings. In this context, the question on the requirements on surface quality was raised , as finishing contributes directly to total model manufacturing costs.

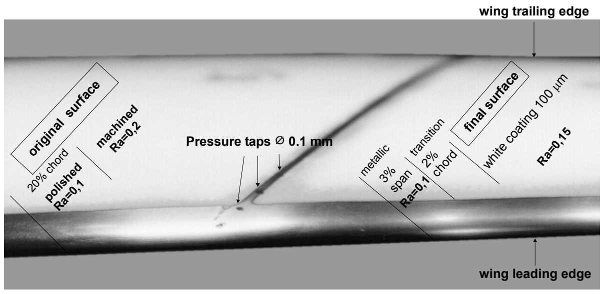


Figure 10 : Surface quality requirements

Nowadays it is felt that an adequate solution is given by the status presented in figure 10. The model wings can be machined providing a surface quality of Ra = 0.2. Manual polishing to improve the surface finish to Ra=0.1 is only applied over the first 20% of wing chord. Preparing the model for IR-imaging or MDMS measurements, 100µm thick white coating is sprayed onto the main part of the wing . Subsequent polishing keeps the upstream 3% chord in its original metallic condition, whilst a roughness of Ra = 0.15 can be achieved on the remaining surface. As a consequence, the leading edge area is more tolerant against particle impacts providing a major improvement in the confidence for aerodynamic data quality and boundary layer behaviour.

INSTRUMENTATION

Angle of incidence measurement

One of the most essential quantities to be measured in aerodynamic testing refers to the incidence of the model. ETW have developed a concept, based on Q-flex incidence measurement, which is able to fulfil the challenging demands of accuracy, resolution and operational reliability. This can only be achieved by an optimised design of a suitable inclinometer housing, protecting the instrument, which has to be operated at ambient temperatures, from the surrounding conditions. Here, special care had to be addressed to suppress any heat transfer from heated joints and internal structures on one hand as well as the conduction of cold from the model parts exposed to the gas stream. Also, positioning effects of the inclinometer housing due to inhomogeneity of structural temperature inside the model have been taken into account.

The present ETW standard offers the specified accuracy in incidence measurements of 0.01 degree together with a resolution of 0.001 degree.

Model vibrations

The test envelope of wind tunnel models in pressurised facilities is mostly restricted by load limits due to increased dynamic pressure levels. Further operational constraints may be caused by excessive vibrations of the model, generated by the supporting system or the test object itself. The latter are often based on amplified eigenfrequencies, if another oscillating element has a critical exited frequency nearby.

To improve model safety in pitch–pause and continuous traverse mode with a simultaneous gain in aerodynamic data quality, an **A**ctive **D**amping **S**ystem (ADS) has been design and commissioned at ETW.

The concept of the ADS is based on the consideration of the balance / model assembly as a vibratory spring / mass system with very low damping properties. Originated by a broad band excitation, the model may start oscillating as a rigid body with its eigenfrequencies, sourced by the model mass and its distribution and the balance stiffness for the corresponding degrees of freedom. Generated amplifications can be exploited for the attenuation of the vibrations by counteracting the oscillations in a controlled way by an excitation produced by piezo elements.

The final set up of the complete system is installed between the flanges of the six-component strain gauge balance and the sting, in the support line of the wind tunnel. The load capacity of the 14 piezoceramics is identical to the full load range of the 26kN balance, mostly used for transport aircraft testing. Simultaneous excitation of all piezo elements allow for an attenuation of vibrations in 5 degrees of freedom (without roll).

Recently performed commissioning trials have demonstrated the extraordinary capability of the device as given in figure 11. Running in continuous pitch without ADS (classical configuration), encountered vertical accelerations on the model exceeding gravity levels of 1g leading to a cancellation of the polar to protect the model and the balance. With the ADS system activated the vibration level could significantly be reduced allowing an extension of the pitch range.

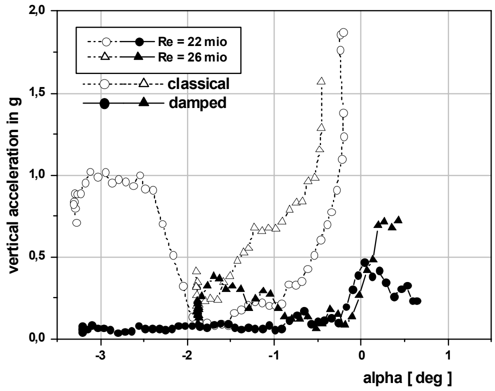


Figure 11 : Vertical acceleration with and without ADS

The beneficial influence of the ADS system on aerodynamic coefficients can be seen in figure 12, presenting the lift coefficient CL versus model incidence. With reference to figure 11, significant distortions of the data can be observed for vibrations exceeding levels of about 0.5g. In these areas, the ETW correction method for alpha had formerly to be applied.

