THE STATUS OF ROTOR NOISE TECHNOLOGY

ONE MAN'S OPINION

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SUMMARY

In the last two decades, the somewhat "black art" of rotor noise prediction has grown into a science that might be called Rotor Noise Technology. This transformation has been due to many reasons, not the least of which has been the growing interest of the aerodynamicist in rotor acoustics. This paper will approach the problem of establishing the state of the "technology" by first identifying the various characteristics of rotor noise and then assessing the state of technology in understanding and predicting the most important of these rotor noise characteristics in a real-world environment.

INTRODUCTION

On the basis of experience gained in investigating propeller noise (i.e., refs. 1 to 6), some basic aeroacoustic investigations were conducted on the mechanisms associated with helicopter rotor noise prior to 1960. Most of these investigations, however, were centered around defining the characteristics of helicopter noise and evaluating the effects of basic helicopter parameters on these characteristics (i.e., refs. 7 and 8). These investigations determined the effects of basic rotor parameters such as blade number, disk loading, tip speed, blade chord, and forward flight velocity on the noise output of helicopter rotor systems. The results of these studies illustrated that the best way to reduce rotor noise is to reduce the rotor tip speed and disk loading and increase the number of blades. Many of these early investigations were prompted by the thought that if the other noise sources, such as engine, gear boxes, accessories, etc., could be reduced, then rotor noise would become the primary noise source of helicopters and, therefore, means of controlling rotor noise should be investigated.

As these investigations continued into the sixties, they were expanded such that the origin of the various sources of rotor noise and ways of reducing these sources of noise were considered (i.e., refs. 9 to 12). Although these studies were more directed toward the measurement of the effects of various parameters on the noise being generated by a rotor system, they began to highlight the effects of various aerodynamic parameters as the sources of rotor noise. As it became apparent that, due to the rapid advances being made in the development of new helicopter configurations because of the extensive application of gas turbines, a detailed understanding of the aerodynamic forces associated with the various sources of rotor noise must be obtained if adequate methods of predicting rotor noise in a variety of flight conditions were to be developed. Because of this need, the aerodynamicist has become increasingly involved in the exciting and challenging research associated with the development of satisfactory rotor noise prediction techniques. The evolution of the understanding and the development of rotor noise prediction technology over the last decade is the subject of the review that is presented herein.

Because this symposium is basically establishing the current "State of Rotor Noise Technology," it would be inappropriate for this review paper to microscopically examine the technology of predicting the noise output of various sources. This task has been relegated to the session reviewers. The review presented herein is much more general in nature and tries to highlight the state of the technology in understanding and predicting the noise characteristics of rotors in the real-world environment of helicopter flight.

SYMBOLS

b	span of reflection plane model, meters
C_T	coefficient of thrust
c	chord of reflection plane model, meters
c_{MR}	chord of main rotor, meters
c_p	coefficient of pressure
Δp	differential pressure, newtons per centimeter^2
q	dynamic pressure, newtons per centimeter^2
R	radius of tail rotor, meters
T_o	disk loading, newtons per meter^2
V	velocity, meters per second
V_D	velocity of descent, meters per minute
V_F	free-stream velocity, meters per second
V_T	velocity of translation, meters per second
x	chordwise location of reference point on reflection plane model, meters
y	spanwise position of reference point on reflection plane model, meters

α angle of attack, degrees

λ sweep angle of leading edge with respect to relative airstream, degrees

μ advance ratio

ψ azimuth angle, degrees

Ω_{MR} rotational speed of main rotor, radians per second

Ω_{TR} rotational speed of tail rotor, radians per second

TECHNICAL DISCUSSION

In considering the noise generated by helicopter rotors operating in a real-world environment, it is obvious that the helicopter configuration plays an important role in the rotor noise signature. Even to the casual observer, it is obvious, for example, that the noise produced by a UH-1, a CH-47, or a CH-53 is quite different. While these differences can be caused by many configuration parameters, such as disk loading, blade number, number of rotors, and blade tip speed, the basic sources of noise are all present in varying amounts for each configuration. The degree to which each noise source contributes to the overall noise signature of a helicopter depends upon the helicopter and rotor configuration.

The picture of a CH-53E shown in figure 1 can be utilized to point out some of the real-world environmental effects that strongly influence the noise generated by rotors operating in a forward flight on a realistic helicopter. The CH-53E was not chosen as an example because of its noise characteristics but because it was a good picture of a helicopter in flight that could be used to point out the environmental effects of interest that will be discussed in further detail herein.

