METHOD OF ANALYSING DATA ON A SWEPT WING AIRCRAFT IN FLIGHT

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Abstract

Over a number of years the Aeronautical Research Institute of Sweden, FFA, has operated a swept wing aircraft for aerodynamic investigations. The results from more than 120 flights have been organized in a computerized database where information may be obtained for a number of different flow problems. In the present paper an account is given of the database, its characteristics as well as examples of how it may be used to describe the flow on a swept wing at subsonic Mach numbers.

Background

Comprehensive and accurate experiments are mandatory for development of today's computational codes, especially in connection with the classical problem of extrapolating wind tunnel results to flight. However, in most cases neither wind tunnel tests nor flight evaluations are sufficiently detailed to serve as test cases. At best the measured local properties at selected flight or tunnel conditions allow an interpretation together with a combination of computational techniques; i.e., panel methods etc.[1], to help obtaining smooth distributions of pressures and other flow parameters. Often it is also necessary to use comparisons with other computational techniques to evaluate new codes.

Although advanced flight tests have been performed to explore aerodynamic data for more than fifty years, there is still a common assumption that the flight data is too much affected by environment to represent a truly valid set of data. However, in particular during the last decade, a series of tests have been performed utilizing modern data acquisition techniques along with a variety of sensors, demonstrating a high degree of accuracy and repeatability. Flight mechanics and avionics have been pacing subjects in the development of the digital techniques, and flightworthy instrumentation has developed quickly[2]. To make use of these techniques in flight test centers is fairly common.

However, it has not been common to utilize these possibilities for aerodynamic investigations. One reason is the inherent high cost of obtaining aerodynamic information in flight. In general wind tunnel tests are more cost effective, need shorter lead time and are easier to repeat. Thus the role of comprehensive aerodynamic investigations would be mainly to support computational code development or prove new concepts in real environment.

Also, while flight experiments previously involved developing exploratory prototype aircraft to prove new concepts or investigate certain features of flight, this approach has been virtually unused for the past two decades. Two important exceptions are the aircrafts Himat and X-29[3,4]. The first was utilized to evaluate new fighter concepts and novel structural materials, and the second as a technology demonstrator for the forward swept wing idea. Thus, with very few exceptions, wind tunnel tests have been the only source of experimental data to support code development[5].

Twenty years ago code development for aircraft design meant little more than finding simple correlation techniques for the overall flight parameters of an aircraft. Viscous flow might at best be computed with an integral method, and estimates of transition location were very uncertain. Today three-dimensional finite-difference codes for the boundary layer computations are common and transition is estimated with the help of stability codes. Code development today involves modelling Navier-Stokes equations or a subset of these. This involves a completely different requirement on the supporting experiments as well as in the handling and analysis of the data. Whereas the bulk of the information found in[5] concerns measured pressure distributions, time-averaged boundary layer profiles are required and extensive turbulence measurements are desirable for new databases being made.

When considering the features of three-dimensional flow, the amount of data required for proper handling and presentation of essential results exceeds by several orders of magnitude the written-report format. Hence it becomes necessary to extract only the most important flow features in normal report form, and let the remainder of the information be available only as a computerized database.

In the present paper an account will be given of an experiment where this data handling problem was treated, and also some illustrations of what may be obtained through proper use of a database system will be discussed.

Experiment

Over a number of years The Aeronautical Research Institute of Sweden, FFA, has operated a laboratory aircraft for documentation of flow properties on swept wings in flight through the entire subsonic regime. The purpose of the project was to:

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- Investigate Reynolds number extrapolation procedure.

- Generate experimental data for code development.

The first of these is clearly the most limited, as it in reality means examination and improvement of existing rules-of-thumb. Support of code development is far more demanding, as was outlined in the previous section. It is not difficult to foresee how future requirements will include questions concerning non-steady flight conditions and the interaction with flight mechanics and structures that is the case for example during a demanding manoeuver. In the present experiment an effort has been made to support this variety of interests. However, as is natural, time-averaged (and to some extent turbulence) data obtained under stationary conditions are analyzed initially, and although data has been obtained in a variety of manoeuvers, only the stationary conditions have been included in the database discussed here.

Figure 1 shows the aircraft, a SAAB 32 Lansen attack aircraft, and the flight envelope with conditions repeated every flight. The figure shows one of the problems associated with generation of aerodynamic flight test data: angle of attack, Reynolds number and Mach number are always coupled. In an atmospheric tunnel there is a coupling between Mach- and Reynolds numbers, but at least the angle of attack is a free parameter; in a pressurized tunnel all three are independent. The solution chosen for "decoupling" in the present experiment is to fly at different altitudes and also, in some repeat conditions, have a different gross weight. Bunts and turns have been used, but are not considered sufficiently stationary or well defined for the present purposes.

Data is gathered partly with fixed sensors, partly with movable sensors where a number of flights are performed with sensors shifted to different patterns in order to generate a distribution. The wing profile is "classical", NACA 64A010 normal to the 25% chord line, but the wing exhibits most of the flow problems that can be observed even on very modern profiles – be it at other flight conditions. The results may be used to examine several different aspects of flow, such as leading edge flow, transition, turbulent separation and shock/boundary layer interaction etc. In addition, various add-on experiments have been performed: use of large-eddy breakup devices for turbulent manipulation, passive shock control, transition on a 10 deg. cone, as well as a variety of dynamic flow cases.

In addition to a test case defined for[5], containing only static pressure distributions on the main part of the aircraft, description of the tests and accounts of some of the experiments performed on the aircraft, can be found in[6-8]. The main interest so far has been the examination of laminar/transitional/turbulent attachment flow, as well as the general flow found in the leading edge region.

The measurements included in the present database are:

- static pressure distributions (more than 1500 pressure taps),

- local skin friction from impact probes as well as heated film gages,

- time-averaged boundary layer data from pressure rakes,

- turbulence characteristics in the boundary layers using hot wires and split films,

- pressure fluctuation measurements.

Figure 2 shows a schematic of the data acquisition system and indicates the types of sensors used and how the recording of aerodynamic as well as reference parameters is performed. The data may be split into two parts depending on the frequency content. The "slowly" varying parameters are recorded using a 40 Hz PCM system. This allows sufficient information to be gathered to take into account rigid-body motion and also, for example, the wing bending, control surface deflections, etc. The turbulence data as well as several structural modes require much higher frequency response. While turbulence production is generally occurring below 2-3 kHz, the dissipation is of interest up to 100 kHz or more; clearly this is not realistic to examine with present techniques in flight. Therefore, a selected number of parameters have been recorded with a frequency content up to 10 kHz. In these cases the signal-to-noise ratio has been improved through AC-amplification of the data, and FM-recording on tape.

The time-averaged results published in the AGARD test case collection[5], as well as the computerized database system discussed in this paper, concern only time-averaged data. The turbulence data has to be treated in an even more selective manner, reducing only a small fraction of the data available, and storeing only selected quantities like RMS-levels with the main database.

