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SUPERSONIC AXIAL-FLOW FAN FLUTTER

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ABSTRACT

The development of a supersonic axial-flow compressor has been the subject of a limited amount of research over the past 32 years (Ferri, 1956; Klapproth, 1961; and Savage et al., 1961). During the middle 1970's a supersonic axial-flow compressor was constructed, but it encountered a blade failure before reaching its design point (Breugelmans, 1975). Many reached the conclusion that the supersonic axial-flow compressor was a very difficult, if not practically impossible, design problem. However, recent renewed interest in supersonic and hypersonic flight vehicles have rekindled interest in the supersonic axial-flow fan. For example, a research project to design, build, and conduct experiments on a single-stage supersonic axial-flow fan is now underway at the NASA Lewis Research Center (Schmidt et al., 1987; and Wood et al., 1987).

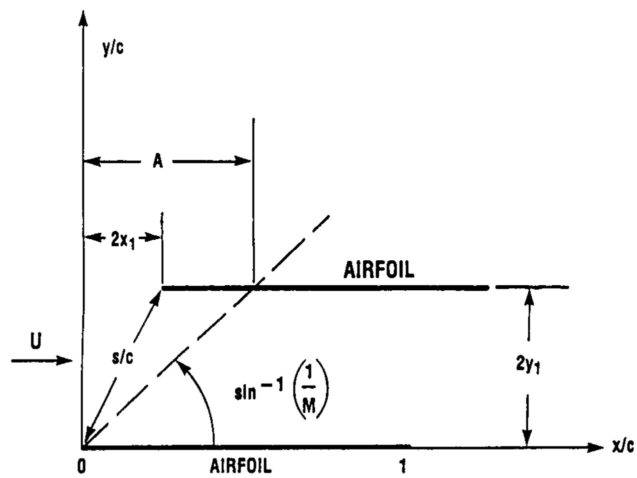
Although past experimentation on this type of compressor has been rather sparse, some useful analytical technology has been developed. One example is in the area of aeroelastic stability. Since the aeroelastic stability of the NASA supersonic through-flow fan was a concern, an analytical capability was needed to predict the unsteady aerodynamic loading. Consequently, a computer code based on Lane's (1957) formulation was developed for the case of supersonic axial flow (Ramsey and Kielb, 1987). This presentation will discuss this code and its application to the flutter analysis of the NASA Lewis supersonic through-flow fan.

The flutter analysis was performed by incorporating this code into an existing aeroelastic code and applying it to the NASA blade. The analysis (Kielb and Ramsey, 1988) predicted the blades to be unstable at supersonic relative velocities. As a consequence, the rotor blades were redesigned by reducing the aspect ratio to bring the through-flow fan into the stable operating range.

UNSTEADY AERODYNAMIC MODEL

Lane's (1957) formulation for the unsteady pressure distribution was used to calculate the unsteady aerodynamic loads. This formulation considers a cascade of two-dimensional flat plates with arbitrary stagger (provided the locus of blade leading edges is located ahead of the Mach lines) and arbitrary interblade phase angle. The upper figure shows the cascade geometry, and the lower figure defines the airfoil unsteady pressure distribution.

CASCADE GEOMETRY



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UNSTEADY PRESSURE

$$P_-(x,t) - P_+(x,t) = 2\rho U^2 \Upsilon(x)\, e^{i\omega t}$$

$$\Upsilon(x) = -B^{-1}(\partial/\partial x + iK)\left\{\int_0^x \alpha(\zeta) e^{-i\kappa M(x-\zeta)}\right.$$

$$\times\left[e^{-i\Omega}\sum_{n=0}^{\infty} J_0\left\{\kappa\sqrt{(x+2x_1-\zeta)^2-(1+2n)^2A^2}\right\}1[x+2x_1-\zeta-(1-2n)A]\right.$$

$$+\,e^{i\Omega}\sum_{n=0}^{\infty} J_0\left\{\kappa\sqrt{(x-2x_1-\zeta)^2-A^2(1+2n)^2}\right\}1[x-2x_1-\zeta-(1+2n)A]$$

$$\left.\left.-\sum_{n=0}^{\infty}\epsilon_n J_0\left\{\kappa\sqrt{(x-\zeta)^2-4n^2A^2}\right\}1[x-\zeta-2nA]\right]d\zeta\right\}$$

