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Experimental techniques used in the NASA GRC Transonic Flutter Cascade

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Application of several experimental techniques to measurements in the NASA Transonic Flutter Cascade, and experience with facility operation are presented in the paper. The discussed experimental techniques are shadowgraph shockwave structure visualization, dye-oil surface flow visualization, surface pressure measurements using pressure sensitive paint, and unsteady pressure measurements using miniature pressure transducers. Two flow phenomena were encountered during experimental runs: intermittent flow regimes in the transonic cascade, and facility resonance and self induced flow oscillations. Experience gained and suggestions for improvements to ease applications of these techniques for future research are summarized in the conclusion.

NASA Transonic Flutter Cascade Facility

The NASA GRC Transonic Flutter Cascade (TFC) is dedicated to studying the unsteady aerodynamics of oscillating airfoils, and is used to provide data for modeling the aerodynamics of blade stall flutter. The facility combines a linear cascade wind tunnel with a high-speed drive system that imparts pitching oscillations to cascade blades. The cascade consists of nine blades. All the blades or any single blade can be oscillated at frequencies up to *500 Hz* with amplitudes up to *2.4 dg*. Interblade phase angles can be varied in increments of *15 dg*. A view of the TFC test section is given in Fig. 1 (Ref. 1).

Shadowgraph shockwave structure visualization

The TFC is outfitted with a double-pass schlieren system. The cascade blades are mounted between end walls with a mirror on one side and an optical quality glass window on the other side. As seen in Fig. 2, the mirror and the window comprise a part of the schlieren system. Difficulties were experienced with the alignment of the schlieren system. Perfect alignment of the knife edge before the test would rapidly deteriorate once the flow started. It is believed that this was caused by minute deflections of the facility walls due to the pressure difference across the tunnel walls. Also, there was a noticeable unsteadiness in the image position at the knife edge location. For this reason, the pictures with shock wave structures present were acquired as shadowgraphs. A series of photographs in Fig. 3 depicts shockwave structure for two cascade inlet Mach numbers for high and low cascade solidity, as well as for a single airfoil (Ref. 2).

Dye-oil surface flow visualization

The surface flow visualization technique is based on a dye, which is of a liquid or a solid form, being smeared over the surface by the flow. Dots or lines of dye are made on the test surface just before the test facility is started. This is relatively simple and straightforward. However, very often, it is difficult to distinguish which of the resulting dye traces were made during the facility start-up process and which were made while running at the test conditions. The shut-down transient is usually no problem because the facility can run until the dye traces are sufficiently dry. To avoid this problem, two aspects must be resolved: (1) very short facility start-up time, and (2) a dye-oil mixture of suitable viscosity (in conjunction with small dye-oil marks on the test surface). By modifying the facility start-up procedure, the maximum Mach number of *1.2* was reached within *6 s* from the time the main shut-off valve started to open. This extremely short start-up time caused vibrations in the facility and should not be overused; however, so far no detrimental effects on the facility have been noticed. An oil/paint mixture was used for visualization. The right mixture ratio was determined for each color by testing the color mark fluidity. Color dots *2* to *3 mm* in diameter were made on a test surface, and the surface tilted in the vertical direction. Oil was added to the mixture as long as the color marks did not start to run after the test surface was tilted. Behavior of the color dots during the facility start-up procedure was observed to verify that the recorded flow pattern is not contaminated by the transient phenomenon. As mentioned above the flow in the cascade reached the test condition in *6 s*, during this period, and for an additional *10 s* to *15 s*, the fluid in the dots was pushed to the downstream side of the dot. Approximately *20 s* to *25 s* after flow onset the color dots started to break, and thin streaks ran over the blade surface. It took about 3 minutes for all the fluid from the dot to smear onto the blade surface. The flow was maintained at the test condition for 15 minutes in order to dry the paint. Examples of results accomplished are in Figs. 4 and 5, where changes in the flow pattern, respectively in the size of the flow separated region, on the blade suction side are shown for three cascade configurations. Finally, Fig. 6 combines the results of shock structure visualization with the blade surface flow pattern obtained using the dye-oil technique.

Surface pressure measurements using pressure sensitive paint

The technique of pressure sensitive paint (PSP) is based on luminescent coatings, which are painted on flow containing walls, and excited by light of selected wavelengths (Ref. 3). The excited paint emits light that is inversely proportional to the surface pressure. The emitted light – the surface pressure map – is imaged with digital cameras. There are two basic methods of retrieving a pressure signal from emitted light. In the first one, the intensity method ("wind-off/wind-on"), the pressure at a point is determined from the ratio of the wind-off to wind-on recorded light intensities for continuous illumination. In the second one, the lifetime method, pressure is determined from the decay time of luminescence after the paint has been excited by illumination flashes. A specific feature of the application of PSP to channel flows is the problem of possible signal misinterpretation due to signal reflections from neighboring walls (ghost images). There is no generic solution for eliminating ghost images and signal contamination in curved narrow channels. It is good practice to apply PSP to only one surface at a time and to paint remaining surfaces with antireflection coatings. If possible, the recording camera should be placed perpendicularly to the surface to be investigated (Ref. 4). Finally, in situations where optical access allows only observation at oblique angles, two data sets should be recorded, each taken at different illumination and recording angles. Because the position of a ghost image depends on the observation angle, the pressure maps retrieved from the above mentioned two data sets will be identical within the experimental error band only if they are free of ghost images and therefore represent the true surface pressures. Very often a camera can only be placed in a location that records the three-dimensional vane surface as a skewed two-dimensional image. Consequently, a data reduction algorithm must be devised for data conversion into a meridional projection view (Ref. 5). Of course, if possible, a camera should be placed in a position with a perpendicular view of the investigated surface. An example of PSP measurement on the cascade sidewall together with a shadowgraph of the shock wave structure is presented in Fig. 7 (Ref. 6).