In viewing the picture of the CH-53E, it can be visualized that there are many free and self-induced environmental effects that can affect the noise characteristics of the blades in the main rotor. At least the following are of primary importance:

(a) The aerodynamic turbulence in the free stream through which the rotor flies

(b) The aerodynamic compressibility effects below the critical Mach number on the advancing blade velocities

(c) Shock waves generated by airflow above the critical Mach number near $\psi = 90^{\circ}$

(d) Separated flows generated by high angles of attack on the retreating side of the rotor disk

(e) The concentrated vortex flows generated at the blade tips

(f) The highly turbulent flow field induced by the complete rotor wake

(g) The effects of fuselage blockage and separated flow

The tail rotor has all of these same environmental effects induced by its own blades as well as those which the concentrated and general nonuniform wake of the main rotor induce when it interacts with the fin and blades of the tail rotor. When the noise generated by the rotor blades operating in this type of aerodynamic environment is measured, a spectrum similar to the generalized spectrum shown in figure 2 is obtained. For the sake of discussion, the spectrum has been separated into three different categories of noise:

(a) Noise due to steady loads

(b) Noise due to unsteady loads

(c) Noise due to incoherent or random loads

The noise due to the steady loads are related to the integrated torque and thrust forces developed by the rotor system to maintain flight. The basic characteristics of the noise produced by the steady rotating loads were predicted by Gutin many years ago (ref. 13). References 2, 4, 5, 14, 15, and 16 outline improvements made to Gutin's basic theory to account for noise in the near field, thickness noise, and far-field distortion due to source translational motion. Reference 17 was an early attempt to extend Gutin's basic theory to remove its inherent limitation when applied to helicopter rotors. Since then there have been numerous investigations to improve the prediction of the rotational noise produced by the steady rotating loads which has resulted in theories which can adequately predict the primary characteristics of this type of noise. The noise labeled incoherent noises are nonperiodic noises that are generally related to the viscosity effect of the air and are due to such phenomena as inflow turbulence, boundary-layer effects, separated flows, and vortex shedding. Wright (ref. 18) has aptly referred to this noise as "self-noise." These two noise sources, noise due to the steady loadings and self-noise, are believed to be unavoidable when operating a helicopter and thus may be considered to be the lower limits to which helicopter noise might be lowered. The third source of noise that has been listed in figure 2 is that due to unsteady loads which are generated by the previously noted real-world environmental effect in which helicopter rotors must operate. Since these noise sources do not arise from the loads needed to fly the helicopter, some authors have labeled the category of noise as "excess noise" (ref. 19). Since this type of noise generally is the major contributor to the noise in the frequency spectrum of interest to this symposium, i.e., annoyance, detectability, etc., establishing the status of the technology in understanding, predicting, and modifying this category of noise is the one to which this review will be directed.

The excess noise sources can result from loadings having frequencies in the noted range or from a loading impulse that happens only over a short time interval. The shorter the time period, the greater is the number of harmonics of noise that is produced. While the various types of loadings that result in the excess noise consists of both types of noise sources, the majority of the noise is due to the harmonics of loading impulses which occur one or more times in the azimuth. The major sources of rotor noise which contribute to the excess noise will be discussed in some detail in the following sections of this review in an attempt to establish the state of the technology in understanding, predicting, and modifying these sources of noise.

Blade Stall

Blade stall can occur in hover around the entire azimuth when the blade angle is increased to sufficiently large values or over relatively small portions of the azimuth in forward flight. Since the blade stall angle of attack is strongly affected by the relative Mach number, stalling can occur over the advancing side of the rotor disk for highly loaded rotor system in forward flight as well as over the retreating side of the rotor disk. When stall effects do occur on the advancing blade they produce a greater amount of excess noise than on the retreating side because of the higher dynamic pressure at which it occurs and the smaller increment of azimuth angle over which it occurs.

Hubbard and Maglieri, in 1958, demonstrated the large effect blade stall could have on the noise characteristics of a hovering rotor. Figure 3, taken from reference 7, presents data which show the effect of stall on the overall noise level at different rotor tip speeds as well as the effect of stall on the frequency spectrum for a rotor tip speed of 183 m/sec. In the top half of figure 3, the solid symbols represent the conditions at which the authors indicated that blade stall was present and for which the noise output was considerably greater than the conditions at which stall was not present. The spectrum shown in the bottom half of figure 3 shows how stall affects the frequency content of the noise. The authors reported that the noise presented in the spectrum did not have discrete spikes and correlated well with that which would be calculated based on the experimentally determined Strouhal numbers. The authors suggested, therefore, that the noise due to stall was probably due to vortex shedding from the blades. Schlegel, et al., reported in reference 17 that the noise generated by small pockets of stall on the retreating blade had signature characteristics similar to that of impulsive noise and was believed to be associated with the modulation of high frequency loadings due to discrete vortex shedding. These results indicate that blade stall might produce a significant amount of noise in the higher rotor frequencies, particularly if it occurs over just a small portion of the azimuth. As noted by the authors of the referenced investigations, the noise produced during stall may be associated with discrete vortex shedding on a continuous basis or on a modulated basis during blade vortex interactions on the retreating side.