WHERE $\Omega = \sigma + 2\kappa M x_1$, $\kappa = KM/B^2$, and $A = 2By_1$

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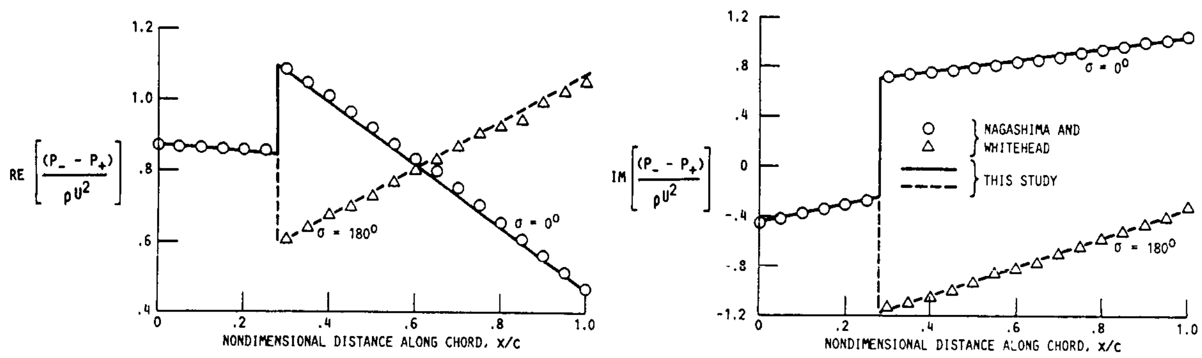
COMPUTER CODE VERIFICATION - PITCHING MOTION

The pressure distribution and lift and moment coefficients due to torsional motion were compared with Nagashima and Whitehead's (1977) published results. Close agreement can be seen.

PRESSURE DIFFERENCE
(M = 2.5, s/c = 1.0, STAGGER ANGLE = 60°, K = 1.0, x_0 = 0.5)



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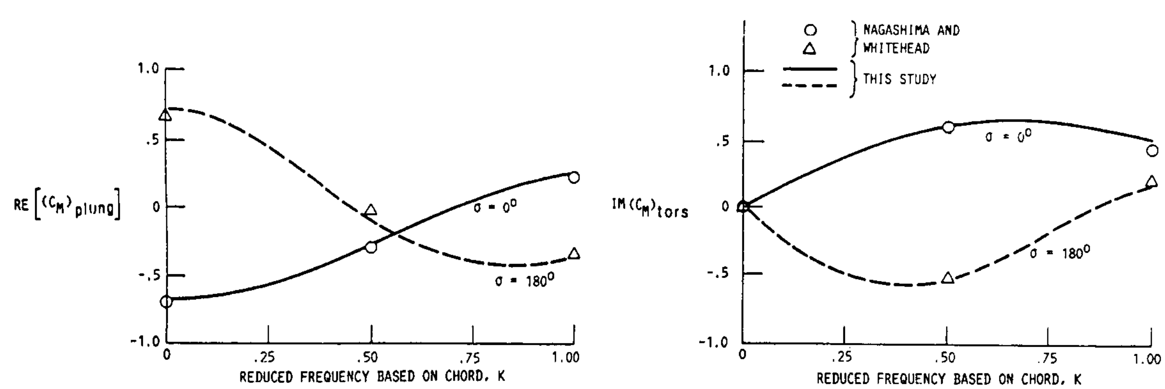
MOMENT COEFFICIENT
(M = 1.2, s/c = 1.0, STAGGER ANGLE = 0°, x_0 = 0.5)



