1982-07

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The Shock and Vibration Digest, Volume 14, No. 7, July 1982.
http://hdl.handle.net/10945/43292

VOLUME 14, NO. 7
JULY 1982

THE SHOCK AND VIBRATION DIGEST

A PUBLICATION OF
THE SHOCK AND VIBRATION
INFORMATION CENTER
NAVAL RESEARCH LABORATORY
WASHINGTON, D.C.

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ENGINEERING

THE SHOCK AND VIBRATION DIGEST

Volume 14, No. 7
July 1982

A publication of

THE SHOCK AND VIBRATION INFORMATION CENTER

Code 5804, Naval Research Laboratory
Washington, D.C. 20375

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TRANSONIC BLADE FLUTTER: A SURVEY OF NEW DEVELOPMENTS

M.F. Platzer*

Abstract. This paper presents a review of current work in transonic blade flutter research. Aerodynamic analyses for the prediction of attached flow flutter, choke flutter, and stall flutter are described. Also reviewed are unsteady aerodynamic measurement and flutter test programs that have recently been completed or are in progress to investigate transonic blade flutter phenomena.

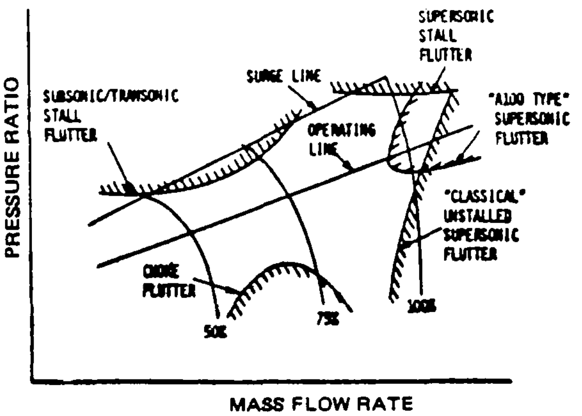
An earlier survey [1] presented a general description of the transonic blade flutter phenomenon and summarized the analytical and experimental investigations devoted to the problem prior to 1978. This paper updates the current status of transonic blade flutter research.

The term transonic flutter is used here to indicate that flow over the outer span of a blade is either transonic or supersonic. Depending on flow conditions, any of four types of flutter can occur:

- transonic choke flutter, when the blade is operating at near-choke conditions
- supersonic unstalled flutter, when the attached flow over the outer span of the blade is either fully supersonic or at least transonic
- supersonic stall flutter, when the outer portion of the blade is operating supersonically but flow is partly or fully separated
- subsonic/transonic stall flutter, when the blade is operating at high subsonic or transonic speeds and is partly or fully separated

Typical flutter boundaries that have been observed on modern compressors are shown in the figure. The choke flutter boundary is encountered during part-speed operation. The blades operate transonically at negative incidence angles and, due to a choked flow, in-passage shocks with possible flow separations are likely to occur.

The supersonic unstalled flutter boundary imposes an important high-speed operating limit. Recent tests have shown that the blade flutter mode during this type of flutter either can be predominantly torsional or consist of a large vibrational deformation of the blade camber line (chordwise bending mode). Still another type of supersonic torsional flutter, designated A-100 flutter, has recently been identified; it occurred only above a threshold level pressure ratio. Two additional flutter boundaries can be encountered during operating near surge: supersonic stall flutter and subsonic/transonic stall flutter.



Types of Fan/Compressor Flutter [27]

AERODYNAMIC THEORY

As pointed out in the earlier review [1] viscous flow effects are usually ignored at the outset, so that the analysis is based on a suitable approximation of the Euler equations. The fully linearized equations have been widely used in recent years, especially for supersonic oscillating cascades, but more recent work has concentrated on modeling nonlinear flow effects.

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FLOW MODELS

Because the analysis of three-dimensional non-steady flow through a transonic multistage machine is prohibitively complex, various simplifying assumptions must be made in order to make the problem mathematically tractable. The most simple and widely used model -- the cascade flow model -- is obtained by unwrapping an annulus of differential radial height from the flow passage of an axial-flow turbomachine. Only one cascade is usually considered because of the complex interactions between neighboring blade rows. As has been pointed out [1] the case of supersonic cascade flow requires further differentiation, depending on the axial through-flow Mach number. This number can be either subsonic (causing propagation of disturbances upstream of the blade leading edges), moderately supersonic (causing interactions only between the reference blade and its adjacent blades), or highly supersonic (causing no interactions between neighboring blades).

Three-dimensional flow models have also been introduced. Examples include flow past a vibrating blade row of finite blade height situated between end plates and flow past an annular blade row with a finite number of vibrating blades rotating at a constant angular velocity in an infinitely long cylindrical duct.

ANALYSIS OF COMPRESSIBLE FLOW PAST OSCILLATING CASCADES

A subsonic non-steady potential flow analysis [2-4], which accounts for the effects of blade geometry and steady turning, is being developed at the United Technologies Research Center. The formulation is based upon the full nonlinear potential equation, which is linearly perturbed to describe small amplitude blade motions. Hence the problem is reduced to the solution of linear equations with variable coefficients; these in turn must be determined from the steady flow solution.