While the referenced investigations were conducted a number of years ago, it is believed that the understanding of the noise produced by the aerodynamic forces generated during stall has not increased markedly since that time. This lack of an intensive effort to predict stall-induced loadings and associated noise has probably been due to the lack of a suitable theory to realistically predict stall-induced loadings and because of the need to reduce the more dominant excess noise sources caused by blade vortex interactions and unsteady potential airloads. It is believed, however, that because of the potentially stronger aerodynamic interaction between the rotor and fuselage that is possible with the newer helicopter configurations that stall-induced noise may have more significance in the future, particularly if the noise produced by some of the more dominant excess loading sources is reduced. If this type of noise source does become of more importance in the future, its prediction will be a very difficult undertaking if it is attempted by other than empirical or semi-empirical methods. This is believed to be the state of the technology as the effort that has been directed towards predicting the dynamic stall characteristic of rotor blades in forward flight has not produced a reliable and useful prediction technique. It is reasoned, therefore, that the true prediction of stall-induced noise will be very difficult and will require a reasonable amount of additional research effort.

Compressibility Effects

I have chosen to separate those effects which are due to drag divergence and those which are due to the aerodynamic mass acceleration around a solid airfoil section at higher Mach numbers commonly referred to as thickness noise. The Mach number at which drag divergence occurs is a fairly strong function of angle of attack, i.e., the higher the angle of attack the lower the drag divergence Mach number. For rotors in which there is not a strong blade vortex interaction, the increased noise due to drag divergence effects usually occurs in the rotor azimuth range of 10° to 80°. Since the compressibility effects due to drag divergence are pronounced only over a small azimuth range, it would be expected to produce a significant amount of noise at the higher harmonics of blade passage. Arndt and Borgman tend to confirm this conclusion as they showed in reference 20 that including the effects of drag divergence in the prediction of rotational noise increased the noise significantly in the higher harmonics of blade passage. Comparison of predicted results with experimental data presented by Cox in references 21 and 22 also showed that the inclusion of drag divergence effects significantly improved the correlation between the predicted and experimental results in the frequency range of excess noise.

Figures 4 and 5, taken from reference 23, present the angle of attack and Mach number contours, respectively, determined for a UH-2 flying at an advance ratio of 0.48. It is noted that, in the azimuth range of 10° to 50°, the data indicate that the compressibility boundary moves inward leaving the outer 20 percent of the blade radius operating above the critical Mach number. This inward movement of the compressibility boundary is due to the increase in the angle of attack and relative velocity over the same range of azimuth angles. With 20 percent of the blade radius operating above the critical Mach number it would be expected that the impulsive increase of drag loading would generate a significant pressure wave in the plane of the rotor. Since the movement of the compressibility boundary happens only over a small portion of the azimuth, it would be expected that higher harmonics of rotational noise would be produced. Increasing the rotor speed, flight speed, or the rotor thrust increases the angle of attack and velocity over a larger region of the advancing side of the rotor disk and, therefore, the noise due to drag divergence would increase in intensity and be noticeable over a larger range of frequencies.

With the advances that have been made in airfoil designs for rotor blades, the problem of noise due to compressible drag divergence effects has decreased. In addition, since most modern rotor noise prediction techniques include the effects of Mach number in the definition of the airfoil sectional characteristics, the prediction of the noise due to compressibility effects can be readily handled if the angle of attack and velocity distributions over the disk are known. Unfortunately, this knowledge cannot be gained experimentally and can only be gained by the utilization of predictive free rotor wake flow analyses such as that presented in reference 24 and extensively expanded since that time. While direct correlation between theory and measured angle of attack distribution cannot be made, correlation of harmonic loadings indicates that relatively good predictions of the radial and azimuthal distribution of angle of attack can be obtained using such analyses techniques. It is believed, therefore, that the understanding of the effects of compressibility and the associated noise produced by helicopter rotors in forward flight is on firm ground and that means of predicting the effects of various real-world parameters on this noise source are available if the existing analyses procedures are properly utilized.