Unsteady pressure measurements using miniature pressure transducers

Blade flutter and associated high cycle fatigue problems are very detrimental to the structural health of airfoil cascades. Origins of stall flutter are still not fully understood. Conventional static taps average the fluctuating pressures and thus do not allow efficient unsteady analysis of flow and blade structure interaction. The blades, in particular the oscillating blades, must be instrumented with miniature pressure transducer to acquire needed data on flutter effects. Traditionally, the blades are instrumented with special flat surface mount transducers flush with the blade contour. An example is shown in Fig. 8 (Ref. 7). The main reason for this approach is to maximize the frequency response of this pressure measurement system. However, there are several disadvantages to this approach. The flat transducers are very thin, however, they are of a relatively large diameter, and therefore they measure unsteady pressures averaged over a circle of at least *2 mm* in diameter. The sensing elements are practically exposed with minimum protection, therefore very vulnerable, requiring extremely careful handling. Our experience is that the ‘fatality’ rate due to accidental mishandling is very high. To protect the sensing element on the blade surface, the transducers are overlaid with a layer of silicone rubber (Fig. 8). This layer, however, also acts as a damper and decreases transducer frequency response. Also, the blade surface is disturbed to an extent as seen in Fig. 8.

Nevertheless, by far the largest problem is that the inertial loads on the transducer diaphragm due to the blade oscillations cause diaphragm deflections that can be misinterpreted as fluctuating pressures. The left-hand diagram in Fig. 9 illustrates this problem. The data was acquired for a blade oscillating at *500 Hz* with an angular amplitude of *1.2 dg*, and with no flow in the cascade. The peak acceleration at the location of this transducer was *375 g*. This acceleration causes apparent pressure fluctuations with amplitude up to *5 kPa*. Fig. 10 explains the root of this problem for surface mount transducers. Measured pressure fluctuations while blades are oscillated in the flow must be corrected for the apparent pressures, which is complicated due to the phase difference between the apparent and flow generated pressure fluctuations. An example of pressure signal correction for acceleration effects is in Fig. 11. The operational conditions were as follows: cascade inlet Mach number of *0.8*, frequency of blade oscillations *500 Hz*, amplitude of blade oscillations *1.2 dg*, and the peak local acceleration *375 g*. Amplitude and phase for each signal are stated in the figure. As seen here, the acceleration effects are quite significant and necessary.

Alternatively a conventional miniature pressure transducer can be used as a body-mount one as shown on the right-hand side in Fig. 10. In this approach, the pressure transducer is mounted just below the blade surface. All the ‘digging’ in the blade is done from the opposite side of the blade, thus leaving the surface of interest untouched. The transducer is connected with the surface by a *2-mm* long tube of *0.5 mm* in diameter. The transducer diaphragm is oriented perpendicular to the blade surface, which minimizes the transducer sensitivity acceleration effects. As a result, the amplitude of apparent pressure fluctuations decreases by an order of magnitude (about *0.5 kPa*), as seen in the right-hand diagram in Fig. 9. This is a major improvement in accuracy of pressure measurement on oscillating blades.

Intermittent flow regimes in a transonic cascade

Available pressure data from a transonic airfoil, measured on the suction side in the leading edge region using conventional static taps, exhibit a smooth and continuous drop with increasing inlet Mach number. Such data indicates that the local flow velocity continuously increases from subsonic to low supersonic values. However, this contradicts the observations of unsteady and intermittent behavior of the flow shock pattern for transonic inlet flow conditions (Ref. 8).

Surface flow visualization using an oil-paint mixture clearly shows that there are different flow patterns for subsonic and supersonic inlet Mach numbers (see Figs. 4 and 6). For subsonic inlet Mach numbers there is a large separated flow region on the blade suction surface just past the leading edge exhibiting a complex three-dimensional flow structure. For supersonic inlet flow, however, the flow past the leading edge is fully attached to the blade for a considerable length. This abrupt change of surface flow patterns is also not indicative of a smooth velocity increase through the transonic flow region.

Time resolved pressure signals clearly show that, starting at an inlet Mach number of *0.95*, there are momentary pressure level drops to a level for which the flow velocity jumps to a supersonic value. These bursts of supersonic velocity are at first very short (a few milliseconds) and infrequent. However, with increasing inlet Mach number the duration and number of appearances of supersonic flow velocities increases dramatically. For an inlet Mach number of about *0.95* to *0.97*, the local flow velocity at the blade leading edge is supersonic half of the time. However, it appears that the velocity is still switching randomly. As the Mach number increases further, the regions of supersonic flow velocity rapidly lengthen with very sporadic instances of subsonic velocity pockets. Finally, for inlet Mach numbers of *1.01* the pressure level has settled at a value which corresponds to established continuous supersonic flow. An intermittent pressure signal is shown in Fig. 11. Flow appears to be bi-stable in the transonic region, and thus simple averaging over the interval of measurement yields a value that is strongly dependent on the rate of intermittency, and biased toward the level with longer dwell time. The averaging should be made independently for both levels as is indicated in Fig. 12. Pressure jumping between two levels generates large intermittent loading on the blade leading edge region and can lead to the onset of blade vibration that may interfere with the being studied flutter phenomenon.

Facility resonance and self induced flow oscillations

It must always be remembered that any linear cascade only simulates flow in a real transonic compressor stage. Some of the flow phenomena observed in a linear cascade may be associated with the facility only and may not exist in real machine. Flow resonance in a test facility resonance may be a very good example. It has been observed in the NASA Transonic Flutter Cascade facility that for inlet Mach numbers of about *0.8*, the cascade flowfield exhibited intense low-frequency pressure oscillations (Ref. 9). The origins of these oscillations were not clear. It was speculated that this behavior was either caused by instabilities in the blade separated flow zone or that it was a tunnel resonance phenomenon. Because all unsteady instrumentation was located only on the blades, it was necessary to instrument the tunnel wall with unsteady pressure transducers to prove that the observed phenomenon was associated with the tunnel unsteady behavior. A plug-in-wall pressure probe, shown in Fig. 14, was built and use for this purpose (Ref. 10).

