Wind-Tunnel Sonic-Boom Testing Techniques

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A study is made of techniques that have been developed to permit wind-tunnel experiments of sonic-boom phenomena to contribute to the state of knowledge in this vital area. The problems encountered in attempting to make accurate measurements of weak pressure fields produced by the necessarily small models are described in some detail. Among the challenges presented are those associated with accurate construction of extremely small models, with nonuniform and nonsteady tunnel test conditions, with model and probe vibration, with boundary-layer effects, and with complex near-field signatures. It is shown how some of these problems have been solved, how some have been treated to minimize their importance, and how others have been handled by providing a means of compensating for their effects.

Nomenclature

P = ambient or reference pressure
ΔP = pressure increment due to model or airplane flowfield
h = lateral distance from model to probe or airplane flight altitude
l = characteristic length of model or airplane
x = distance in freestream or flight direction from model or airplane nose to point in flowfield
Δx = incremental x distance
M = Mach number
β = $(M^2 - 1)^{1/2}$
K_r = reflection factor
C_L = lift coefficient
S = wing reference area

Introduction

IT is generally recognized that the sonic boom poses one of the most important questions concerning the future of supersonic aircraft, particularly with regard to commercial air transportation. As in all other aspects of aircraft aerodynamics, the wind tunnel has an important role to play. At the NASA Langley Research Center, wind-tunnel studies of sonic-boom phenomena have been conducted since 1959. Early tests made use of simplified research models in order to study the physics of the problem[1-3]. Later tests were performed to establish the sonic-boom characteristics of specific airplane configurations.[4-5] The small size of the models and the requirement for extreme sensitivity in the measurement system led to the development of unique test apparatus and methods.

Discussion

Models

Figure 1 shows a photograph of some of the airplane models tested. These models, representative of supersonic transport and bomber configurations, are about 1 in. in over-all length. At Langley, sonic-boom wind-tunnel tests have been performed with models ranging in size from $\frac{1}{4}$ to 4 in. The choice of model size is the result of a compromise. The need for very small models that allow an approach to far-field "N" wave conditions for the measured signatures must be balanced against the more accurate airplane representation and more exact signature definition possible with relatively large models. In the Langley 4- by 4-ft supersonic pressure tunnel, the 1-in. size shown in the photograph generally permits a reasonable approach to the N shape. Of course, great care must be employed in the construction of models of this size in order to achieve an accurate scaling. Construction methods employ specialized machine tools that allow the use of oversize master models. Alignment jigs are used to a considerable extent, and a number of operations are performed under binocular microscopes.

Flowfield

The flow disturbances that wind-tunnel sonic-boom tests attempt to measure may be visualized through the use of schlieren photography. The flowfield at a Mach number of 2 for a 1-in. bomber airplane model is illustrated in Fig. 2. The model itself does not appear distinctly in the photograph because of model vibration during the 8-sec exposure. For the same reason, there is also some softness evident in the model-induced shock field. Model vibration is one of the prime difficulties encountered in tunnel sonic-boom testing. Careful examination of the photograph may reveal another potential problem area. There is barely perceptible evidence of tunnel-wall disturbances extending completely across the tunnel test section. The effect of intrusions on the model flowfield of outside influences such as this must be minimized.

Extraneous Influences in Measurement

The basic concept of the flowfield measurement is quite simple. A model of the desired configuration is string-supported at the required incidence with respect to the local flow angle, and measurements of the model-induced pressures are made at orifices drilled in the surface of a reflection plate or a slender probe. It is in execution of the plan that difficulties

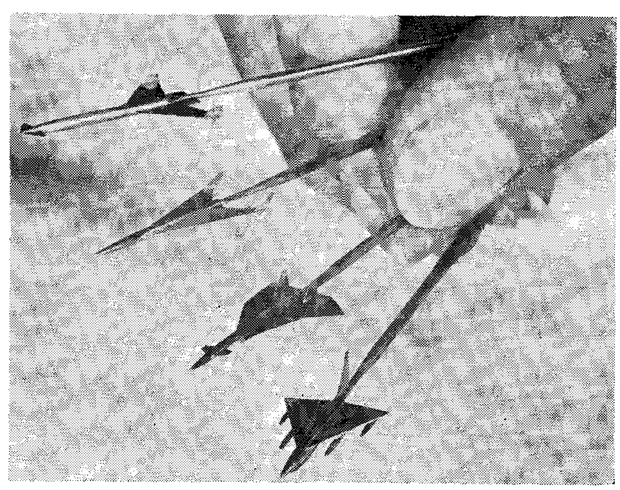

Fig. 1 Typical models.