Rotor Noise Resulting From Blade Vibration

Prior to discussing other primary sources of excess noise because of pressure pulses at higher harmonics of blade passage frequencies due to discrete happenings in the azimuth, a brief discussion of the noise that can be generated by the structural vibration of rotors will be presented. This type of excess noise occurs at the frequency of the motion and not at higher harmonics of discrete impulsive pulses. It is believed pertinent to discuss this possible problem at this point in the review as much of what will be presented in the following portions of the review is associated with model tests. While some investigators in the past have postulated that higher harmonic blade vibration of full-scale rotor systems could affect the noise signatures in the higher frequency spectrum, no strong evidence of this type of noise source has been found for full-scale rotors. For small scaled models of full-scale systems, however, this source of noise may be of significance due to the higher structural frequencies of the scaled models. A recent experience, during wind-tunnel tests of small remotely piloted vehicle (RPV) propeller blades, reminded me of the possible contamination of rotor noise data due to structural vibration. Figure 6 shows a photograph of an RPV propeller blade that generated a significant noise due to the near coalescence of the third flapwise bending frequency with the 8/rev harmonic of rotational speed. Figure 7 presents the spectrum of the noise measured for this configuration when it was operating near the resonant condition. As can be seen from the data presented in this figure, the noise produced by the blade bending vibration dominated the other sources of aerodynamic noise generated by the propeller. While it was obvious that the noise produced by blade vibration had contaminated the noise signature, it might not be so obvious for model rotor systems that have higher damping in the bending modes and for conditions that are not as close to a resonant condition as they were for the propeller blades. In order to prevent the contamination of the noise signature produced by aerodynamic forces under investigation by that produced by structural vibration, it is suggested that the vibration and stability characteristics of model rotor systems be determined prior to the collection of model noise data to insure that no unwanted structural or aeroelastic motions and the associated noise due to these motions are present at the test conditions.

Rotor Impulsive Noise

As many investigators have said time and time again "impulsive noise is one of the most annoying and easily detectable sounds a helicopter can generate and when it occurs, it is the dominant source of noise." As such, it is also one of the more challenging and exciting areas of research in helicopter noise as attested to the large number of research efforts that have been conducted and reported by investigators in universities, industry, and government research organizations throughout the world. Impulsive noise can be generated by many sources and for the purposes of this review the discussion will be divided into two different general areas of impulsive noise as indicated in figure 8. One area is high-speed impulsive noise and the other is blade vortex interaction. While these areas of impulsive noise have been somewhat arbitrarily separated in this manner for purposes of discussion, it has been shown for some configurations that the two areas of impulsive noise, shown separated in figure 8, are in fact connected.

High-Speed Impulsive Noise

High-speed impulsive noise has been attributed to intense compressibility effects on the advancing blade of a helicopter in high-speed forward flight when the advancing tip Mach number approaches or exceeds unity. In the past, there have been some differences of opinion as to the major source of this noise These differences are the result of the limitations and difficulties of making suitable acoustic measurements which have restricted the evaluation of the noise source to qualitative observations. Test data, which are obtained during aircraft flyovers with ground-based microphones (i.e., refs. 25 and 26), are difficult to assess on a quantitative basis due to uncertainties in the retarded time effects, the acoustic transmission path, and ground reflection effects. Data obtained from tests conducted in conventional wind tunnels may have serious limitations, as regards its quantitative value, because of reverberation effects and high ambient operational noise levels. Another approach that has been utilized to obtain inflight noise data is to place microphones on the exterior of an aircraft (i.e., refs. 27 to 29). This technique is somewhat limited in that the microphone placement is, by necessity, limited to the helicopter's low to mid frequency acoustic near field and thus, it can be difficult to quantitatively assess how much of the noise actually radiates to the far field. In addition, to obtain directivity patterns for noise sources that radiate in the tip path plane, the use of inflight microphones attached to the vehicle generating the noise is extremely difficult, if not impossible.

In order to surmount the above noted difficulties and limitations, Schmitz and Boxwell (ref. 30) developed a rather unique inflight far-field measurement technique to obtain quantitative data of the effects of various flight parameters on the high-speed impulsive noise source. Figure 9 presents a schematic of the inflight far-field measurement technique. This measurement technique utilizes a quiet fixed-wing aircraft, instrumented with a microphone, and flown to maintain fixed relative positions with a helicopter. Because impulsive noise is thought to have its maximum intensity of radiation in the general direction of forward flight, the microphone was installed on the tail of the monitoring fixed-wing aircraft which is flown in front of the helicopter as illustrated. Estimated values of microphone wind noise and monitoring aircraft noise levels indicated that with the proper choice of a fixed-wing aircraft, the periodic phenomenon of helicopter high-speed impulsive noise could be quantitatively measured. By using this testing procedure, acoustic far-field impulsive noise radiation patterns have been obtained by Schmitz and Boxwell for a wide range of steady operating conditions.