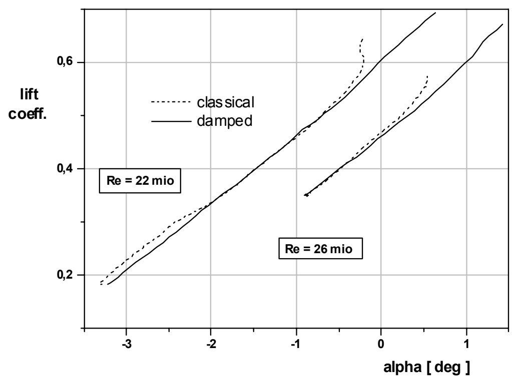


Figure 12: Effect of ADS on lift coefficient

Balance temperature stability

For full model testing the ETW specification demands a temperature gradient of less than 1K across the balance to provide the required data quality. In practice, this is achieved by an optimised tunnel/model conditioning procedure applied to the cold balances. While the final outcome is satisfactory, in terms of data quality there is a significant potential for improvement in terms of cost and time saving. Relevant activities and experimental investigations with an active balance conditioning system using gaseous nitrogen are progressing and initial results quite promising.

In opposite, the half model balance operates within a thermally conditioned housing. A powerful heating system arranged as eight individual circuits with individual closed loop controllers guarantee a homogeneous thermal environment around the balance across the full tunnel operating envelope. The achieved combined system of thermal protection with multiple heating systems turned out to be extremely efficient, allowing to establish a tight tolerance on temperature control. An impressive balance temperature stability down to 0.1K has been demonstrated as shown in figure 13.

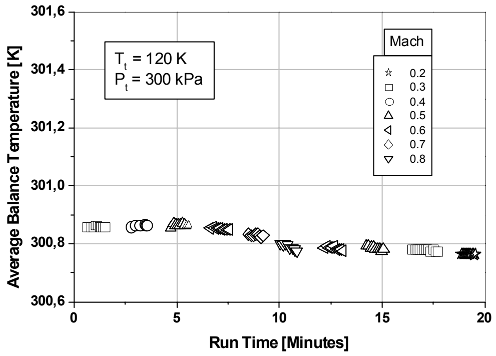


Figure 13 : Thermal stability of the half model balance

The figure reflects the average balance temperature at cryogenic conditions for a series of consecutive polars covering Mach numbers between 0.2 and 0.8.

Stable balance conditions form an essential prerequisite for the later generation of accurate and repeatable aerodynamic data but operational aspects and wind tunnel controllability also provide a major contribution. Some information on repeatability is available from the first trials with the new half model concept established in ETW. It has to be pointed out that the load capacity of the ½ model balance is adapted to transonic speed requirements. Consequently, when operating at low Mach numbers and at reduced levels of dynamic pressure, it will become more challenging to meet the tight quality specifications. For medium term repeats deviations of ΔCL = +/-0.002 in lift and Δ CM = +/-0.0005 in pitching moment can be quoted.

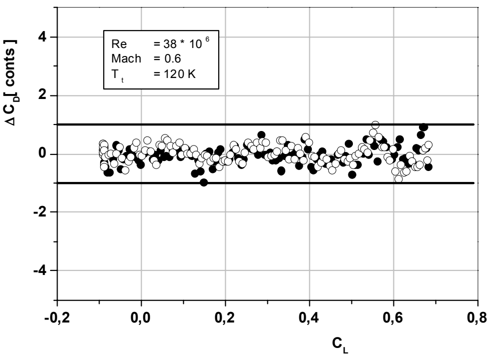


Figure 14 : drag repeatability for half models

The achieved excellent short term repeatability in drag is shown for a cryogenic test case in figure 14. Individual deviations from an average over all selected polars confirm that the standard deviation in drag was well within 1 drag count. The presented data are partly corresponding to results shown in figure 13 documenting the close link between the stability of balance temperature and resulting data quality.