Presented as Paper 66-765 at the AIAA Aerodynamic Testing Conference, Los Angeles, Calif., September 21-23, 1966; submitted October 6, 1966. [3.11, 10.09]

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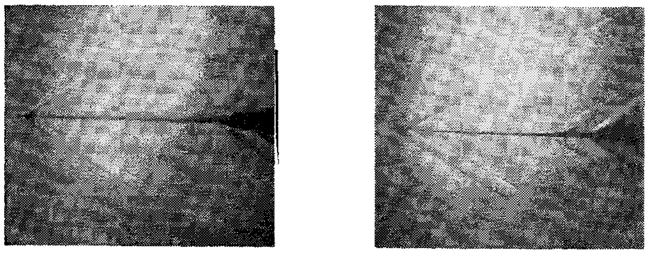

a) Top view b) Side view

Fig. 2 Schlieren of model flowfield.

arise. The first problem encountered is that of extraneous variations of pressure at the measuring orifices which must be eliminated or minimized. In Fig. 3 is illustrated the nature of some of the pressure variations found to exist in the presumably uniform and steady tunnel flow in the absence of a test model. The first plot (top of figure) illustrates the variations of measured pressure due to small tunnel-wall deformities that may be encountered as a probe or plate is moved from position to position. By normal standards the changes are small, perhaps several percent of the average freestream static pressure, and present no difficulty in the usual wind-tunnel testing. However, those same variations may be several times greater than the maximum pressures produced by the model. It is therefore obvious that sonic-boom tests may be attempted only if these variations are eliminated or at least greatly reduced by positioning the measuring orifice at a fixed tunnel location, and by moving the model itself from position to position. Of course, there are changes in the strength of the model-created disturbances as the model is moved from position to position, but these deviations are proportional to the absolute tunnel pressure levels (not the variations from nominal) and introduce little error.

With a fixed orifice location there still are problems. The necessary sensitivity of the measurement is achieved only if a differential pressure gage with a maximum range not too much greater than maximum model-created pressures is employed. Tunnel freestream static pressure is the logical reference. However, as will be evident from the following discussion, the location of the freestream static-pressure measurement and the pressure tubing arrangement must be selected with care. In attempting to maintain a constant tunnel-pressure level, the control system cycles from underpressure to overpressure, and tunnel conditions are not constant either from time to time or from position to position. These oscillations of the tunnel control system can result in extraneous variations of pressure measured by the differential gage, as shown in the second plot of Fig. 3. Again the variations can be larger than those to be measured. It has been found that relatively close locations of the reference and measuring orifices and careful balancing of time lag in the tubing can virtually eliminate this source of error. Achieving equal time lag assures that the measurement for both orifices refers to the same instant of time so that time-dependent variations are eliminated. Spacing the orifices at a distance no more than

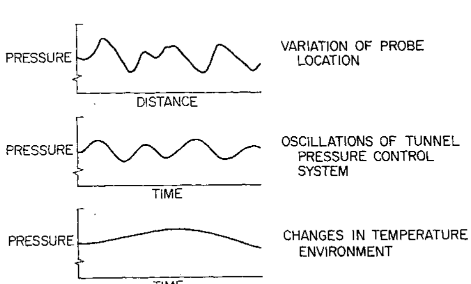

Fig. 3 Extraneous influences on measured pressure signatures.

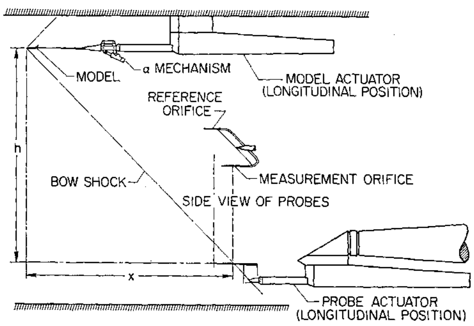

Fig. 4 Wind-tunnel apparatus.

that necessary to obtain a complete signature of the model pressure field minimizes the position-dependent variations.