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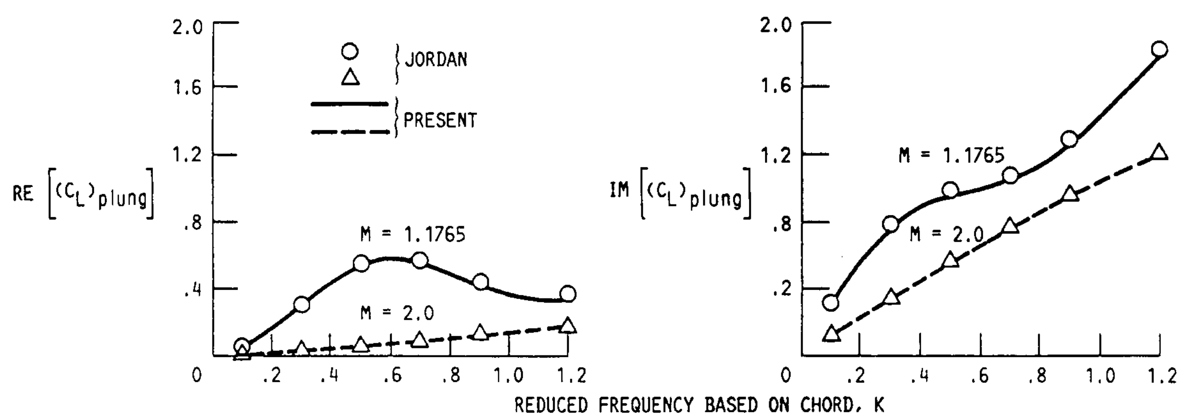
COMPUTER CODE VERIFICATION - PLUNGING MOTION

It would have been ideal to compare the lift and moment plunging coefficients with those of Nishiyama and Kikuchi (1973). However, it was felt that the published graphs were too small to accurately digitize. Therefore, the plunging coefficients obtained from this code were compared to those of Jordan (1953) for an isolated airfoil in supersonic axial flow. Close agreement can be seen.

LIFT COEFFICIENT



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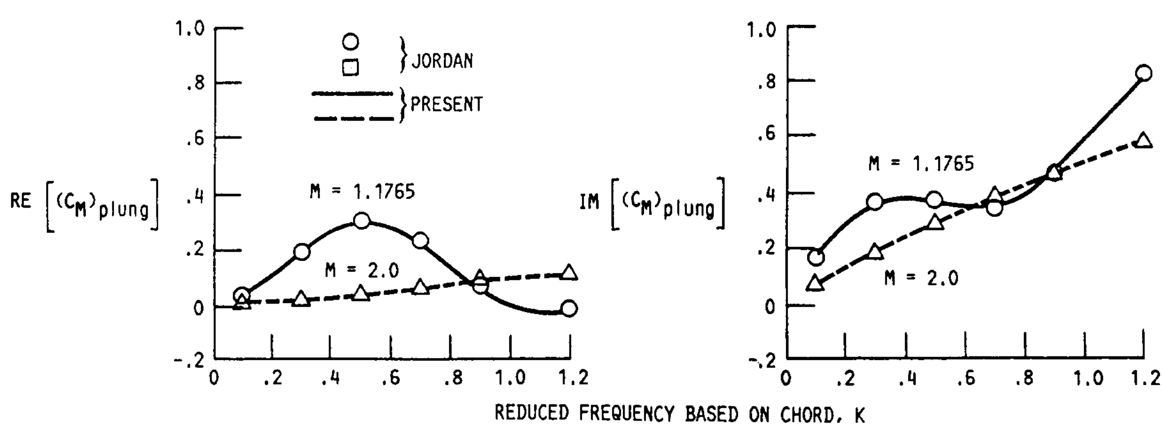
MOMENT COEFFICIENT



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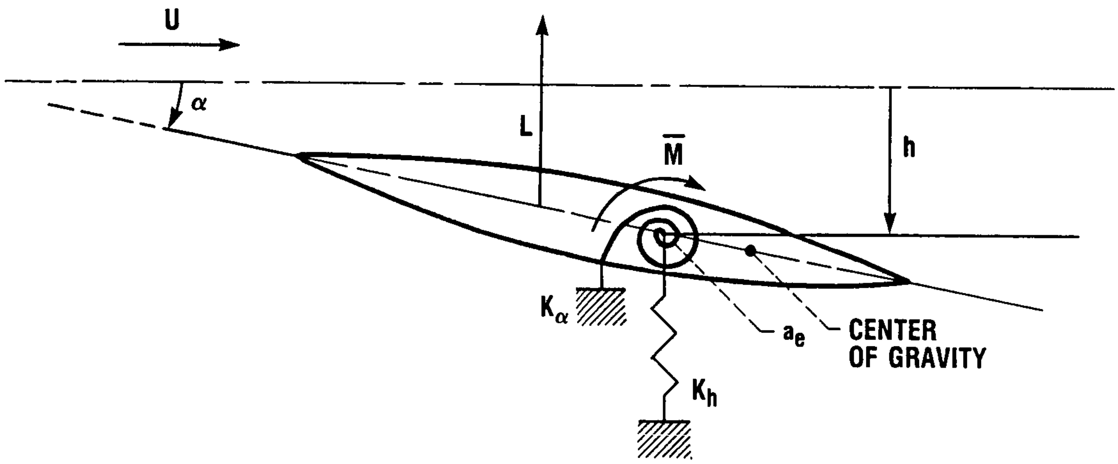
STRUCTURAL MODEL