Model Deformation Measurement System

ETWs capability to vary dynamic pressure at constant Reynolds and Mach number offers a superb tool to analyse aeroelastic effects. Normally, the model design will include a built-in wing deformation, corresponding to the estimated deformation at a particular lift. Consequently, variations of tunnel pressure will deform the wing even with the use of high stiffness alloys. To satisfy an increasing demand on accurate monitoring of wing twist and bending during wind-on, and to improve the level of information about true model geometry for CFD input, ETW has procured a **M**odel **D**eformation **M**easurement **S**ystem (MDMS) based on Moiré techniques. By comparison of the patterns of the loaded wing with the unloaded wing at the same model incidence, a quantitative determination of the degree of twist and bending can be provided. The system is mounted inside the structure of the test section top wall and presently only operational for full models. Further technical details can be taken from reference[6].

Targeting for accuracies of better than 0.5mm and 0.2 degree respectively, which caused a system calibration will be performed a prior to testing. Figure 15 shows the result of a pure wing bending generated by a defined static loading on the wing tip, a total deflection of –3 mm. Assuming the wing root section to be considered as the stiff part of the model, the evaluation of the system data is in nearly perfect agreement with the measured deflection. Applying a polynomial fit, noise on the data in the order of +/- 0.1 mm can be found.

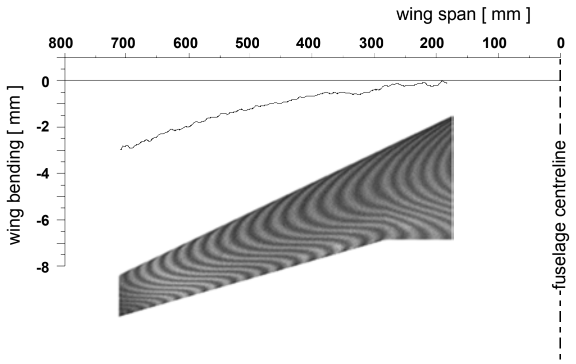


Figure 15: Wing bending evaluated by the MDMS system

The benefit of the availability of a suitable system for assessing the model deformation during the test can be impressively demonstrated by some results given in figure 16. To increase transparency of the effects, the curves for Reynolds numbers of 7.6 and 14.8 $*10^6$ have been shifted laterally by 0.01.

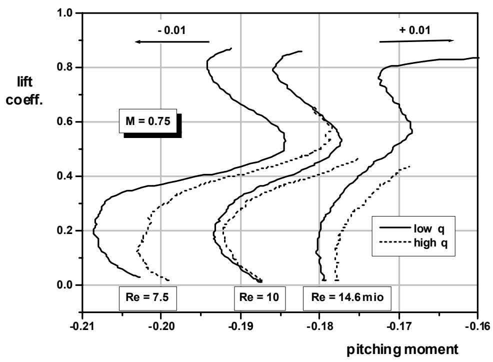


Figure 16 : Aeroelastic effects on pitching moment

The presented data have been taken for natural transition.
It is to be noticed that duplicating the dynamic pressure by a factor of about two will move the pitching moment at identical lift to less negative values. The individual variation of the $C_L = f(C_M)$ curves depends on the character of the boundary layer showing the largest changes in areas with a major contribution of laminar flow.

Transition detection

When testing an aircraft model in a wind tunnel, the aerodynamicists major interest may be on gaining the aerodynamic coefficients with a high level of data quality and reliability. To improve the understanding of the results, additional information may be requested on flow development and behaviour. In this respect any knowledge about the characteristic of the boundary layer is of common interest, either to investigate Reynolds number effects or to ensure that for a given test condition the flow status is fully turbulent everywhere on the wing.

For such purpose ETW has selected the infrared imaging technique. This allows remote sensing of infrared radiation, emitted by the temperature patterns on the model surface. During the test, a step change in temperature is induced which causes different heat transfer rates to occur in the laminar and turbulent boundary layer for example.

Figure 17 : Shock position on a wing

Presently, two systems are owned : The standard AGEMA system which is operable down to about 210K and the so called CRYSTAL system, a complex unit which requires cooling by liquid Helium and a vacuum insulated housing. CRYSTAL can be operated down to 100K covering the flight Reynolds condition of most of the models.

For full model testing both cameras are installed in the top wall structure of a model cart. Testing half models, the AGEMA unit in its insulated and heated housing can be also become integrated in the side wall of the test section, to view the surface of a half model in its vertical alignment after its attachment to the ceiling.