Database system

In the database system it is possible to select, compress and analyse data without prior knowledge of the structure of the tests. It is based on a pre-processing of the gathered data for check and compression, and the general structure of the data handling system is shown in Fig. 3.

As can be seen, the pre-processor reduces the amount of data almost two orders of magnitude while checking the data, keeping the important statistics and giving the data an easier accessible structure.

A large variety of database systems are available for different computers. The most efficient are relational, working with binary files. However, it turns out[9] that few systems capable of handling the present type of engineering database are available. Also, the general systems require a considerable amount of space, and are not generally portable from one computer system to another. Thus, it was decided to design a specific hierarchical system for the present test, with the following characteristics.

- Portable

The database consists of ASCII files and FORTRAN 77 source code, with a minimum of non-standard features in the code. Thus curve-fitting and graphics are not included in the basic code, as it would have involved access to specific library routines. The database was developed on HP1000 and VAX computers in Sweden, and is also implemented on the same type of computers at NASA Langley.

- Interactive and menu-driven

Due to the complexity of the experiment, it was suitable to generate a large number of codes for various types of analysis. All are related, in the sense that they all use exactly the same files as input, and one master program exists capable of doing all different tasks. However, it is very slow, and the specialized codes are reasonably fast to work with an interactive mode. As the codes are menu-driven, it is not necessary for the operator to know or keep in mind details concerning the experiment to be able to work with the database.

- Source code

Due to the variable environment it is essential to have the source code available. Memory constraints, difference in computer library routines, graphics, or the introduction of different aerodynamic parameters or correlations may make changes in the source code advantageous.

- Geometry defined

The geometry of the aircraft has been described in detail, as this is a very important issue to be able to use the experimental data together with a computational code. The aircraft geometry has been digitized for computations with panel methods at NASA Langley, and this information is included in the database to facilitate comparisons with the experimental data. Figure 4 illustrates the geometry obtained and the type of panelling used. As illustrated in[10], it is important to have a reasonably well defined geometry for the entire aircraft, although the layout of panels, grids, etc. depends on the specific computational codes in mind. This is in line with the trend in several recent textbooks[11,12], where the traditional material is supplemented by listings of computer codes that can easily be entered into the reader's own computer for computation of pressure distributions, boundary layer characteristics, etc.

The low-frequency information gathered in the tests is recorded using a 32 parameter, 12 bit PCM (pulse-code- modulated) signal with a sampling rate of 40 Hz (filtered at 10 Hz to fulfill the Nyquist theorem). This allows all rigid-body motion of the aircraft to be monitored, and generally yields sufficiently high frequencies for stability derivatives to be determined. As the total flight time is roughly 1 hour, it means approximately 8 Mb of data each flight, 1000 Mb total for the entire test of more than 120 flights. Clearly several parameters varies more slowly than the 10 Hz. For example, the scanivalve were stepping at an interval of 0.4 or 0.8 seconds depending on pressure line length etc. (i.e. 16 or 32 PCM frames covered the change from one pressure port to the next and the stabilization of pressure). Figure 5 shows the type of verification used to check that the pressures stored in the database were really valid data. It was necessary to detect clogged or leaking tubes and too long pressure leads, as well as a variety of other factors that might jeopardize the quality of the data.

All PCM data was reduced through a statistics approach, where blocks of data were checked through determination of 4 values per exposure instead of utilizing the entire time-history. Average, minimum and maximum as well as root-mean square of deviations from the average were entered in the database. As Fig. 2 shows, the PCM information recorded, includes a complete rigid-body motion definition, control surface deflections, and information on flight speed, Mach number etc. This allows a computerized determination of the type of manoeuver when working with the database, and it has been frequently used to avoid use of invalid data. As this statistics is based on filtered PCM values chosen at 2.5 Hz rate (only 1/16 of the PCM frames have ben utilized), it is sufficient for a rigid-body description of aircraft movement, but fails to describe the influences of, for example, wing bending or violent manoeuvers on the data: here all PCM data has been utlized to generate statistics for every 0.4 seconds.

Figures 6 and 7 illustrate how the basic data has been used to generate a compressed file containing all esssential information obtained during an exposure. First data has been checked and statistics gathered concerning its variation of a certain information-typical time. Then the time-averaged values are stored together with the statistics information that during the use of the database may be utilized to accept or reject certain pieces of data.

Figure 6 shows a correlation of Mach number and altitude variation during an exposure. Depending on the type of flow problem under investigation, this variation may or may not cause the database user to reject the data taken during this exposure. For example, if one is examining shock/boundary layer interaction and is close to shock-induced separation, variations in Mach number may jeopardize the whole set of data. On the other hand, at lower speeds the variation is entirely negligible.

Figure 7 shows two channels of anemometry, with data from a rotating x-wire. As a time record, the first half is taken from an x-wire in the freestream. During the second half the x-wire is located in the viscous interaction part of the wing-body junction. It is rotated through 360 degrees and back at distinct 30 degree intervals. Here a correlation between the two signals clearly shows the two x-wires, and how the inner depicts a lateral and vertical flow component. This data may then be reduced further to yield the heat flux and through the calibration curve the actual velocity components. This is not discussed further here as it is not part of the database system itself.

The computer code STROM may be regarded as a pre-processor to the database, as it mainly creates the data files to be used by the database itself. Figure 8 shows the database with its aggregate of computer codes and files.

It consists of several types of files:

- MENU is the key data file, relating the file numbers in the experiment; i.e. the tape number, configuration number etc. to flight date, pilot information etc.

- STX-- is data files containing the actual measured information with redundancy checks and statistics.

- LS-- yields the time-of-day for various exposures of each flight.

- AUX-- yields the hookup of various sensors.

- CON-- yields the positions of the sensors.

- HOT-- yields characteristics of hot wire-, hot films- and other sensors, as well as required data on the electronics.

Samples of investigations

In the present section two examples will be given concerning the use of the database to extract data for two flow problems:

- shock/boundary layer interaction,
- transition on 10 degree-cone.

These have been chosen from a large number of possible problem areas where the database contains relevant information.

Shock/boundary layer interaction

Figure 9 shows the instrumentation used in the experiment, which took place in the outer part of the swept wing outside of the stall fence. The work has a double purpose. To some extent the idea is to document a high Reynolds number shock/boundary layer interaction without tunnel interference, but it was also considered important to explore the ideas under investigation elsewhere concerning passive shock control through surface perforation. To this end, the instrumentation used consists of:

- 85 static pressure taps distributed all around the wing section with a concentration close to the shock position. (This allows an integration to overall profile characteristics.) In addition, two rows of pressure taps were located inboard and outboard of the row to detect spanwise gradients and shock sweep angle.

- Pressure rakes to measure the boundary layer characteristics upstream of the shock as well as downstream as far back as possible.