The classical typical section is used to model the structure. Each airfoil is assumed to be a two-degree-of-freedom oscillator supported by bending and torsional springs. The airfoil is assumed to be rigid in the chordwise direction. Coupling between bending and torsional motions is modeled through the offset distance between the center of gravity and the "elastic axis."



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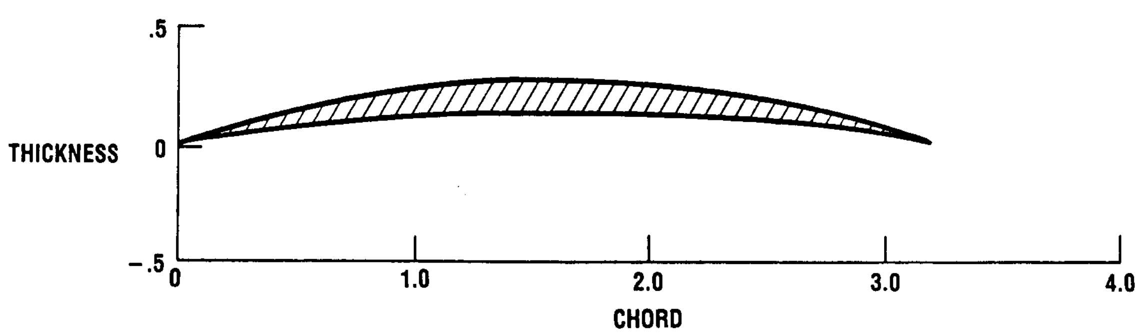
ORIGINAL DESIGN - 73.3-PERCENT SPAN

The NASA blade is much higher in solidity and lower in stagger angle than typical fan stages. However, the airfoil cross section is similar to that of conventional fan blades. The first mode is primarily bending, and the second mode is primarily torsion. The physical properties of the 73.3-percent span location were chosen as being representative and were used in the flutter analysis.

NUMBER OF BLADES	58
MASS RATIO	456.2
RADIUS OF GYRATION (MIDCHORD)	0.431
STAGGER ANGLE, deg	28
SOLIDITY	3.215
AT DESIGN POINT	
AXIAL MACH NUMBER	2.300
RELATIVE MACH NUMBER	2.606
REDUCED FREQUENCY (BENDING MODE)	0.376
REDUCED FREQUENCY (TORSION MODE)	0.663