The flight test envelope over which data have been obtained for a UH-1H using this technique is indicated in figure 8 and a sample of the high-speed impulsive noise data obtained is presented in figure 10. The data presented in this figure were averaged 128 times to eliminate the slight variability due to small blade differences as well as to eliminate the random background noise which had an amplitude less than 1/10 of the smallest of the primary pressure pulse.

As can be seen from the data presented in figure 10, the negative pressure spike, due to intense compressibility effects, dominates the noise signature and the amplitude of the spikes is a stronger function of forward speed than it is of descent rate. The authors of reference 30 noted that the noise associated with the large negative pressure peaks shown in figure 10 is rich in low frequency harmonics (10 to 300 Hz) and radiates not only near the tip path plane of the rotor but over wide azimuth angles in the general direction of forward flight. In addition, they noted that the extremely rapid increase in pressure which closely follows the negative pressure disturbance, forming a sawtooth-shaped pressure pulse with some apparent overshoot at high airspeeds (fig. 10), dominates the middle and high frequency harmonics (above 300 Hz) and radiates within narrow azimuth angles in the direction of forward flight near the tip path plane of the rotor. Using the flight test data presented in reference 30 as a basis for evaluation, Schmitz, Boxwell, and Vause (ref. 31) showed that, through careful testing in an acoustically lined wind tunnel, the high-speed impulsive noise characteristics of the full-scale flight vehicle could be duplicated by using appropriately scaled models. Having established the scaling and modeling technique required to duplicate the high-speed impulsive noise developed by full-scale blades with the use of scaled models, it is believed that the more detailed investigation of the source(s) of high-speed impulsive noise using advanced flow measurement techniques, such as schlieren photography and laser velocimeters, can be conducted with confidence.

It is believed that the data obtained by Schmitz and coworkers in their pioneering efforts to quantify the characteristics of high-speed impulsive noise (refs. 30 and 31) show the character and dominance of high-speed impulsive noise and provide an excellent quantitative data base for analyzing this type of noise source being generated by helicopter rotor blades operating in a real-world environment.

Tangler (ref. 32) has presented some interesting insight into the formation of the shock wave associated with high-speed impulsive noise. Through the use of schlieren photograph techniques he has shown that the shock waves formed on the upper and lower surface of the blade in an azimuth angle range 50° to 90° leave the blade as the relative velocity decreases ($\psi = 90^\circ$ to 150°), coalesce into a unified shock front, and propagate at an azimuth angle approximately 20° to the flight path. Tangler reasoned that the more rapid compression and apparent overshoot of the pressure wave measured in the far field and noted by Schmitz and Boxwell in reference 30 are due to the strong crescent shock wave that is formed and propagated forward. (See fig. 10.) Tangler also noted, on the basis of data obtained during an extensive wind-tunnel test program, that blade thickness was a significant parameter in high-speed impulsive noise and that blade thrust had a significant effect on the directivity pattern of the noise that is propagated.

On the basis of the measurements that have been taken and analyzed during the above noted investigations, which represent the present state of experimental technology in this area of research, it is believed that considerable knowledge has been obtained as regards an understanding of the physical characteristics of the aerodynamic flows associated with high-speed impulsive noise. It is obvious, however, that, while some effort has been directed towards determining the effect of blade parameters on the shock and noise characteristics of high-speed impulsive noise (refs. 31 and 32), much more needs to be done in this area to further our understanding and to provide quantitative data in support of the development of adequate prediction techniques.

While improvements in experimental and measurement techniques have led to significant advances in the investigation of the flow physics associated with high-speed impulsive noise over the last few years, theoretical means of predicting this dominant source of noise has also recently received much attention. The papers presented at this meeting, if nothing else, attest to the significant effort that is being devoted to this subject area. Although a great deal of theoretical effort is now being directed toward the prediction of high-speed impulsive noise, a considerable amount of theoretical research effort has been conducted over a number of years in this area. As early as 1933, Deming (ref. 33) looked into the effects of blade thickness on radiated noise. Lyon (ref. 34) represented the thickness noise using monopoles and used dipoles to represent the force noise. With these representatives he replaced the blade by a progression of accelerating "torpedoes." Using this rather unique approach he found that monopole thickness effects may be important at advancing tip Mach numbers near unity. Arndt and Borgman (ref. 20) related the high-speed impulsive noise to the drag divergence phenomenon at high advancing Mach numbers. Although they did indicate that the high-speed impulsive noise could dominate the lower frequency spectrum (up to 300 Hz), their results did not correlate well with experimental data.

It was not until 1969 when Ffowcs Williams and Hawkings rederived the classical acoustic equations for bodies moving at high Mach numbers and emphasized the noncompactness of the problem (ref. 35) that the basic theoretical formulation for studying high-speed impulsive noise was formed. Much of the recent analysis effort has been centered about this basic formulation.