- Stanton tubes (i.e. razor blades to obtain a measure of local skin friction).

- Modified Preston tubes[13-15], mounted both in the flight direction, and parallel to the leading edge.

Figure 10 shows how the shock develops with Mach number for two different flight altitudes. Here the static pressure distributions have been used, and the good resolution obtained across the shock is seen. Figure 11 shows how the shock movement can be reduced from the previous and similar figures. The measured pressure coefficient may be transformed into a distribution of local Mach number, allowing a judgement of the magnitude of the region with local supersonic flow.

One of the main parameters to determining shock strength is the Mach number normal to the shock itself, It may be of interest to explore the relation between shock strength and upstream boundary layer conditions, as well as downstream boundary layer. This has been done in Fig. 12; there the pressure coefficient might be replaced by the Mach number normal to the shock. The actual downstream development of conditions has been exemplified in Fig. 13 the inviscid distribution of Mach number normal to the shock has been computed with a two-dimensional full potential transonic code[16,17].

As the figure indicates, it is feasible to interpret the 2D-results in several ways depending on the local sweep; it influences both the comparable normal freestream Mach number and the angle of attack. In this experiment passive shock control was explored as a means of decreasing drag, see Ref. 18 for details.

Transition on a cone

A 10 degree slender cone was used for in-flight transition measurements. The purpose was to get a free-flight comparison to wind tunnel measurements carried out with the same cone[16-19], and thereby obtain a measure of the flow quality of the wind tunnels. Figure 14 shows the cone and its instrumentation *) Three test-cases, each with different instrumentation, was flown with the cone. The instrumentations were:

i) Twelve Kulite fast pressure transducers placed in a spiral row along the cone, 38 pressure taps for static pressure measurement and two boundary layer rakes for measurements of the boundary layer profiles near the end of the cone.

ii) Six dantec V-type miniature hot-film gauges for fluctuating skin-friction measurements.

iii) Three of the hot-film gauges together with four pieces of piezo-electric film and a hot wire rake with one X-wire probe for freestream turbulence measurements.

A thermocouple was used for all cases to obtain the surface temperature of the cone. The cone was extended with a cylinder and mounted on a blind Sidewinder missile. The missile with the cone-cylinder was placed under the starboard wing at the most outboard pylon station.

Transition location was determined from the peak in the RMS-value of the fluctuating signal from either of the sensors. Figure 15 shows a typical measurement of the fluctuating pressure along the cone.

*) The cone was borrowed from NLR in the Netherlands and the test was done with assistance of Mr. B. Rohne from NLR.

The mounting of the cone made it impossible to retain zero angle of attack over the whole flight regime. Due to the angle of attack – Reynolds number – Mach number coupling. Therefore a strong angle of attack dependence on the transition Reynolds number was found. Figure 16 shows the transition Reynolds number as function of, a) Unit Reynolds number (Re/m), b) angle of attack, and c) surface temperature variation, at Mach number = 0.5. As can be seen, there is almost a linear dependence on unit Reynolds number in Fig. 16a but this is simply a disquise of the same angle of attack dependence shown in Fig. 16b. The dependence on wall temperature is of small importance compared to the angle of attack effects as can be seen in Fig. 16c. Figure 17 shows transition Reynolds number against unit Reynolds number for the first flight test case. Here no correction due to angle of attack has been done. Some wind tunnel results has also been included in the fig. and this shows that transition Reynolds numbers obtained in the flight test are of the same order of magnitude as the wind tunnel results. The conclusion drawn from this is that although flight tests are free from wind tunnel walls interference and the turbulence level is smaller, transition is not improved very much. This is due to the fact that new factors like angle of attack and moving wall effect[20] appears that causes instabilities and transition of the boundary layer.

Conclusions

A flight experiment involving a variety of aerodynamic properties has been described, as well as the database presenting the results from the experiments.

The paper shows:

- That it is both essential and feasible to make a complicated experiment available to other scientists.
- It is at present not possible, or feasible, to utilize any commercially available database system, advanced as they may be, because:
  - they are non-portable
  - require involvement by the analyzer before any information can be extracted
- A flight experiment can produce data of the same accuracy and repeatability as a wind tunnel experiment, and may offer an easier scale for the instrumentation work.

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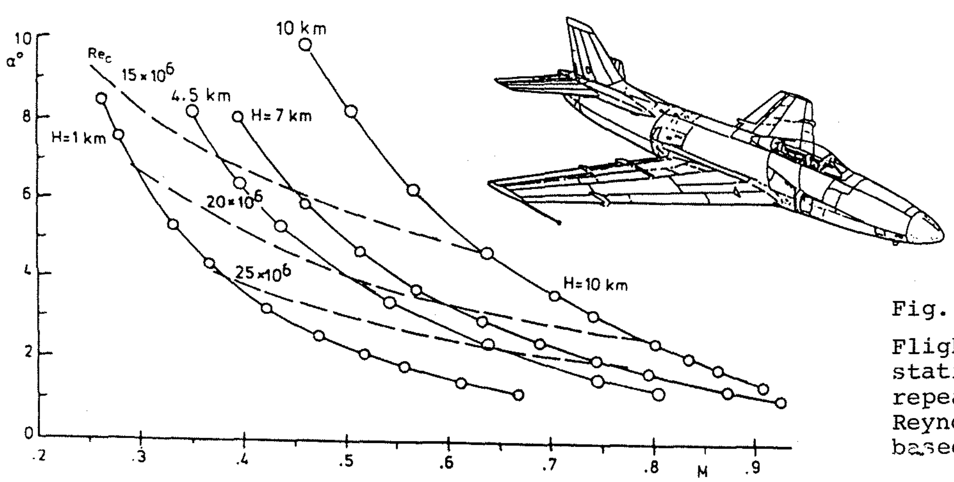


Fig. 1

Flight envelope with stationary conditions repeated each flight. Reynolds number is based on mean chord.