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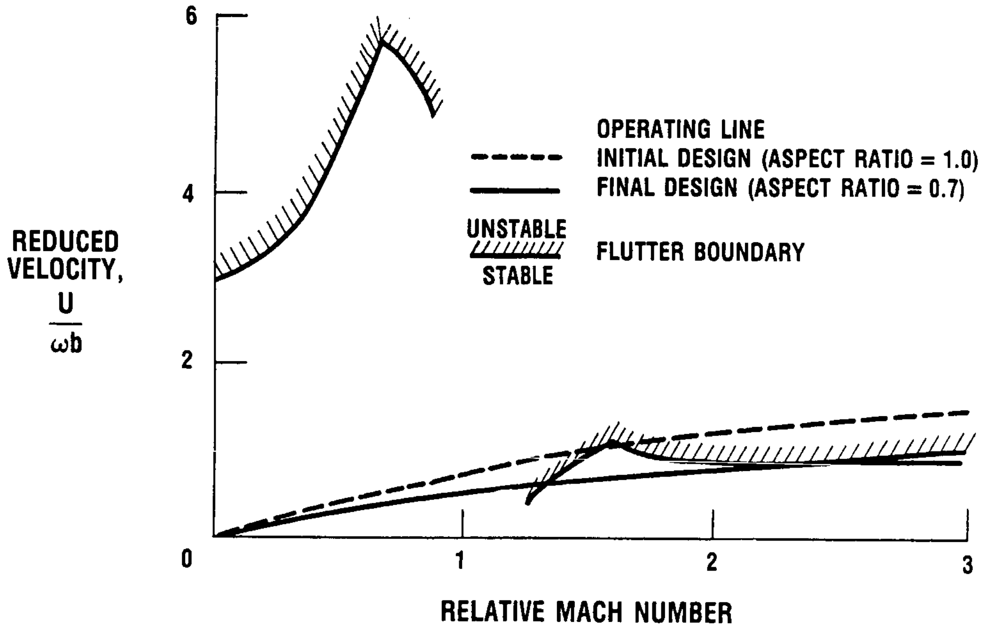
FLUTTER ANALYSIS

The flutter analysis was performed by incorporating this unsteady aerodynamic code into an existing aeroelastic code that solves the stability problem. The aeroelastic code was then applied to the NASA through-flow fan blade (Kielb and Ramsey, 1988). The analysis predicted that the through-flow fan would be torsionally unstable at supersonic relative velocities. As a result, the blade aspect ratio was reduced in the final design to bring the rotor into the stable operating range.

TORSIONAL FLUTTER



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SUMMARY

Lane's (1957) analytical formulation, of the unsteady pressure distribution on an oscillating two-dimensional flat plate cascade in supersonic axial flow, has been developed into a computer code. This unsteady aerodynamic code has shown good agreement with other published data. This code has also been incorporated into an existing aeroelastic code to analyze the NASA Lewis supersonic through-flow fan design. A more sophisticated aerodynamic model that takes into account blade camber and/or thickness is being considered as a follow-on to this work.

- **LANE'S (1957) FORMULATION HAS BEEN DEVELOPED INTO AN UNSTEADY AERODYNAMIC CODE**
- **THE UNSTEADY AERODYNAMIC CODE HAS SHOWN GOOD AGREEMENT WITH PREVIOUSLY PUBLISHED DATA**
- **THE UNSTEADY AERODYNAMIC CODE HAS BEEN INCORPORATED INTO AN AEROELASTIC CODE**
- **AN UNSTEADY AERODYNAMIC MODEL THAT INCLUDES THICKNESS AND/OR CAMBER EFFECTS IS BEING CONSIDERED FOR FUTURE WORK**

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APPENDIX - SYMBOLS

A	$2By_1$
a_0	speed of sound
a_e	elastic axis position
B	$\sqrt{M^2 - 1}$
b	semi-chord
c	chord
$(C_L)_{plung}$	lift coefficient due to plunging motion
$(C_L)_{tors}$	lift coefficient due to pitching motion
$(C_M)_{plung}$	moment coefficient due to plunging motion
$(C_M)_{tors}$	moment coefficient due to pitching motion
h	plunging displacement
i	imaginary unit
IM()	imaginary part of ()
J_0	Bessel function of the first kind of order 0
K	reduced frequency based on chord, $\omega c/U$
K_h	bending stiffness
K_α	torsional stiffness
L	aerodynamic lift
M	Mach number
$\bar{M}$	aerodynamic moment
P_-	pressure on lower surface of airfoil
P_+	pressure on upper surface of airfoil
RE()	real part of ()
s	blade spacing
t	time
U	free-stream velocity
x	streamwise coordinate

x_0	x/c coordinate of pitching axis with respect to the leading edge
y	transverse coordinate
α	complex amplitude of incidence
ϵ_n	$\epsilon_n = 1$ if $n = 0$; $\epsilon_n = 2$ if $n \geq 1$
ζ	dummy variable of integration
κ	$KM/B^2 = \omega c/B^2 a_0$
ρ	air density at free stream
σ	interblade phase angle
Υ	complex amplitude of dimensionless pressure difference
Ω	$\sigma + 2\kappa M x_1$
ω	angular frequency
1[ ]	unit step function

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