The first run with the instrumented wall and the empty tunnel immediately proved beyond any doubt that the self-induced low frequency oscillations are a tunnel phenomenon, probably some kind of tunnel resonance (Fig. 15). The entire duct system that connects the cascade facility with the NASA central exhaust system was inspected and several potential resonators were found. Several tubes were attached to the main duct that served either as connecting flanges for other facilities or compensatory elements to lower noise level in the central duct. It was found that reduction in length of some of the attached tubes noticeably changed the tunnel behavior. An effect of 'tuning' the tunnel is shown in Fig. 16. As seen here, correct tuning significantly suppresses the self-induced oscillations. Even though the frequency content is similar, the amplitude of pressure oscillations was reduced by an order of magnitude.

Another way to extract information about the blade forced oscillation response, particularly about the waveform, is to subject the data sets to ensemble averaging procedures. The results of ensemble averaging for the wall probe for mistuned and tuned tunnel and for the inlet Mach number of *0.8* and blade oscillation frequency of *500 Hz* are shown in Fig. 17. As seen here, the waveform of the forced signal does not differ significantly for mistuned and tuned tunnel configuration, however, the level of pressure unsteadiness (RMS values) for the mistuned tunnel is about twice that for the tuned tunnel.

Luckily, in the case described the tunnel resonance frequency and the blade forcing frequency were distinctly different, and so the tunnel resonance was not that detrimental. Spectral and ensemble averaging analyses showed that the interaction between self-induced oscillations and blade forced oscillation was weak, and did not alter the overall results significantly. The self-induced oscillations, however, increased the scatter in the forced oscillation signal from one period to another.

Summary of findings

The following areas require special attention while performing flutter experiments in transonic cascades.

- Cascade facilities dedicated to flutter research with a blade driving mechanism have optical access from one side only, and therefore a double-pass schlieren or shadowgraph system must be used. If possible, a back wall reflecting mirror should be mounted so that structural vibrations caused by the blade driving mechanism are eliminated. Excessive vibrations of the back-wall mirror make the use of schlieren visualization extremely difficult.

- Liquid dye used for surface flow visualization must satisfy conflicting requirements for dye fluidity; in short it must smear easily over the surface when driven by the flow, however it must not run on inclined or vertical surfaces due to gravity. To avoid data contamination due to transient flow patterns, the tunnel start-up time to reach the operational flow conditions must be very short.
- The application of pressure sensitive paint requires optical access to the tested surfaces. Signal internal reflection at certain observation angles (ghost images) can severely contaminate and distort the image of the investigated pressure field. Use of fiber optics components may help minimizing signal internal reflection from curved blade passage surfaces.
- The effects of inertial loads on miniature pressure transducers mounted on oscillating blades must be minimized to avoid misinterpretation of the recorded fluctuating pressures. It is advantageous to mount the transducers such that the transducer diaphragm is parallel to the plane of the blade acceleration vector.
- Flow intermittency in a transonic cascade for high speed subsonic inlet Mach numbers generates a large intermittent loading at the blade leading edge region, and can lead to the onset of blade vibration, which can interfere with blade flutter investigations.
- Self induced oscillations and tunnel resonance can severely distort experimental data of blade flutter. The level of self-induced pressure fluctuations can be significantly suppressed for a narrow range of inlet Mach number by tuning the tunnel.

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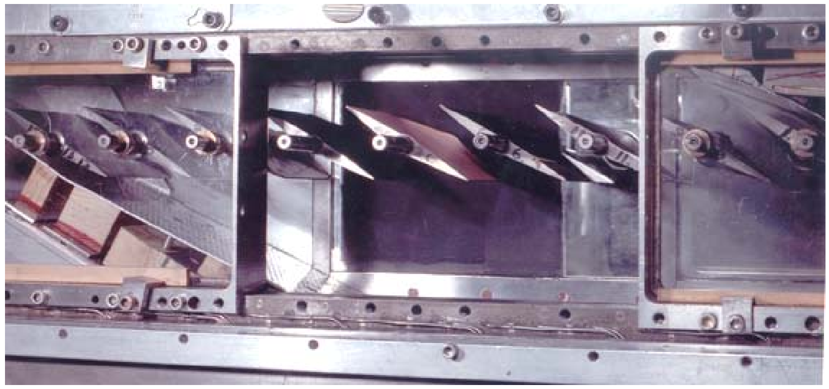

Fig. 1. View of the NASA TFC test section.

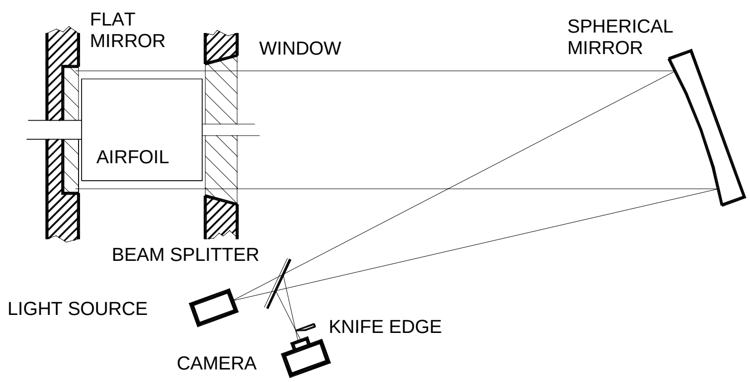

Fig. 2. Double-pass schlieren system.