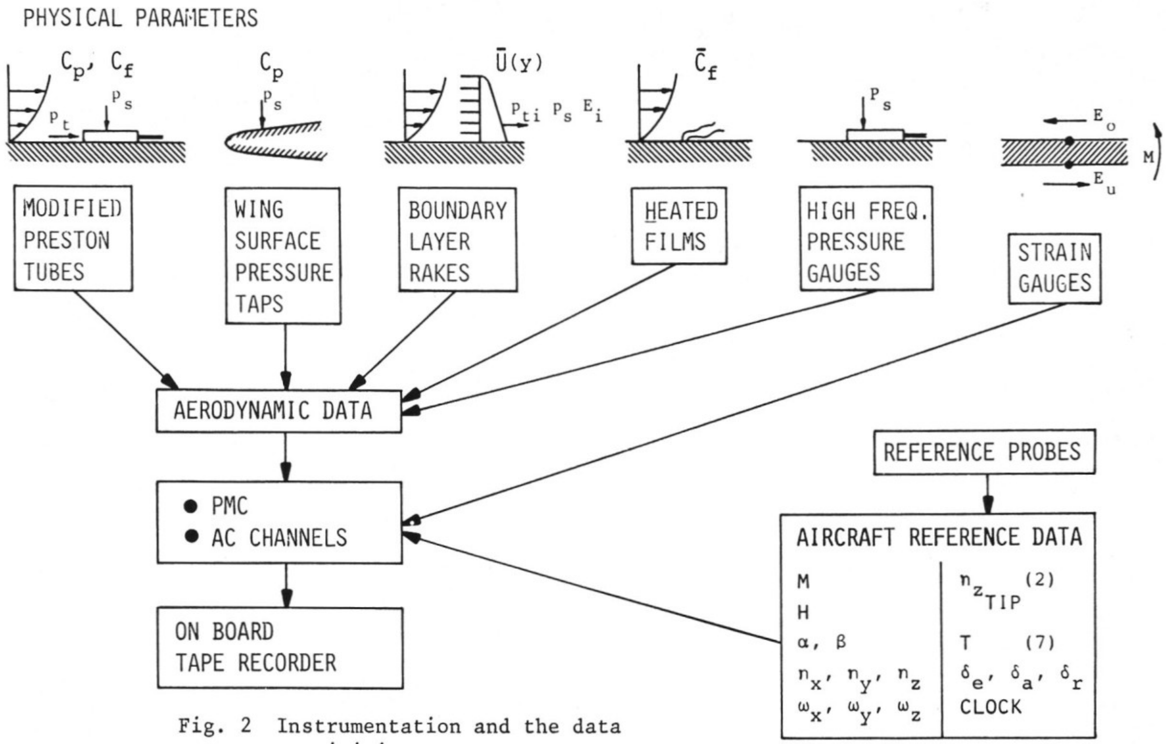


Fig. 2 Instrumentation and the data acquisition system.

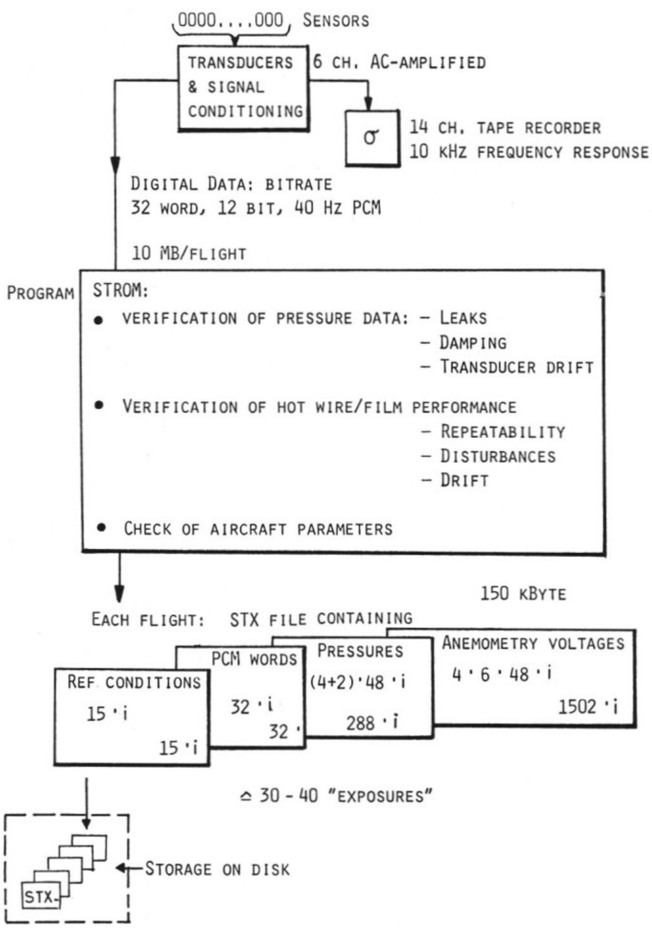


Fig. 3 General structure of the data handling system.

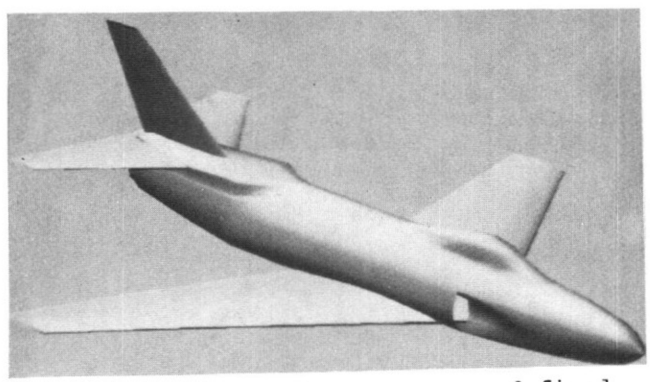


Fig. 4a Geometry of aircraft as defined for panel method computations.

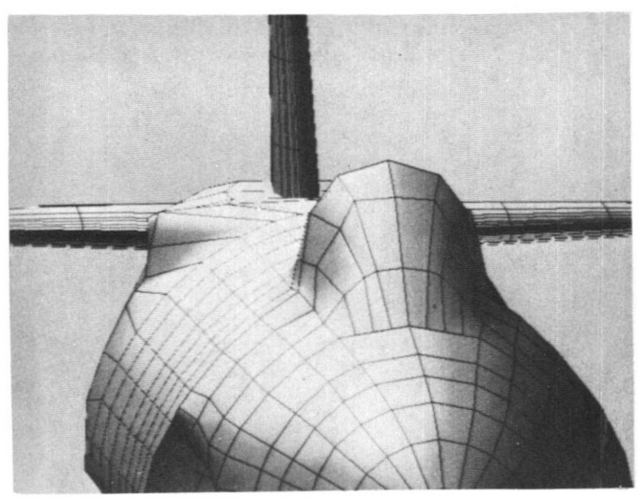


Fig. 4b Front view with panels.

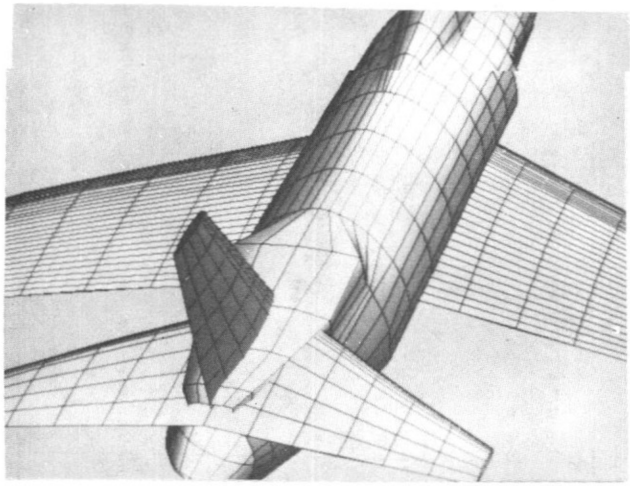


Fig. 4c Rear view with panels.