Farassat (ref. 36), Hawkings and Lowson (ref. 37), and Isom (ref. 38) applied the Ffowcs Williams and Hawkings formulation to the high tip speed rotor problem using noncompact monopole terms to represent thickness and distributed dipoles to represent the localized pressure. Lowson (ref. 39) working in the frequency domain and comparing results with the data of reference 30 reported agreement with experimental data within 3 to 6 dB for a number of measurement points.

Schmitz and Yu (ref. 40) recently used monopoles to represent thickness effects and dipoles to represent local forces and obtained results similar to Lowson. The authors also included the effect of quadrupole sources as acoustic radiators in an attempt to improve the correlation between theory and experiment. While they showed that quadrupole radiation did improve the correlation with experimental data, it was not the reason for the almost 2/1 discrepancy between the measured and predicted pressures. An important result of the work presented in reference 40 was that a relatively simple and easy to use acoustic model can be utilized to conduct a numerical evaluation of high-speed impulsive noise.

Farassat (ref. 36) and Farassat, Pegg, and Hilton (ref. 41) have compared the results of Farassat's prediction technique with experimental data and some of these results are presented in figure 11. As can be seen from the results presented in figure 11, the predicted results compare rather favorably with the experimental data. It is noted, however, that for this case the experimental data do not exhibit the more rapid rise time of the positive pressure gradient reported in references 30 and 32. Since this disparity in the pressure pulse has been shown to be of importance to high-speed impulsive noise, it would be of interest to determine if Farassat's theory adequately predicts the impulsive noise when these measured characteristics were present. Farassat utilized his prediction technique to investigate the effects of the airfoil profile on the characteristics of high-speed impulsive noise. The results of these predictions which are also presented in reference 41 are shown in figure 12. As can be seen, airfoil profile was shown to have a significant effect on the negative pressure pulse associated with high-speed impulsive noise as the supercritical airfoil had a pressure peak almost twice that of the biconvex airfoil.

On the basis of the review of recent theoretical efforts directed toward the prediction of high-speed impulsive noise due to shock effects, it is concluded that great advances have been made and that the basic characteristics of this noise source are fairly well in hand. Based on the effort being applied in this area, as indicated by the number of papers presented in this meeting, one can probably look forward to reliable prediction techniques in the not too distant future.

Impulsive Noise Due to Blade Vortex Interaction

While impulsive noise due to compressibility and blade thickness effects can result in a discomforting noise in high-speed forward flight, the impulsive noise caused by blade vortex interaction during slow-speed descent into a terminal area can be a more troublesome noise source to the passengers, the surrounding community, and the people in the terminal area. Referring to figure 8, the flight region in which this noise source is of primary importance, for at least single rotor helicopters, is generally at forward velocities of 20 to 40 m/sec and for descent rates of 50 to 100 m/min. It is in this range of flight velocities and descent rates that the freely deforming rotor wake, containing the concentrated vortices trailed from the blade tips, can induce a strong and rapid pressure fluctuation on the rotating blades. The number of investigators that have undertaken research directed towards an understanding and prediction of the noise generated by blade vortex interactions are too numerous to reference and discuss the results of all their efforts. A cross section of the investigations that have been conducted can be summarized by a few notable references of previous effort. These references are Sternfeld's work in reference 42 on tandum rotors; Leverton and Taylor (ref. 43); Leverton (ref. 44); Schlegel, et al. (ref. 17); Widnall, et al. (ref. 45); White and Balcerak (ref. 46); and more recently, Charles (ref. 29), Tangler (refs. 32, 47, and 48), and Schmitz and Boxwell (ref. 30). On the basis of the efforts that have been undertaken by these and other investigators, the primary parameters associated with the intensity of the impulsive noise due to blade vortex interactions are the orientation of the previously trailed concentrated tip vortex with respect to the interacting blade, the strength of the concentrated tip vortex, and the distance of the concentrated tip vortex from the interacting blade. Sternfeld (ref. 42) showed that the relative location of the two rotor planes of a tandum helicopter was one of the primary parameters controlling whether impulsive noise due to blade vortex interactions was obtained in steady-state level flight. Figure 13 illustrates the type of vortex interaction that is generally obtained with tandum rotor configurations. The results presented in figure 13 were obtained by a rather unique smoke visualization system developed by the Boeing Vertol Company that could be used on a whirl tower or during flight tests of full-scale tandum configurations. The results obtained by Sternfeld indicate that the impulsive noise due to the interaction of a blade with a concentrated vortex was due to the blade of the rear rotor interacting with the trailed tip vortex from the forward rotor. The orientation of the concentrated vortex relative to the interacting blade is shown in the lower left-hand part of figure 13. Once the type and location of the blade vortex interaction was determined, the aft rotor was moved up so that the paths of the aft rotor plane and the concentrated freely deforming vortex from the blade of the forward rotor no longer crossed, thus eliminating the impulsive noise.