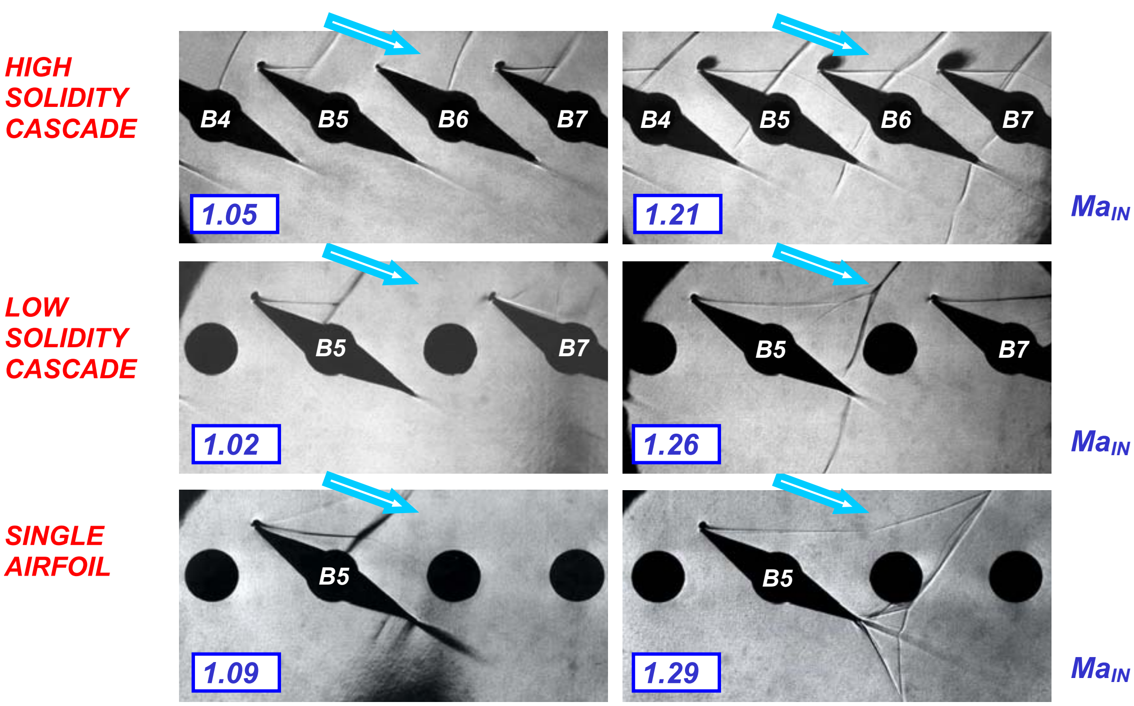

Fig. 3. Shadowgraphs of shockwave structure.

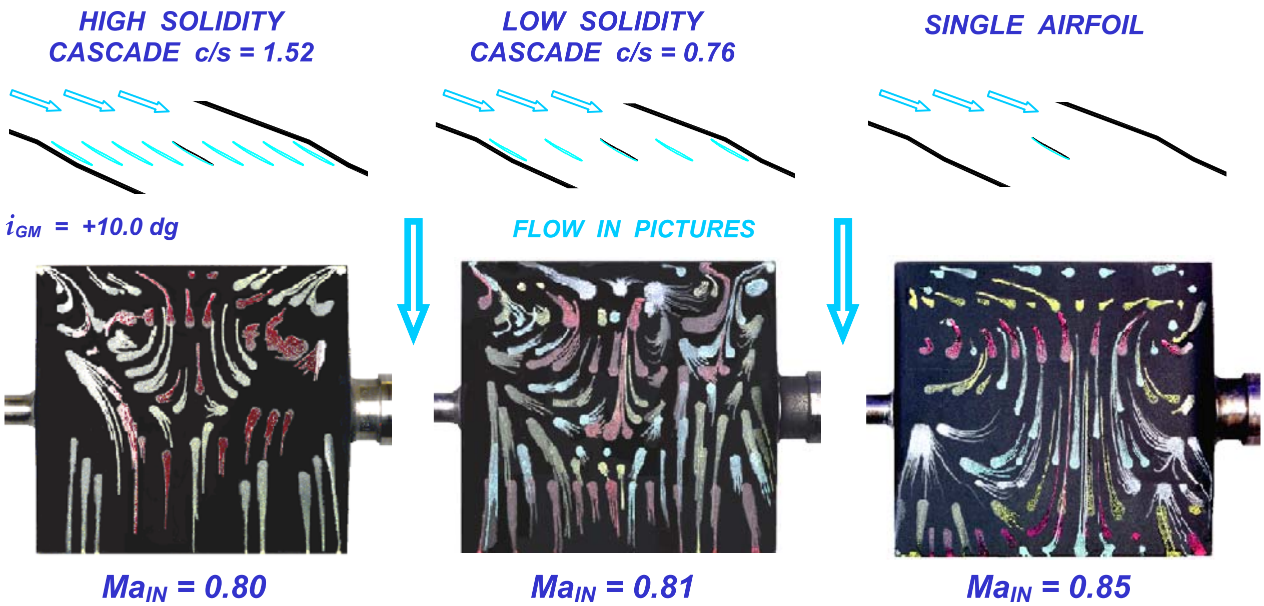

Fig. 4. Surface flow patterns for high subsonic inlet Mach number.