The less severe, more gradual deviations of differential gage pressure with time shown in the bottom plot are the result of changes in the temperature environment of the gage and the tubing external to the tunnel. Variations of this sort were noted for some time, but the cause was not discovered until one cold winter day when it became possible to correlate pressure excursions with the opening and closing of an outside door. Apparently, because the tubing arrangement was not completely symmetrical on the two sides of the gage, the system on one side of the gage became heated or cooled more than the other side, and the pressures were correspondingly affected. Locking of doors or the application of insulation can reduce this effect to insignificance.

Test Apparatus

Figure 4 shows one arrangement of wind-tunnel test apparatus found to yield satisfactory results. The model is sting-supported by a remotely controlled actuator that permits longitudinal positioning of the model. Angle of attack may be set by a remote-control mechanism. Until quite recently, angle of attack was set by the simple expedient of bending the model sting. The set angle is selected so that with deflection under load the resultant angle of attack with respect to the surveyed flow angularity will produce the desired lift. In one instance,[6] for a research model with a fairly large body, lift was measured directly by use of a small internal strain-gage balance.

The measuring probe and reference probe assembly is also mounted on a remotely controlled actuator. This motion, coupled with the lateral motion of the permanent tunnel sting (the usual support for full-size wind-tunnel models) provides a wide range of survey positions. For measurement of a particular signature, however, the probe position is fixed, which, as mentioned before, avoids the extraneous influence of tunnel flow nonuniformities. The measuring orifice and the reference orifice are spaced in the vertical plane to avoid mutual interference and are spaced longitudinally only a distance sufficient to permit measurement of a complete signature before the forward motion of the model and its flowfield influences the reference orifice as well as the measuring orifice. A complete pressure signature is built up point by

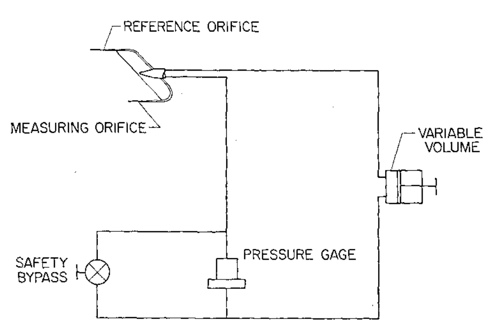

Fig. 5 Schematic diagram of pressure instrumentation.

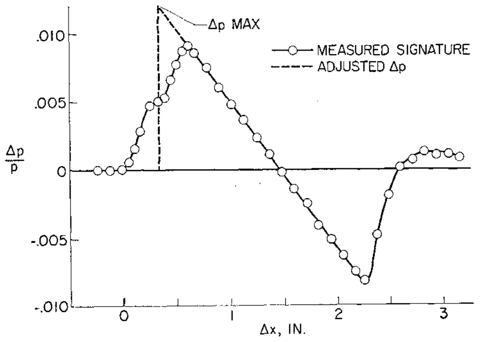

Fig. 6 Typical measured pressure signature.

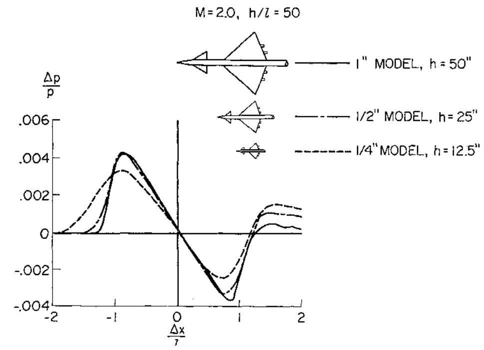

Fig. 7 Effect of model size on measured signature.

point as the model is moved to successive positions. For each position, a sufficient time is allowed for equilibrium condition to be reached, normally a matter of about 30 sec. A complete detailed signature may thus require 15–30 min of testing time. It has been common practice to obtain signatures at three widely spaced lateral stations during each run.