Finite difference approximations determined from an implicit least-squares method and applicable on arbitrary grids lead to a linear system of algebraic equations that is tri-diagonal and thus permits an efficient non-iterative solution. Special care is required to treat rounded leading edges at angle of attack. Results obtained to date include the effects of cascade geometry, inlet Mach number, and inlet flow angle on the non-steady response of flat plate and NACA 0012 cascades. An extension of this approach to transonic flow is intended in the near future.

Whitehead and Grant [5] attacked this same problem using the finite element method. The assumption of small non-steady perturbations superimposed on the steady high-deflection cascade flow was again made, and the approach again was limited to subsonic inlet and outlet flows or at most with small supersonic shockless bubbles. Computed moment coefficients were in good agreement with experiments available for a turbine cascade.

Several attempts are presently being made to develop transonic oscillating cascade flow solutions that remove some of the limitations imposed by the purely linearized analyses described in the last review [1].

Adamczyk [6] used the transonic small disturbance equation as the governing equation. He was thus able to retain the effect of the steady aerodynamic loading and shock wave motion on the non-steady flow characteristics. The resulting non-steady flow equations can be solved analytically, thus leading to rapid flow computations.

Williams [7] considered small amplitude perturbations superimposed on an arbitrary steady nonuniform flow so that the disturbed flow is isentropic but may be a non-potential flow. Both blade vibration and inflow disturbances are considered. The resulting equations split into acoustic and vortical field equations that are independent of each other. They are solved numerically using the indicial response formulation. Hence the main program output are load time histories -- i.e., aerodynamic transfer functions -- that can be used together with a Fourier synthesizer to study flutter and forced response problems.

Kerlick and Nixon [8] also pursued the indicial function approach for the solution of the transonic small disturbance equation together with Nixon's strained coordinate method [29]. This approach permits decoupling of the essentially nonlinear non-steady part of the problem into two linear problems so that the principle of superposition can be applied. A high-frequency version of the Ballhaus-Goorjian code is being adapted to compute transonic cascade flows by imposing the proper cascade geometry and periodic boundary conditions.

A non-steady aerodynamic analysis has been developed [9] to predict choke flutter; the semi-actuator disk model was used. Because in-passage shocks are likely to occur during this flutter condition, a channel flow model was adopted in which a normal shock was positioned as a function of the pressure perturbations and non-steady airfoil motion. The validity of this model was evaluated by comparison with choke flutter data available from an F-100 core engine that produced reasonable agreement between theoretical predictions and experiments.

Adamson and Messiter [10] have produced detailed analyses of transonic channel flows using asymptotic expansion procedures that permit the study of shock formation and propagation in response to downstream pressure disturbances or to blade motion. Under certain conditions large shock motions occurred. The incorporation of this work into a choke flutter analysis might therefore uncover important additional aspects.

Purely supersonic unsteady cascade analyses have been developed by a number of investigators over the past decade. Most of these approaches are based on the linearized non-steady potential equation and are therefore valid only for very thin blades. The influence of blade thickness and steady loading on supersonic blade flutter characteristics has not yet been resolved, but two papers have recently reported progress on this problem.

Namba [11] developed a model for lightly loaded cascades with weak oblique shocks that can be approximated by Mach lines. The model allows for blade thickness and loading by introducing additional multipole sources fixed at time-mean positions but neglects the effect of deviation of the local convection velocity and local speed of sound from the uniform flow values.

Vogeler [12] developed a method of characteristics solution of the transonic small disturbance equation that accounts for weak shock waves. This work is an extension of earlier work [13] and predicts significant blade thickness effects in some cases.

Halliwell [14] recently evaluated the usefulness of purely linearized supersonic non-steady cascade theory and of a strong normal shock model developed earlier [15] for practical flutter predictions. Halliwell observed that the purely linearized model predicts a much greater range of instability than the normal shock model; he advocates development of an oblique shock model that would reproduce the correct pressure rise-Mach number relationship in the blade passage.

Progress has also recently been made in analyzing supersonic stall flutter of high-speed fans. An analytical model for predicting the onset of supersonic stall bending flutter has been developed [16]. It is based on a modified two-dimensional compressible non-steady actuator disk theory. This model comprises a cascade of airfoils at 85 percent of the blade span; the airfoils are assumed to be unshrouded and to vibrate in their first flexural mode. Shock wave and flow separation effects are accounted for by incorporating the measured quasi-steady rotor total pressure losses and deviation angles.

EXPERIMENTAL PROGRAMS

Since the last review [1] a number of experimental programs have been continued or initiated to investigate high-speed blade flutter phenomena.