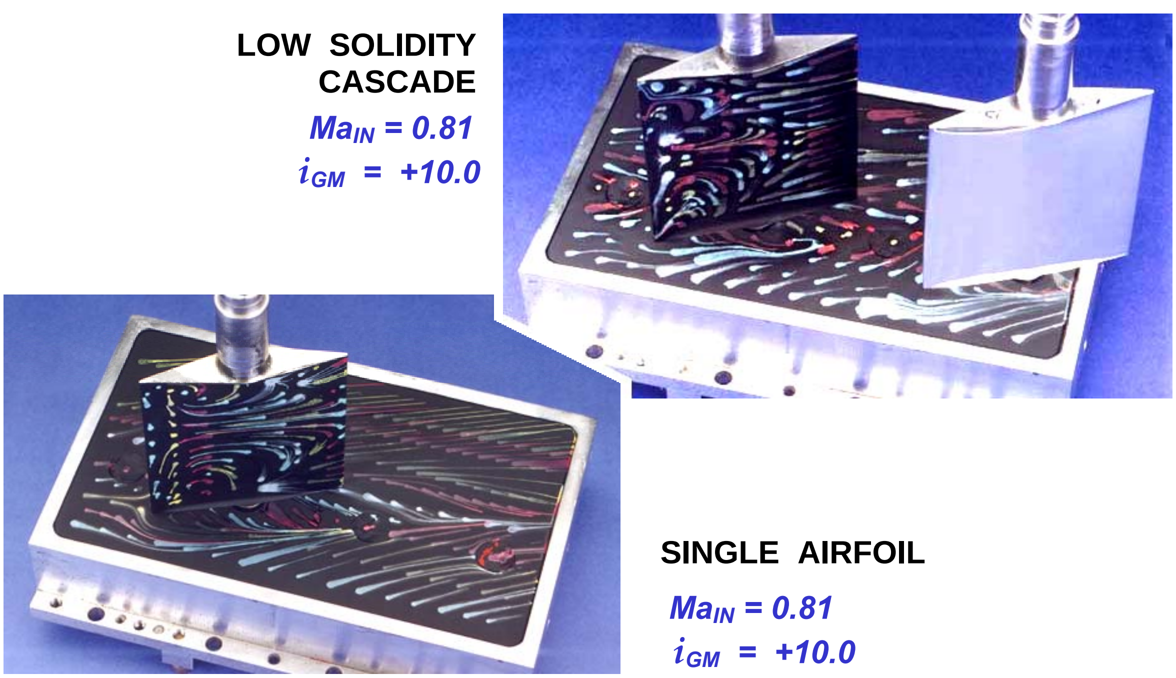

Fig. 5. Surface flow visualization on a blade and cascade sidewall.

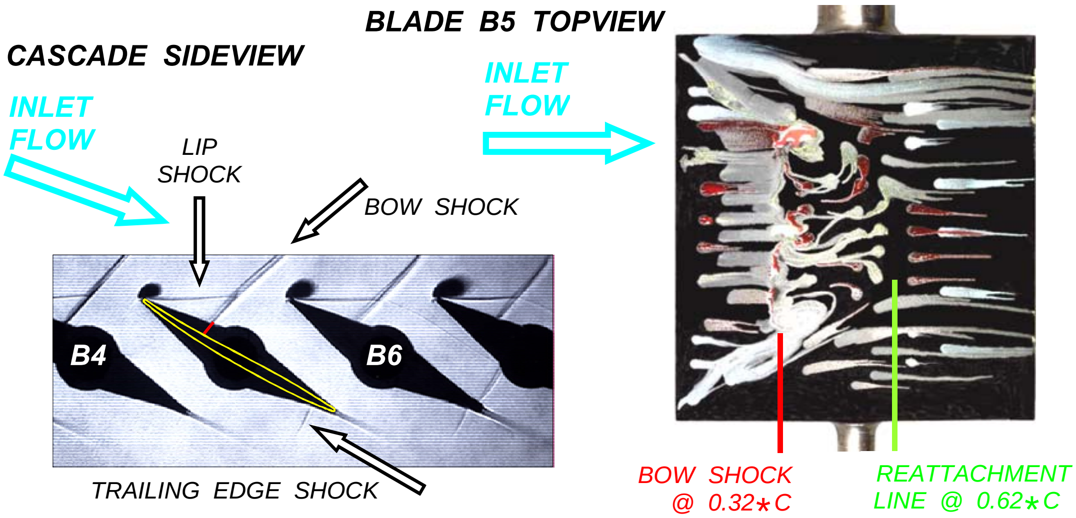

Fig. 6. Shock wave structure and blade surface flow pattern for supersonic inlet flow of Ma = 1.18.

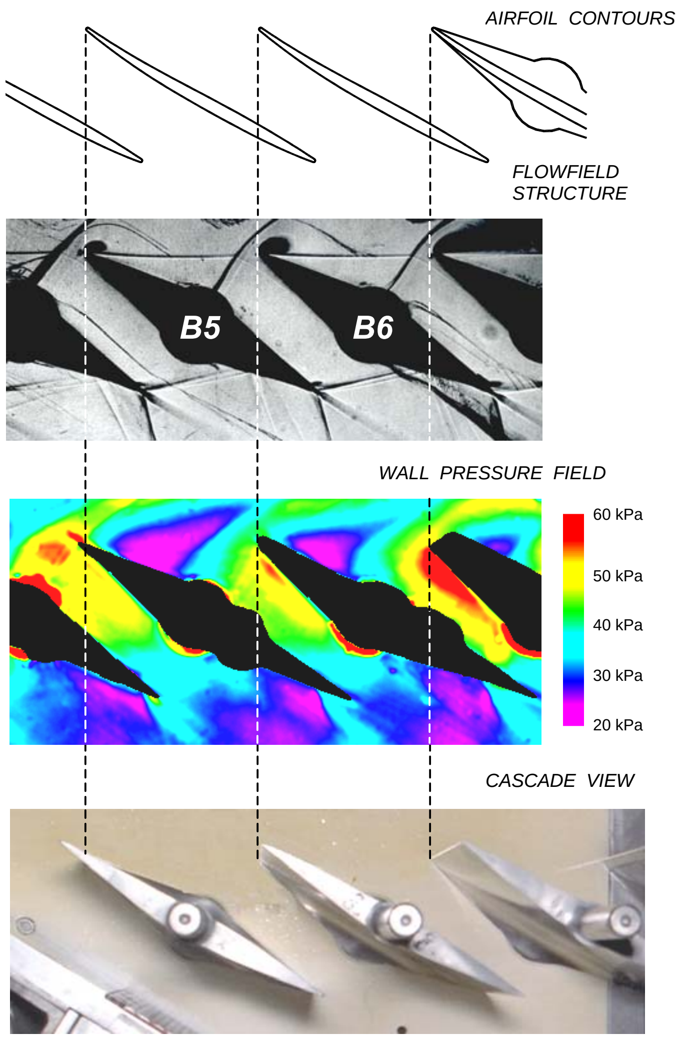

Fig. 7. Cascade flowfield for Mach number 1.3.