Tangler, in reference 32, through the use of schlieren techniques determined at least seven possible locations of blade vortex interaction points for a two-bladed rotor in descending forward flight in the range of 0 to 305 m/min. Figure 14 indicates where these points of interaction are with respect to blade azimuth and descent rate. The two most important of these intersections occur at $\psi = 55^{\circ}$ and 70° as they are heard both in the helicopter cabin and on the ground. On the basis of schlieren pictures, it was determined that the blade vortex interaction at $\psi = 55^{\circ}$ occurred with a vortex that was 1.5 revolutions old while that which occurred at $\psi = 70^{\circ}$ was with a vortex generated by the interacting blade during the previous revolution. Since both of these interacting vortices were below the blade chord plane, the velocity induced on the blade by the vortex generated a near sonic velocity on the blade resulting in strong bow shock waves. These shock waves radiated a strong pressure wave much in the same manner as the high-speed impulsive noise generated by compressibility effects as previously discussed. It is noted, however, in reference 49, that a very similar but less intense impulsive noise was also generated at approximately the same azimuth locations during tests of a two-bladed rotor in descending flight having a tip speed of only 152 m/sec. It is concluded, therefore, that while the trailed tip vortex is the basic cause of the impulsive noise due to blade vortex interaction in the first quadrant of the rotor azimuth, the induced effects of the vortex can also generate noise due to compressibility effects if the tip speed of the blade is sufficiently high.

The investigation conducted by Tangler in reference 32 on the character of the vortex interaction and the noise produced by those interactions on the retreating side of the azimuth in the vicinity of $\psi = 270^{\circ}$ (interactions 6 and 7 of fig. 14) indicated that these interactions induced local stalling on the blades. A similar type of impulsive noise generated on the retreating side of the rotor disk was also noted by Cox and Lynn (ref. 50) many years ago during a 1.5g left turn during a flyby of an HU-1A helicopter. Figure 15 presents a filtered trace of the noise measured during the time of blade slap. The authors of reference 50 reasoned that, on the basis of the characteristics that were measured, the interacting vortex induced stall on the blade which in turn generated high frequency vortex shedding. A similar conclusion was also reached by Schlegel, et al. in reference 17 on the basis of measurements made of the impulsive noise generated on the retreating side of the blade disk. The change in the noise spectrum that was obtained by Schlegel due to vortex shedding induced by stall is shown in figure 16. More noise was obtained in the higher octave bands when stalling, which was induced by an interacting vortex, occurred over a small portion of the azimuth.

On the basis of at least these studies, it might be concluded that the impulsive noise generated by the interaction of a blade and a concentrated vortex can be associated with blade stall and subsequent vortex shedding, impulsive loading of a subsonic blade section, or induced compressibility effects on a high tip speed rotor.

Basic to the prediction of any of these vortex-induced impulsive noises is the prediction of the freely deforming position and strength of the trailed vortices as a function of the azimuth position in which they were originally formed. There have been a few investigators who have conducted extensive investigations into the prediction of the strength and position of the vortices trailed from the tip of rotor blades in hovering and forward flight. Notable among these are Landgrebe (refs. 51 and 52) and the technical staff at the RASA Divison of Systems Research Laboratories, whose initial efforts in the development of the Non-Uniform WAke Induced VElocity (NUWAIVE) prediction technique are reported in references 24 and 53. Since these initial efforts, RASA has extended and refined the force free rotor wake analysis for use in the prediction of the wake geometries and rotor loads developed by advanced and unique rotor systems such as the Advancing Blade Concept (ABC) (ref. 54), the Variable Geometry Rotor system (VGR) (ref. 55), and the "X" Wing configuration (ref. 56) being developed by the Naval Ship Research and Development Center (NSRDC). As an example as to the use of the free wake analysis in conjunction with the time dependent rotor noise prediction technique developed by RASA (refs. 57 and 58) to analyze the noise developed by helicopter rotors, the results of calculations conducted to predict the impulsive noise generated by the HU-1A in a 1.5g climbing left turn (ref. 50) will be given. The results of this investigation were previously reported in reference 59 and figure 17 shows the location at which the blade intersects the predicted deformed wake. The spectrum of the noise heard by an observer located on the flight path and 305 m downstream of the aircraft is shown on the left side of figure 18. The noise due to the blade slap is rather weak because it occurs on the retreating side of the azimuth and the observer is a long distance from the aircraft. The humping characteristic of the noise spectrum is due to the predicted ground reflection effects at the observer's ear located approximately 1.5 m from the ground. When the effects due to vortex interaction are removed from the problem the spectrum on the right-hand side of figure 18 is obtained. By comparing the two spectrums it can be seen that, even though the blade vortex interaction occurs in a low region of dynamic pressure, a significant amount of impulsive noise is generated. Predicted pressure time histories of the noise during blade vortex interaction are presented in figure 19. The time history at the top of this figure includes both rotational and vortex shedding noise while that at the bottom of the figure presents only the rotational components of the noise. It can be seen, by comparing these two signatures, that during the interaction of the blade and vortex, the vortex shedding noise is significantly greater than it is at other times. This characteristic seems to be in agreement with that reported in references 17, 32, and 50. It is noted that the reason the time history of the blade passage noise is so broad is because of the low relative velocity between the blade and airstream on the retreating side of the azimuth.