A schematic diagram of the pressure instrumentation is shown in Fig. 5. The differential pressure is measured by a sensitive gage with a full gage range, which may be as small as ±0.05 psi. Forces due to pressures acting on a diaphragm within the gage are transmitted to an unbonded wire strain sensing unit in the form of a Wheatstone Bridge. Output of the gage is converted to a useful form by means of a null-balancing potentiometer and a digital converter. As mentioned previously, extraneous variations in pressure due to oscillations in tunnel ambient pressure are minimized by close spacing of measurement and reference orifices and by balancing of time lag in the tubing on both sides of the gage. This balancing is accomplished by matching of tubing diameters and lengths and by installation of a variable volume device to provide an adjustment for any remaining unbalance. The volume adjustment is made during tunnel evacuation and pressurization prior to a test run. A safety valve in parallel with the gage is opened during tunnel startup and shutdown to prevent overloading.

Typical Test Results

The test apparatus and instrumentation just described provide a system that allows measurement of model flowfield pressures with little influence from extraneous sources. The experimenter's problems are not yet over, though, as may be seen in the measured signature shown in Fig. 6. The ratio of the pressure increment resulting from the model flowfield to the freestream static pressure is plotted as a function of flowfield position. It is immediately apparent that the measured signature falls far short of the theoretical N wave shape.[7] The signature for this example was deliberately chosen to accentuate departures from the N shape and thus is not typical. But departures of the type illustrated are present to some degree in all of the tunnel signatures. First of all there is a notable lack of sharply defined peaks in any part of this signature. This is attributed to vibration of the model and probe and to probe boundary-layer effects. The highly damped system will record the average pressure acting on the orifice while the entire model pressure field is moving with respect to the orifice. The tendency for the bow portion of the wave to have a double shock is characteristic of the near field. At larger distances the two shocks would merge to form a single front. In this example, the bow portion of the signature shows evidence of both near-field and vibration effects, but the tail wave evidences vibration only. Analytic studies have shown that a simple adjustment may be made to account for both effects and indicate the shape and magnitude of the signature in the absence of near-field and vibrational effects. As shown in the figure, this adjustment consists of creating an N wave signature which preserves the positive and negative impulse areas of the measured signature and which follows the linear portion of the measured signature. A practical test of the applicability of the adjustment is met where adjusted bow shock pressure rise from signature obtained at various distances varies as the inverse of the $\frac{3}{4}$ power of the distance. The $\frac{3}{4}$ power law of the Whitham theory[7] has been verified experimentally in measurement of sharp peaked N waves from projectiles.

The wind-tunnel data of Fig. 7 provide an extreme example of the influence of vibration or signature shape. A comparison is made of signatures obtained at equal nondimensionalized distances from models of a supersonic transport configuration differing only in scale. Theoretically the signatures should be identical and should display sharp peaks. The rounding of peaks and spreading of the signature are believed to be due in large part to model and probe vibration. Since much of the vibration is associated with the support system behind the model, the model vibration does not increase in proportion to model size but tends to remain constant in amplitude. The larger influence of vibration relative to the $\frac{1}{4}$-in. model is readily apparent in the lengthened signature. Adjusted signatures for the 1- and $\frac{1}{2}$-in. models would be nearly identical, but for the smaller model, vibration has such a relative magnitude that the linear portion of the signature is affected and the adjustment is no longer valid. For that reason and because of construction difficulties, no further models of the $\frac{1}{4}$-in. size were considered.

Comparison with Flight Measurement

By employment of scaling laws established by the theory, it is possible to make direct comparisions of signatures measured in tunnel tests with those measured in flight. An example of this comparision is shown in Fig. 8. A signature for a 1-in.-long model is compared with measurements for a 100-ft-long bomber airplane. The signature parameters account for the significant differences in size, distance or altitude, ambient

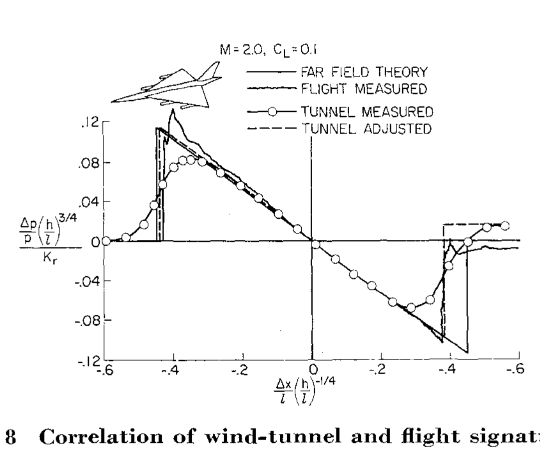

Fig. 8 Correlation of wind-tunnel and flight signatures.