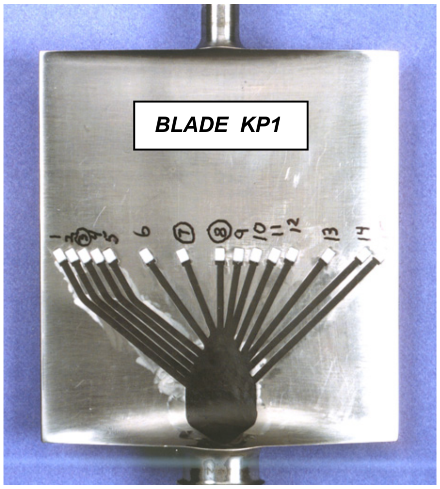

Fig. 8. Transonic fan blade instrumented with 15 Kulite transducers.

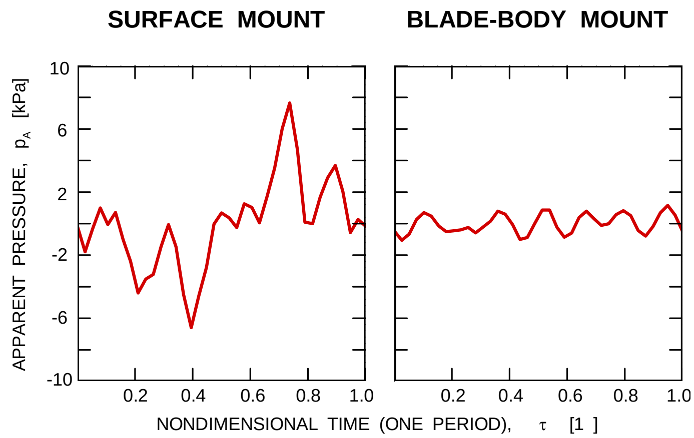

Fig. 9. Effects of transducer mounting on sensitivity to blade oscillations.

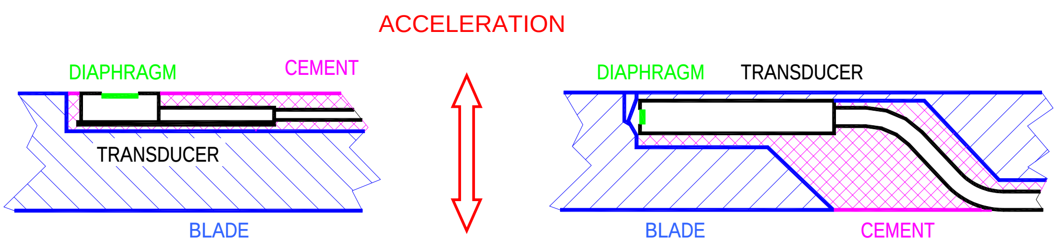

Fig. 10. Surface mount and blade body mount installation of pressure transducers.

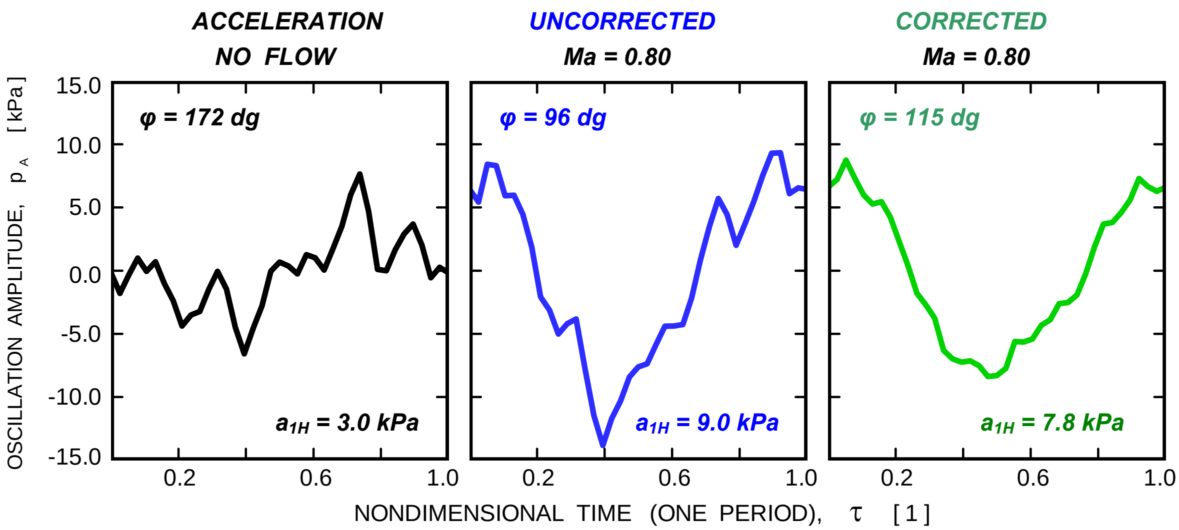

Fig. 11. Correction for acceleration effects.