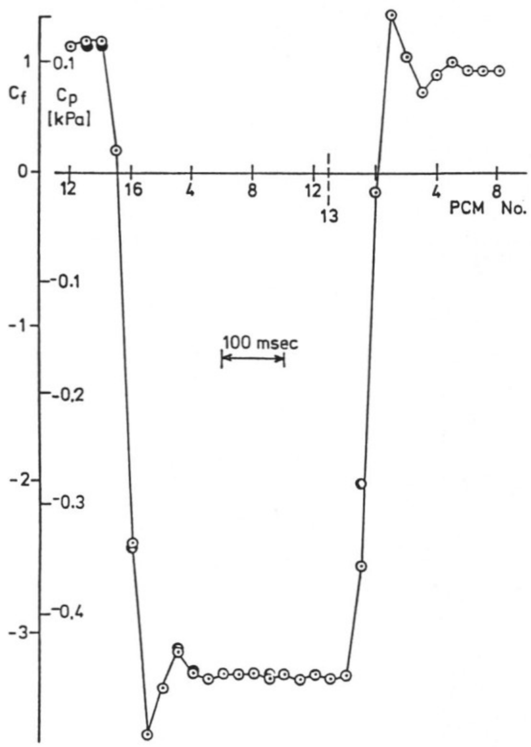


Fig. 5 Pressure as function of time according to PCM data.

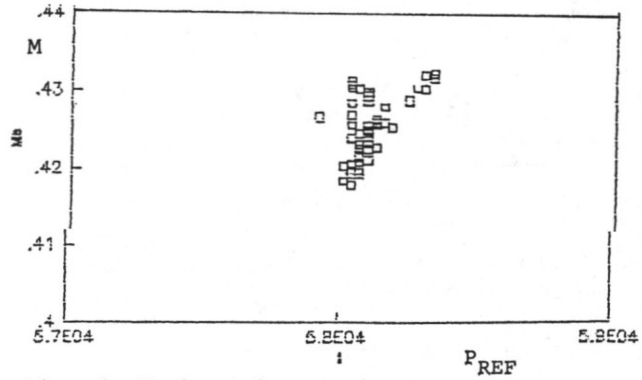


Fig. 6 Mach number versus pressure altitude in [Pa].

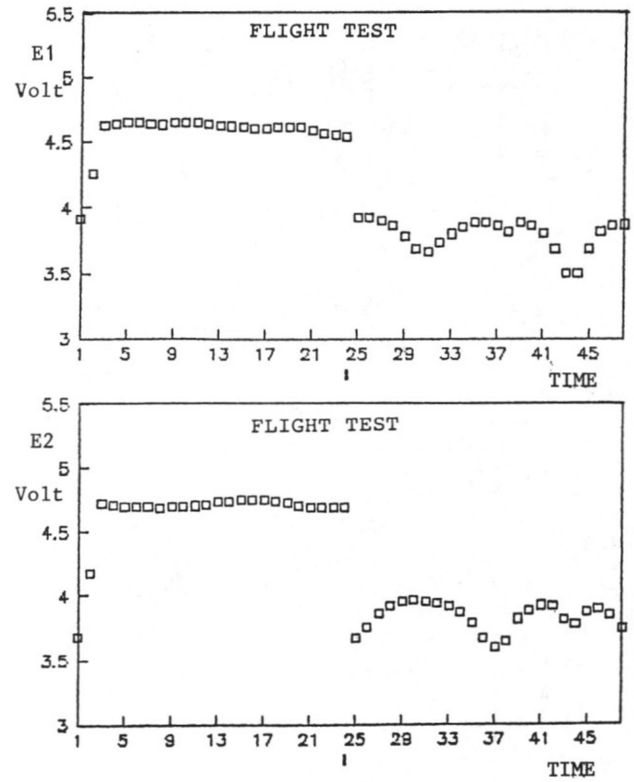


Fig. 7a X-wire signal at function of time.

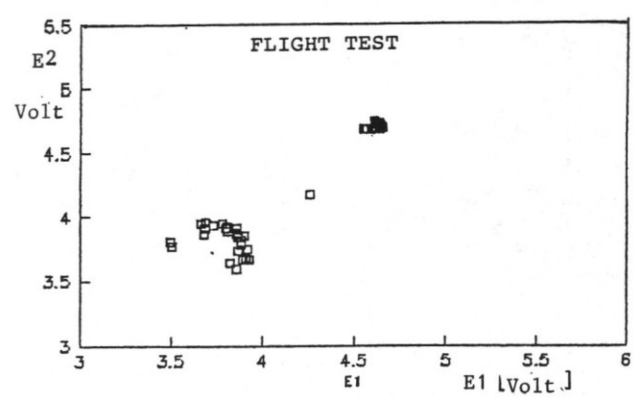


Fig. 7b Correlation of X-wire signal.

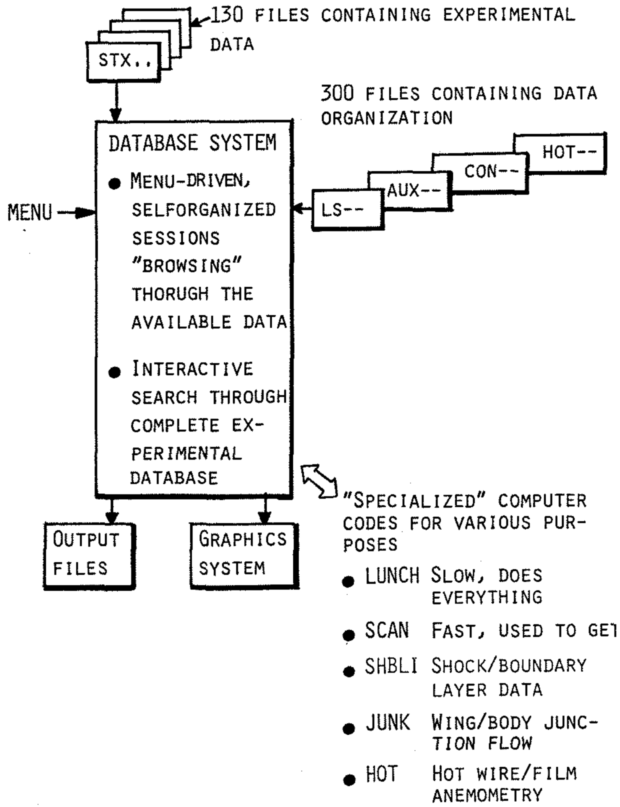


Fig. 8 Database system.