Tests in the Allison supersonic cascade tunnel have been continued. Two cascades of airfoils modeling outboard sections of rotors experiencing supersonic torsional and translation flutter have been investigated; a five-bladed cascade was harmonically oscillated in the proper mode at varying interblade phase angles. Oscillatory pressures were recorded from flush-mounted, high response pressure transducers. Cascade stability plots were obtained over a range of back pressures; it was determined that large amplitude shock motion was not required for torsional instability. The experimental data were correlated with both a low-back pressure and a high-back pressure (normal shock) analysis; a requirement for a moderately loaded cascade analysis was identified. This program was sponsored by NASA-Lewis [17].

Another series of tests was performed in the Allison rectilinear turbine cascade facility. A cascade of five airfoils was used to model the hub section of an advanced design turbine featuring a high subsonic inlet Mach number. Either subsonic or low supersonic exit Mach number was oscillated in the pitch or plunge mode at varying interblade phase angles and frequencies. The oscillatory pressure signals were measured with Kulite pressure transducers and correlated with a state-of-the-art analytical prediction based on a flat-plate cascade [28].

Another NASA-Lewis-sponsored program [18] is being conducted at General Electric. A 32-percent scale rotor of the Quiet Engine Program (QEP) Fan C design is being tested for stall flutter in both the subsonic and supersonic regions. The scaled down fan operates at a design pressure ratio of 1.6. Steady aerodynamic instrumentation was used to measure overall rotor and blade element performance. Non-steady aerodynamic characteristics were determined with blade-mounted sensors, casing-mounted sensors, and traversable dynamic probes. Test results obtained thus far have demonstrated successful duplication of flutter modes and boundaries experienced on the full scale fan C and the acquisition of steady and non-steady aerodynamic data during both steady-state and flutter rotor operating conditions.

At the NASA-Lewis Research Center a linear cascade tunnel is under development [19] to study transonic stall flutter. Flow visualization and dynamic pressure instrumentation are being used to determine transonic flow through an oscillating blade row in order to simulate realistic transonic flutter conditions.

At Modane, France, a high-speed cascade tunnel was put in service by ONERA in 1978. It has been operating at Mach numbers from 0.4 to 1.0 but is designed for supersonic testing. The cascade consists of six blades that are instrumented with pressure transducers [20]. A recently completed stall flutter investigation [21] showed that the non-steady pressures causing stall flutter of thin compressor blades occurred mainly on the forward half of the upper surface.

At the Technical University of Lausanne [22] an annular cascade tunnel capable of testing transonic turbine blades is under development. The Japanese are also interested in the problem of transonic steam turbine blade flutter; a nine-blade cascade has been tested over a range of outlet Mach numbers up to 1.8 [23]. Shock-induced flow oscillations occurred over a limited range of pressure ratios that led to negative aerodynamic pitch damping.

Transonic/subsonic compressor stall flutter has been investigated at Pratt and Whitney Aircraft Company [24]; a modern, shrouded fan stage (TS22 fan) was used. Steady-state structural deformations of the blades were determined by means of blade mounted mirrors. Non-steady deformations during flutter were obtained from mirrors, strain gages, and high-response pressure transducers. Traversing hot film probes were used upstream and downstream of the rotor to detect evidence of flutter in the blade wakes. The major findings were that all 32 blades fluttered at the same frequency but significantly varied in amplitude and interblade phase angle. The measured non-steady pressure distributions indicated that major flutter input occurred close to the blade leading edges. It was also found that the local Mach number had to exceed one for stall flutter to develop and that there was evidence of shock oscillation.

At Massachusetts Institute of Technology (MIT) a method has been developed and demonstrated to measure directly the aerodynamic forcing and aerodynamic damping of a transonic compressor [25]. The MIT transonic compressor -- described by Kerrebrock [26] -- was instrumented with piezoelectric displacement transducers, three accelerometers to measure in-plane motion of the disk, and a leading-edge mounted total pressure transducer.

When operated in rotating stall, the blades were excited at the fundamental frequency of stall cell excitation. The rotor was excited close to the operating point by a controlled two-per-revolution fixed upstream disturbance and then was sharply terminated during the test run. The disturbing and damping forces were determined by an inverse solution of the structural dynamic equations of motion of the blade-disk system; these equations were expressed in individual blade coordinates. The aerodynamic damping of the MIT-rotor was successfully determined; the structural dynamic blade-disk interaction of the rotor was dominated by the in-plane rigid body modes of the disk.

SUMMARY AND OUTLOOK

It is apparent from the major papers published since the previous review [1] that the dominant aerodynamic model is still the cascade flow model -- i.e., the two-dimensional flow representation. However, considerable effort is being devoted to approaches based on nonlinear governing equations. Hence significant advances in non-steady transonic cascade aerodynamics and in the analysis of steady loading effects on blade flutter can be expected in the next few years.

A number of cascade and rotating rig tests have been completed or are being built to provide a data base for assessing the prediction methods. Nevertheless, flutter phenomena strongly influenced by flow separation, shock-boundary layer interactions, three-dimensional flow effects*, and blade row interactions continue to impose great uncertainties on the predictive capability of current methods. These phenomena will require continued theoretical and experimental efforts.

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Volume 14, No. 7 July 1982