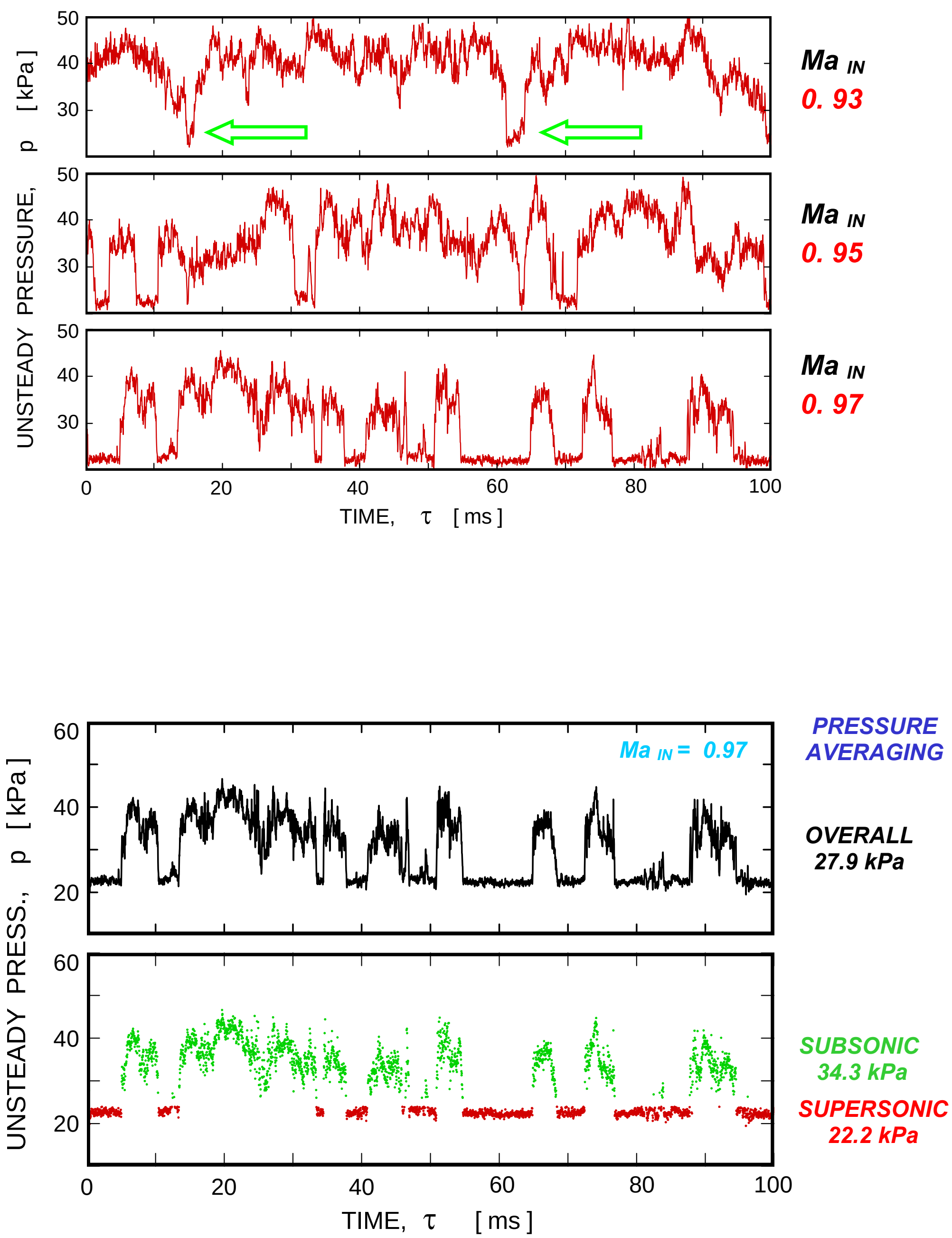

Fig. 13. Averaging of intermittent pressure signals.

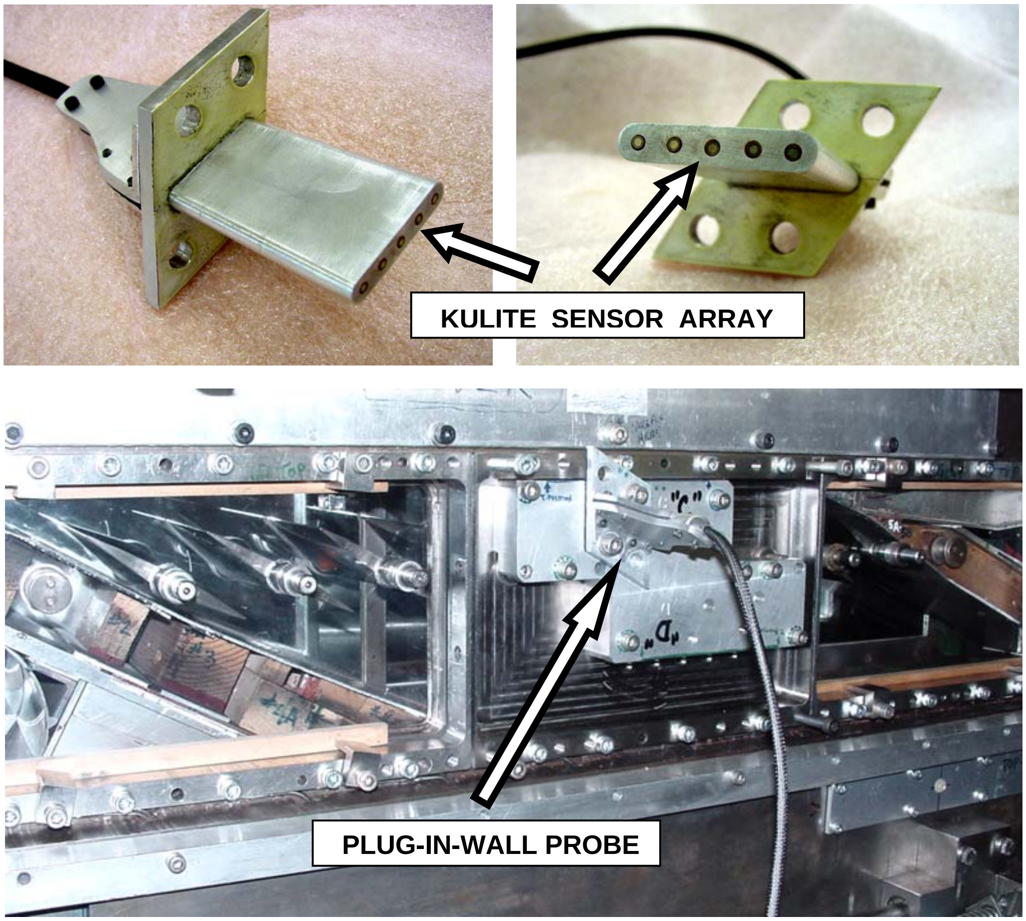

Fig. 14 Plug-in-wall static pressure probe.