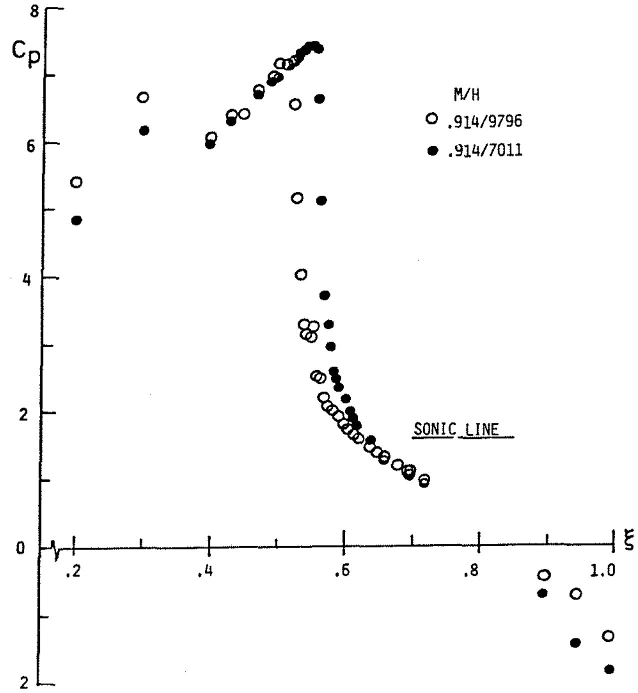


Fig. 10 Pressure distributions for M=0.914 at two altitudes *H* (7 and 10 km) illustrating shock movement.

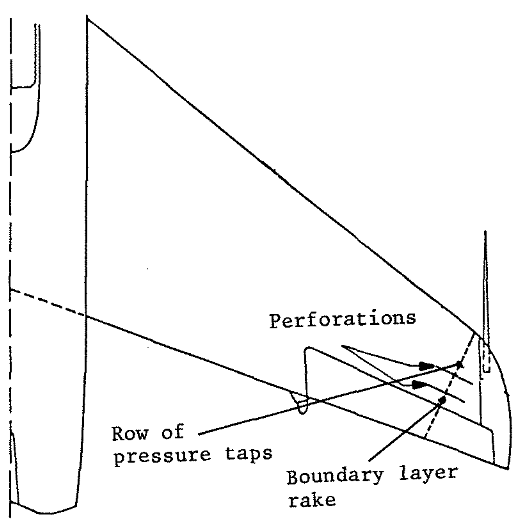


Fig. 9 Instrumentation for shock/boundary layer experiment.

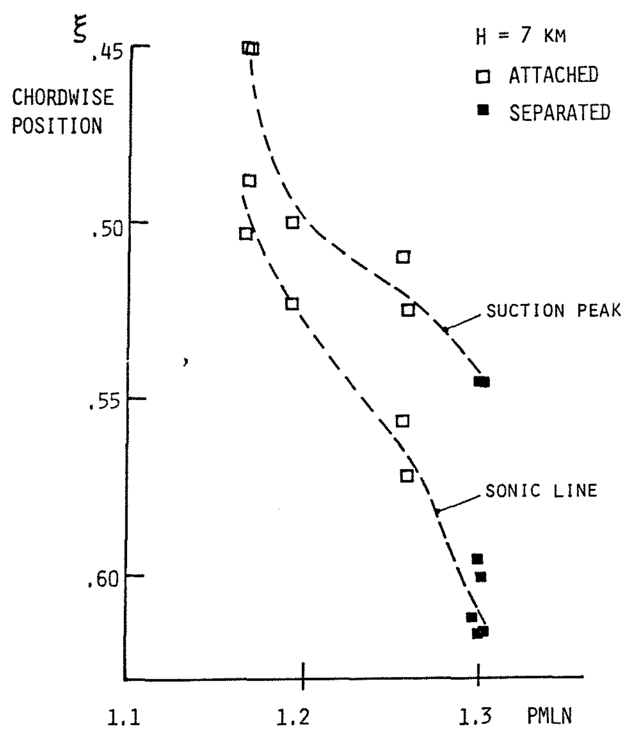


Fig. 11 Shock position versus peak normal Mach number, H = 7 km.

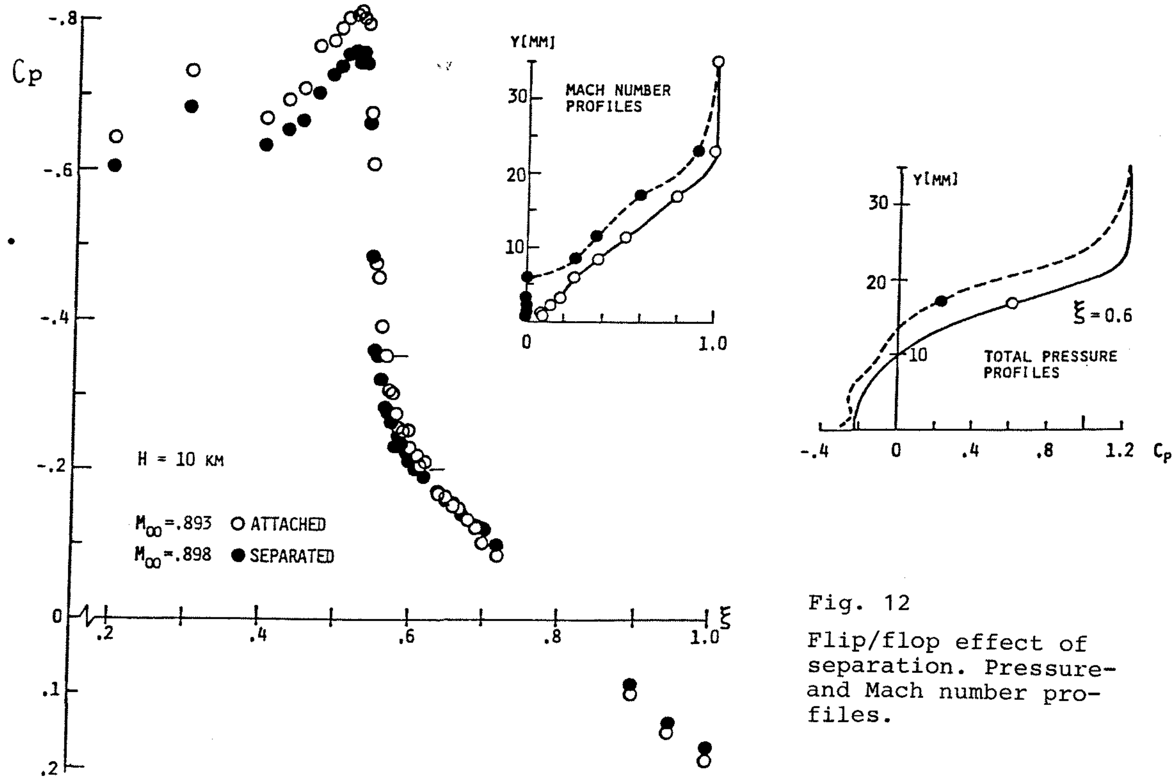


Fig. 12

Flip/flop effect of separation. Pressure- and Mach number profiles.