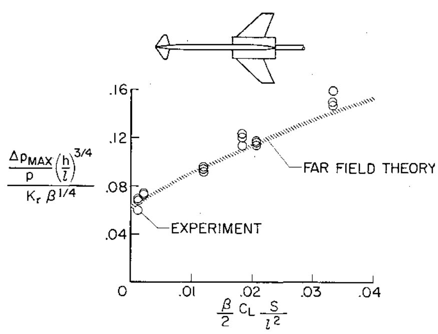

Fig. 9 Tunnel data in parametric form.

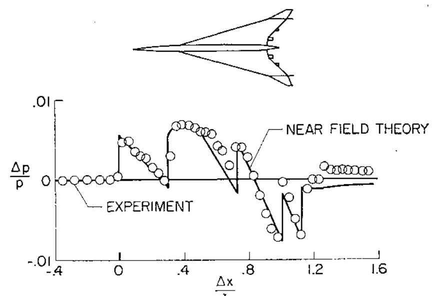

Fig. 10 Near-field signature.

pressure, and reflection factor. In addition to the geometric similarity of airplane and model, it is also necessary for a direct comparision that the lift coefficient be matched. The signatures shown here were obtained for similar lifting conditions. The tunnel data, when adjusted, are found to agree well with the flight measurements. Flight data[8] are shown for a representative signature measured by one of the microphones spaced along the airplane ground track. Except in the vicinity of the tail wave, the adjusted tunnel signature and the flight signature both agree quite well with a numerical evaluation[5] of far-field theory utilizing the equivalent body concept.[9]

Data Presentation

A parametric form of presenting wind-tunnel and theoretical sonic-boom data which has proved useful in estimating airplane ground track bow shock pressures is shown in Fig. 9. An overpressure parameter is plotted as a function of a lift parameter. In this form, a single curve is sufficient to represent the theoretical far-field characteristics of a given configuration for a given Mach number. The β term included in the parameters accounts for all the Mach number effects except for variations in equivalent body shape as determined by supersonic area rule cutting planes, and thus the single curve is also a reasonable approximation for a rather broad speed range. In this figure, maximum overpressures determined from adjusted wind-tunnel pressure signatures are compared with theoretical estimates. Data are shown for Mach numbers of 1.4 and 2.0 and for distances ranging from 12.5 to 50 model lengths. Either the experimental or theoretical data may be used for estimates of airplane ground track overpressures when the proper quantities are substituted in the parameters. However, there are some precautions that must be exercised in the extrapolation of the tunnel data. The model truly represents the airplane only if all components contributing to the bow shock are accurately scaled and only if model and airplane boundary-layer thickness are also in the same scale. Boundary-layer similarity is seldom attained, and a correction based on the theory is usually necessary. Because of these scaling difficulties, extrapolated tunnel measurements have not offered any advantages over the use of estimates based purely on theory. The primary service of the tunnel has been to verify the applicability of the theory to a wide range of specific configurations, and thus establish confidence in its use for flight estimates.

Near-Field Signatures

Having recognized the problems associated with extremely small models, the construction inaccuracies, the vibration, and the boundary-layer effect, an attempt to utilize larger models has been made in more recent tests. Of course, the larger models, although reducing the influence of the effects just mentioned, only increase departures from a simple N wave as a result of increased near-field effects. For wind-tunnel tests of these larger models to yield meaningful results, it was necessary to develop a numerical means of implementing the near-field theory[10] which would then serve as a guide in extrapolation to large distances. In Fig. 10 is shown a measured signature for a 4-in.-long model of a supersonic transport configuration. The near-field theory numerical solution[11] is seen to be in good agreement with the measured signature. This theory indicates that for this configuration, the near-field effects persist to extremely large distances. For instance, at an altitude of 40,000 ft and a Mach number of 1.4, a near-field-type of signature is expected to extend to the ground where maximum overpressure would be about 10% less than given by the far-field theory. Tests of models of this size have shown the sensitivity of near-field signature shape to small changes in model geometry and have served to prove the applicability of current methods of estimation[12] and optimization.[13]

Conclusions

It has been seen that wind-tunnel sonic-boom testing is no easy matter. Among the challenges presented are those associated with accurate construction of extremely small models, with nonuniform and nonsteady tunnel test conditions, with model and probe vibration, with boundary-layer effects, and with complex near-field signatures. This paper has attempted to show how some of these problems have been solved, how some have been treated to minimize their importance, and how others have been handled by providing a means of compensating for their effects. Application of the test methods described herein has enabled the wind tunnel to make valuable contributions toward the understanding of sonic-boom generation and propagation.