As the IR-imaging technique is sensitive to infrared radiation, effects other than boundary layer transition can be visualised. In figure 17 the spanwise location of the shock position is indicated by a white shaped band running from wing root to tip. The wing flow is fully turbulent. Presently, the origin of the phenomenon is not fully understood. It might be caused by a shock induced separation bubble, modified heat transfer in the shock region or a mixture of both. In any case the observed feature turned out to be accurately repeatable. The visualised shock location on the wing reflects the expected dependency on aerodynamic and model characteristics.

TYPICAL REYNOLDS NUMBER EFFECTS

As stated above, direct Reynolds number effects such as skin friction follow defined scaling rules and can more easily be predicted. In this context the relation lift-pitching moment was declared as an indirect Re-number effect.

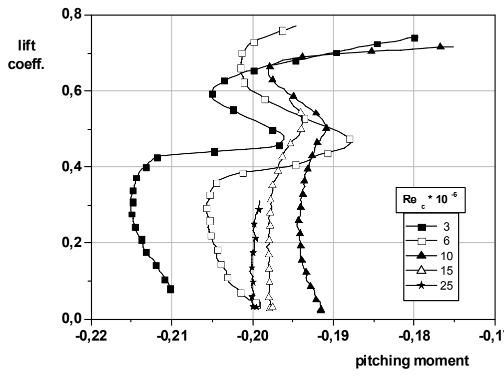


Figure 18 : Reynolds number effect on pitching moment

For the case presented in figure 18 natural transition is applied to the wing at design Mach number. In areas of dominating laminar flow, i.e. low Re and lift coefficients below 0.5, an increase of the Reynolds number from 3 to 10 million leads to a reduction in the pitching moment by about 0.02. A further increase of Re causes an opposite trend due to a downstream movement of the shock in a mainly turbulent flow environment.

Beside testing transport aircraft models, investigations have also been carried out on fighter models. For such configurations there is a demand on testing at higher angles of attack often combined with increased transonic speed levels. An essential indicator for flow separation on the upper wing surface can be found in the behaviour of the trailing edge pressures.

Relevant pressure coefficients referring to the static pressure at the centre point of the model are shown for three spanwise sections of a swept wing in figure 19. Selected cases representing natural boundary layer transition for Reynolds numbers up to flight conditions, as well as an artificial tripping are plotted as a function of model incidence.

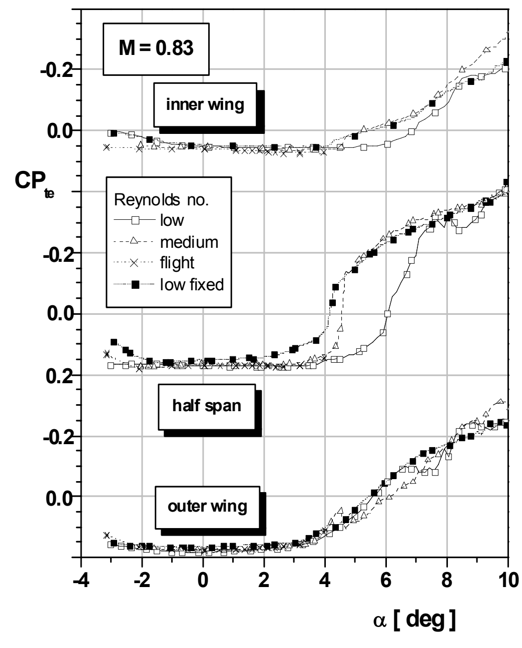


Figure 19 : Reynolds number effects on trailing edge pressures

While the flow on the inner and outer wing seems to be independent to variations of Re, an increased sensitivity can be found around the mid span area. Up to an angle of attack of about 4.5 degrees, all transition free cases match nearly perfect. The occurance of the steep drop in the trailing edge pressure strongly depends on the actual Reynolds number. Transition fixing by boundary layer tripping to simulate flight Reynolds number behaviour in the low Re environment, effects CP_{te} already at a model pitch of 2 degrees, indicating an earlier separation onset of the flow.

Since the end of 1999 ETW has been in a position to offer a half model test capability to clients. A Mach number range from 0.15 up to about sonic conditions can be covered. As high lift model configurations to be tested at low speeds present an area of particular interest, relevant validation tests with a representative model have been performed at an early stage. Figure 20 impressively documents the superior test range of the facility in comparison with conventional tunnels.