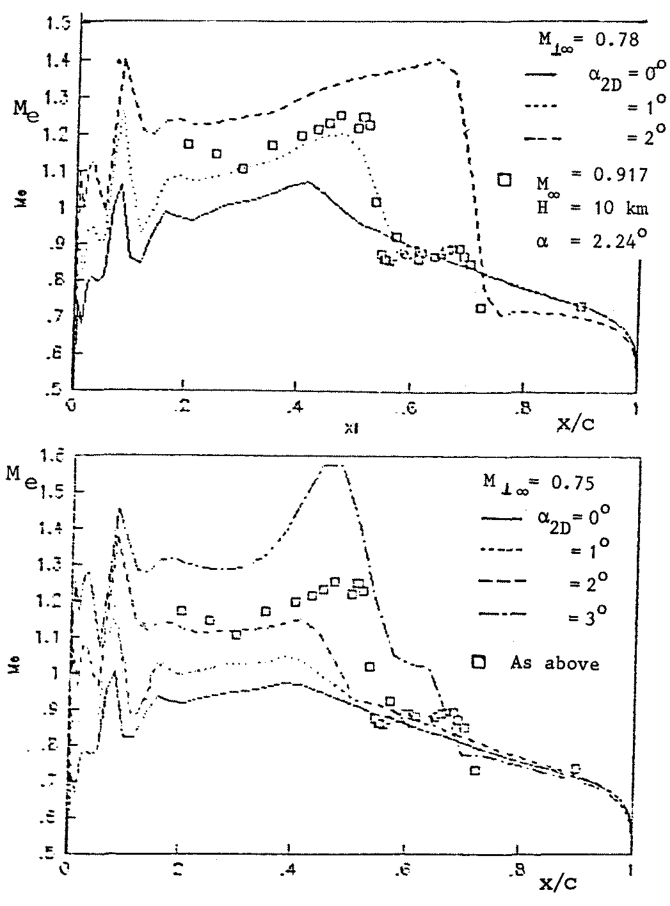


Fig. 13a Mach number distribution.

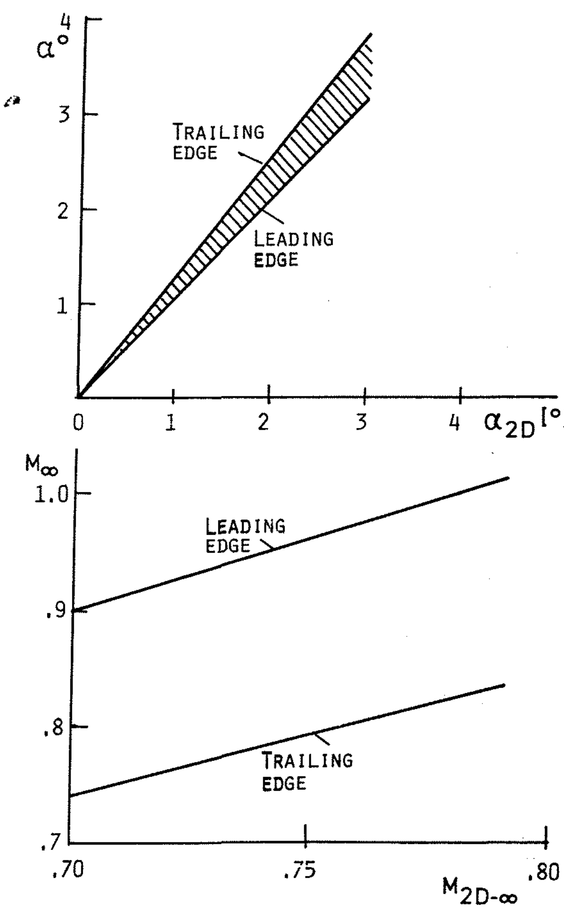


Fig. 13b Interpretation of two-dimensional computations for three-dimensional application.

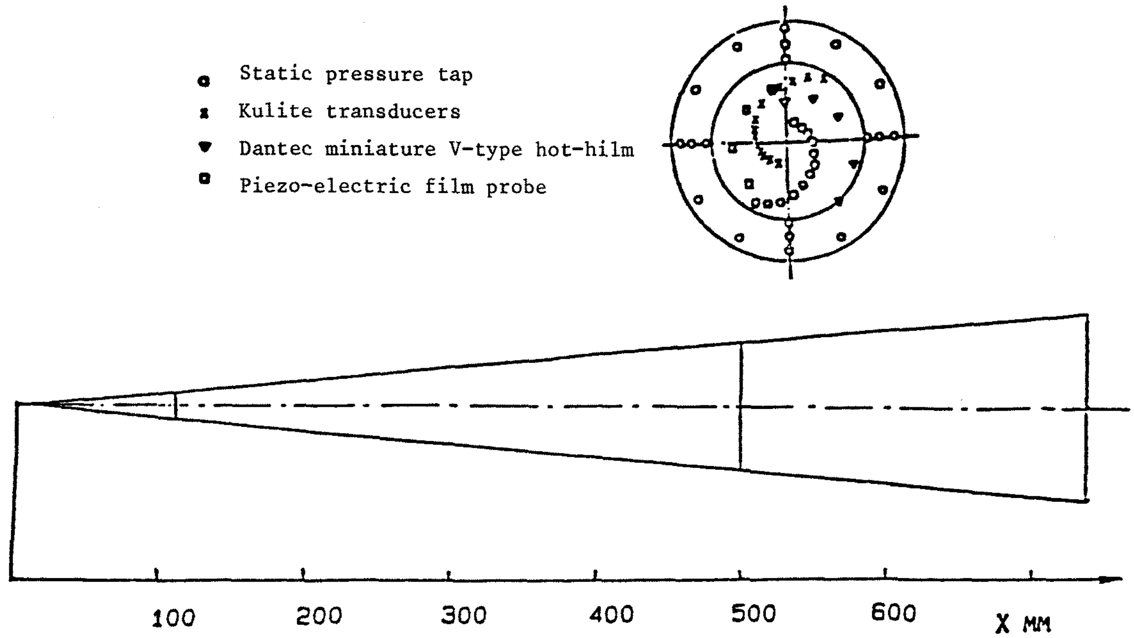


Fig. 14 The NLR Kulite cone and instrumentation.