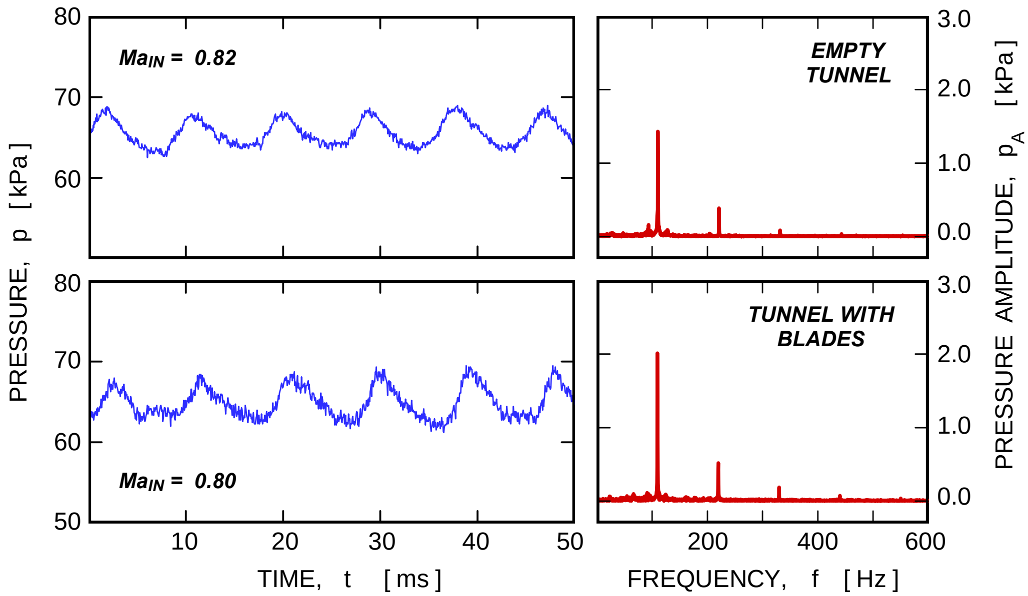

Fig. 15. Pressure oscillations in an empty tunnel and tunnel with blades.

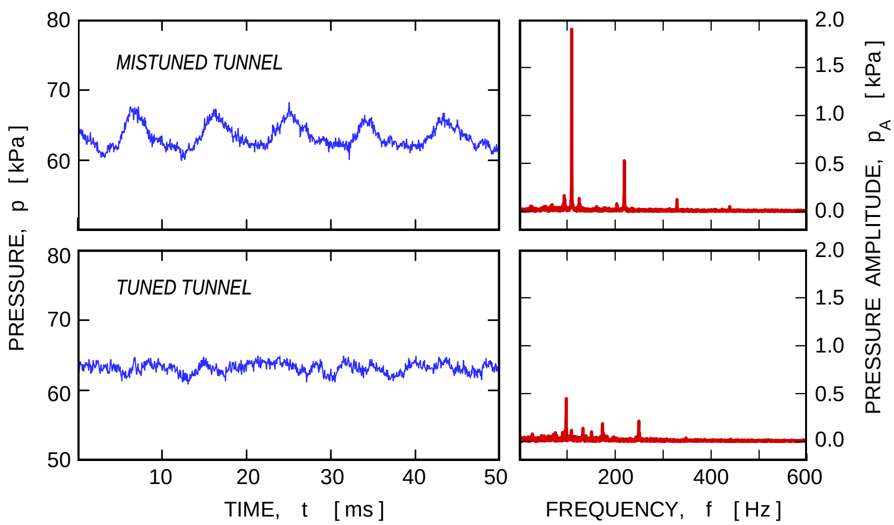

Fig. 16. Tunnel wall unsteady pressures for an inlet Mach number of *0.80* for mistuned and tuned tunnel configurations.

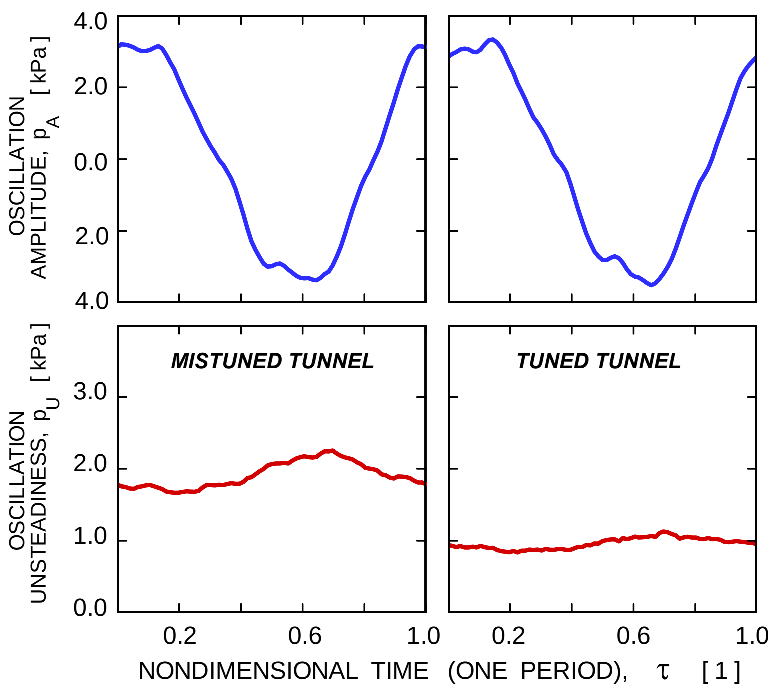

Fig. 17. Effects of tunnel tuning on forced pressure oscillation as recorded on tunnel wall for inlet Mach number of *0.8*, blade frequency 5*00 Hz*, and oscillation amplitude of *1.2 dg*.