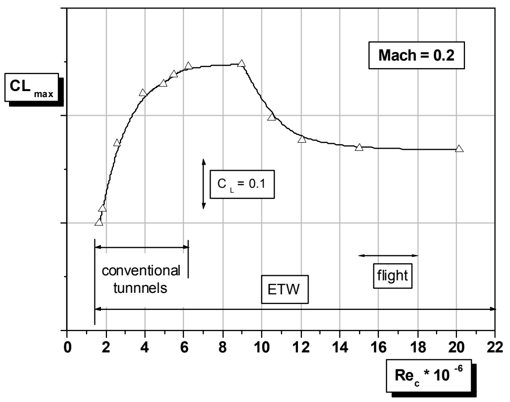


Figure 20 : Reynolds number effect on maximum lift for a half model configuration

Classical scaling rules applied to results obtained in conventional tunnels may not accurately predict flight status as shown for the maximum lift behaviour. Challenging demands on the quality of pressure measurements have to be fulfilled for these type of testing, to ensure the accuracy of the dynamic head which is used to normalise the aerodynamic coefficients. Due to the low absolute Q-levels, small deviations in the static reference pressure can easily generate errors of a few lift counts.

PRODUCTIVITY ASPECTS

From a commercial point of view the target is always the minimisation of testing time. Operating in a cryogenic environment obviously requires more time than ambient investigations due to the limited accessibility to the model and the periods devoted to thermal conditioning of equipment to provide best data quality and high confidence in the aerodynamic results. Hence, the specific test requirements , such as configuration changes of the model, dominate a drafted test-scenario which is subsequently optimised time- and costwise by ETW in agreement with the client.

A representative sample is given by figure 21 for a full model entry, asking for 125 polars at temperatures ranging from ambient down to 140K, and including an removal and later re-attachment of a transition fixing. Working in two shifts from 8 a.m. to 8 p.m., the programme was completed in 4 days without major problems. The presented sequence was started with a balance replacement performed in about nine hours.

To reduce the indicated periods for cool-downs, ETW started operating an independent active balance conditioning system as mentioned above.

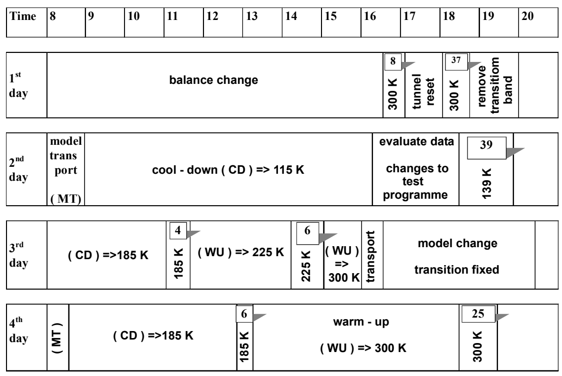


Figure 21 : Typical scenario for full model testing

In half model testing, the temperature controlled ‘warm’ balance is not affected by any time consuming balance conditioning. Consequently, moderate test scenarios covering the full temperature envelope can be performed within one day. Figure 22 documents the setting of tunnel parameters required to complete a typical test programme with a half model. The Mach number ranged from 0.17 to 0.3. For such tests, the maximum cool-down rate of 80 K/hour can be applied. If a pressurisation of the tunnel will be required (e.g. to investigate aeroelastic effects) , the experience gained recommends to run the polars at maximum requested tunnel pressure at first and then to proceed to lower levels in a descending order for time saving reasons. The acceptable pressurisation rate may be dictated by stress limitations on the models which may be severe for high lift configurations.

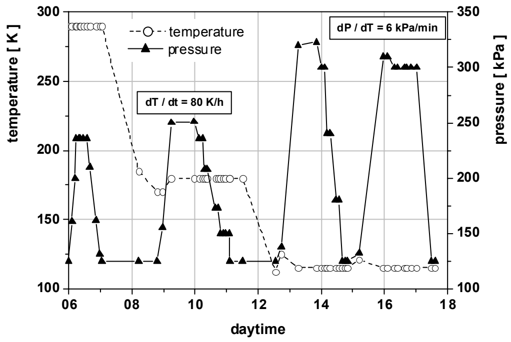


Figure 22 : Low speed test scenario for a half model

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