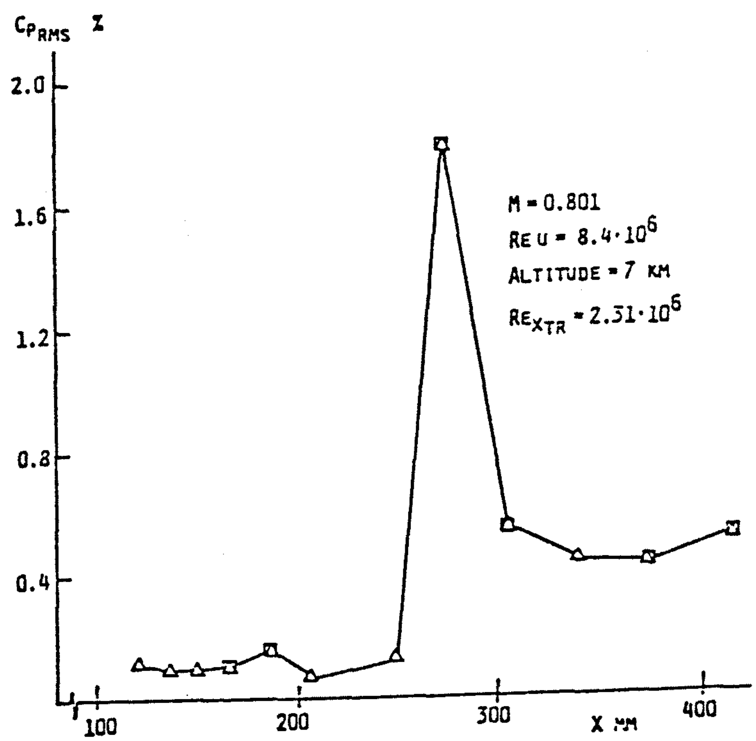
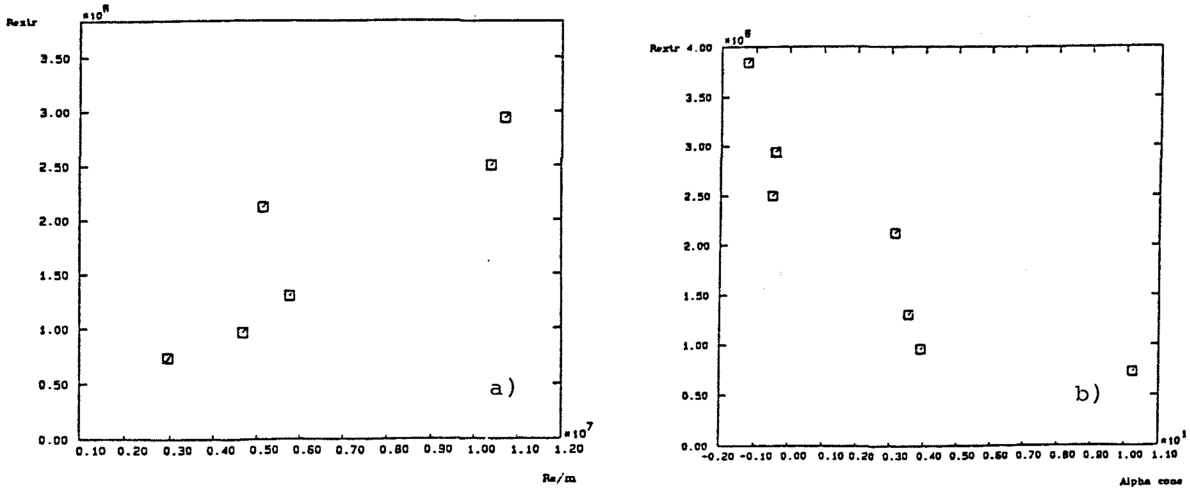


Fig. 15 $C_{p\ RMS}$ variation along 10^{o} cone.



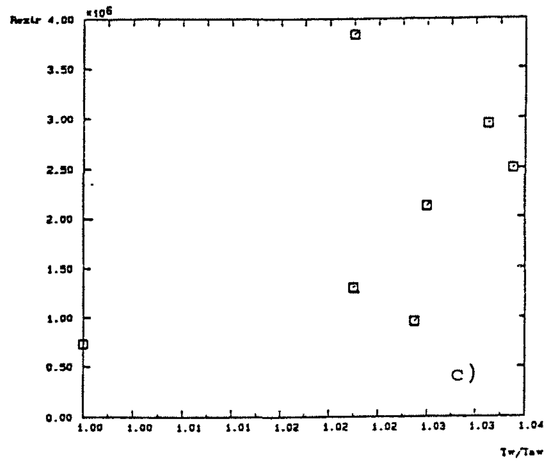


Fig. 16

Transition Reynolds number dependence on a) Re/M, b) α and c) T_w/T_{aw}.

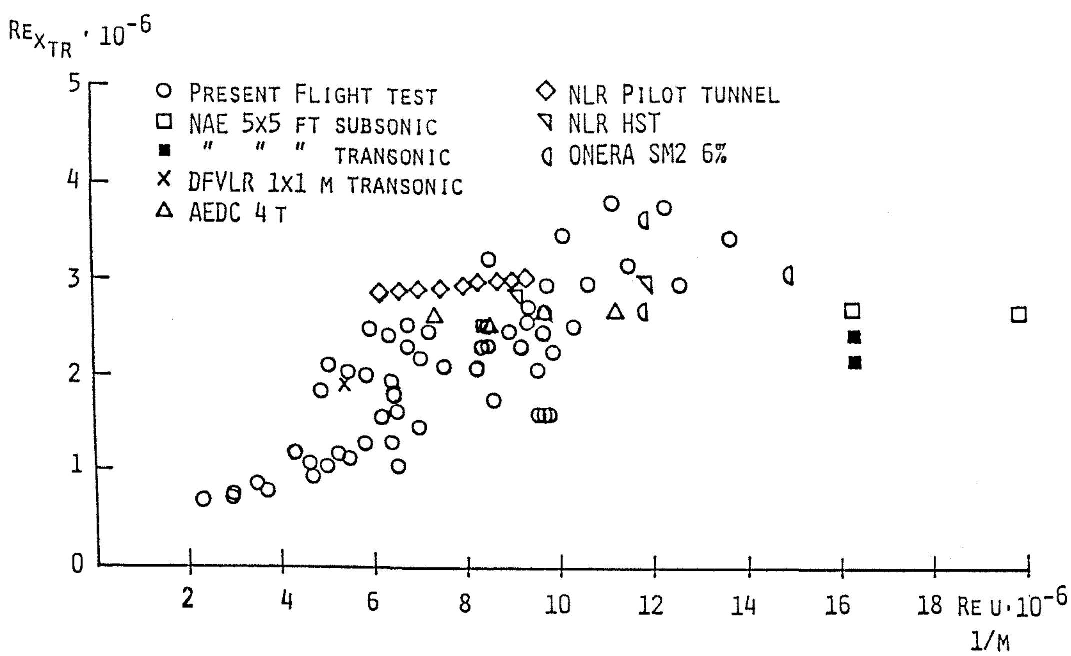


Fig. 17 Transition Reynolds number as function of unit Reynolds number. Flight test and comparison with wind tunnel tests.