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MAY-JUNE 1967 J. AIRCRAFT VOL. 4, NO. 3

Limits on Minimum-Speed V/STOL Wind-Tunnel Tests

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This paper presents the results of a systematic series of wind-tunnel tests, which have determined the maximum size rotor that can be tested in closed-throat wind tunnels both as a function of the downwash angle and as a function of tunnel geometry. For a given size rotor and tunnel there appears to be a maximum value of downwash that can be tolerated. If this value of downwash is exceeded, the flow through the wind tunnel is no longer similar to the flow that would be encountered in free flight but rather represents a flow similar to recirculation. The point at which the maximum downwash is reached is called the flow breakdown point. Similar results have also been obtained using jet flaps and jet-lift models. It is also shown that this flow breakdown is a function of tunnel geometry and that the allowable downwash angles are different for rectangular tunnels with width-to-height ratios of W/H = 1.50, 1.00, 0.67, and 0.50. The addition of fillets to the test section is also shown to have an adverse effect on the allowable downwash angle. At the present time, the optimum tunnel configuration for rotors and other types of V/STOL vehicles is not known.

Nomenclature

A_m = momentum area of lifting system
A_T = cross-sectional area of wind tunnel
C = cross-sectional area of wind tunnel
C_L = coefficient of lift L/qS
H = height of wind tunnel
L = lift
q = dynamic pressure
R = rotor radius
S = wing area
V = velocity
W = width of wind tunnel
α = angle of attack
δ = wind-tunnel or jet boundary correction factor
θ_n = momentum downwash angle or wake deflection angle measured from horizontal axis
μ = advance ratio or tip speed ratio $V/\Omega R$
Ω = rotor angular velocity

Introduction

In June 1963, the University of Washington started a research program supported by the Engineering Sciences Division of the U. S. Army Research Office Durham, Durham, N. C., to study operational and testing problems that might be encountered in testing V/STOL type of aircraft in wind tunnels.

The testing of V/STOL models in wind tunnels presents many problems that are very different from those encountered in the testing of conventional-type aircraft where the testing techniques are relatively well understood. One of the basic differences between the two types of aircraft is, of course, the method of generating lift. A conventional aircraft obtains its lift from a wing or wings which may be characterized by moderate values of lift coefficients and relatively low angles of downwash in the flow behind the wings. A V/STOL vehicle, on the other hand, at speeds near transition flight, has large values of lift coefficients and downwash angles that vary from 90° at hover to 3°–6° at high forward speed. This large wake deflection angle with high energy added to it appears to present one of the most difficult problems that is encountered in wind-tunnel testing of V/STOL type of vehicles.

In an attempt to gain an insight into the problems of testing V/STOL type vehicles in a wind tunnel, a series of tests were run using as a vehicle a helicopter rotor. No attempt was made, however, to duplicate any specific type of rotor, but, rather, the rotor was only considered as a source of downwash. In these tests two rotors were used: 1) a four-bladed, 38-in.-diam rotor with articulated blades and 2) a three-bladed rotor with a 24-in. diam. These rotors were tested at identical operating conditions in the main 8- × 12-ft test section of the University of Washington wind tunnel and in inserts set inside the main test section.

The inserts are plywood boxes open on the ends in the direction of the wind stream and are used to simulate the ceiling, floor, and walls of wind-tunnel test sections of various geometries. The 38-in. rotor was tested at a nominal disk loading of approximately 4 psf and the 24-in. rotor at nominal disk loadings of approximately 4, 7, and 10 psf. The disk loading is defined as the lift of the rotor divided by the swept area of the rotor or the rotor disk area. Both rotors were tested at a

Presented as Preprint 66-736 at the AIAA Aerodynamic Testing Conference, Los Angeles, Calif., September 21–23, 1966; submitted September 30, 1966; revision received January 31, 1967. This work was sponsored by the Army Research Office, Durham under Grant DA-ARO(D)-31-124-G481. [3.01]

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