The NUWAIVE deformed wake analysis and the rotor noise prediction, coined TRAMP by NASA, are in use by a number of firms, notably Kaman Aerospace and Hughes Helicopters to investigate the noise characteristics of various rotor systems under development. It is believed that these above noted analysis techniques are rather versatile and, if nothing else, provide at least an initial step towards the development of a program to predict the noise characteristics of rotor systems operating in a realistic environment.

Possibly because of the challenge or because of the inquisitive and imaginative character of the helicopter community, the helicopter aerodynamicist has for many years been investigating means of altering the characteristics of the trailed tip vortex to either eliminate or significantly reduce the impulsive noise associated with the interaction of a blade and a concentrated vortex. More recently, because of the vortex hazard created by jumbo jets the fixed-wing aerodynamicist has also been looking extensively into ways of altering the characteristics of concentrated trailed vortices. References 60 through 65 present some results of the investigations that have been conducted and figure 20 presents a pictorial summary of some of the various techniques that have been investigated. Tangler, in reference 48, also presented the results of an investigation using differential flaps, sub wing tips, and split tip configurations to alter the characteristics of the tip vortex generated by a rotor blade. While some of the configurations shown in figure 20 had some apparent beneficial effects on modifying the characteristics of the trailed tip vortex, it was found that the performance penalties negated the practical application of most of the techniques. Two techniques that have shown promise, however, are the Tip Air Mass Injection (TAMI) system illustrated in figure 20 and a passive system called the Ogee tip. Since both of these approaches, one active and one passive, have been rather thoroughly investigated and have shown a reasonable degree of success in reducing the impulsive noise due to blade vortex interaction, the approaches being utilized and the success that has been achieved with both systems will be reviewed briefly.

The TAMI system has been developed over a number of years by the RASA Division. References 49 and 66 through 73 present the results of some of these studies. The principle behind the TAMI approach is to inject a high pressure jet of air along the axis of the core of the vortex as it leaves the lifting surface. The mass flow and pressure injected into the core of the vortex causes an instant aging of the vortex (rapid radial redistribution of the vorticity) as well as causing a more rapid dissipation of the vorticity because of the higher level of turbulence induced by the jet stream within the vortex core (ref. 69). Reference 49 presents the results of tests of the system conducted in a wind tunnel using a model rotor system. Figure 21 shows a photograph of the two-bladed 2.13-m-diameter model rotor system mounted in the University of Maryland wind tunnel. The model, during this test series, had a tip speed of 152 m/sec and was operated over a large range of simulated descent rates at an advance ratio of 0.14 for a thrust coefficient $C_T = 0.00455$. This advance ratio was chosen as it was the one that full-scale flight tests indicated passes through the center of the most intense blade-slap noise as the descent velocity is increased (fig. 8). The model tests also confirmed that at an advance ratio of $\mu = 0.14$ the rotor descends through the center of the most intense noise. Figure 22 presents some of the results obtained during the model test. It is noted that the data presented in this figure have been scaled up to full-scale frequencies and descent rates. The results, presented on a dB(A) basis, show that in the continuous-loud banging area (V_D = 183 m/min) the overall dB(A) was reduced by 7.5 dB(A); while in the area of most intense noise, the overall dB(A) was reduced by only 4.5 dB(A). It is noted that the primary reduction in the dB(A) was at the higher frequencies and not at the frequency range that controls the overall dB(A) (150 to 300 Hz). This result is consistent with the change in the pressure time histories which showed that the interactive spikes were eliminated with the TAMI system operating but the level of the other rotational noise harmonics was not altered. These results indicate, therefore, that at least for impulsive noise which is not associated with compressibility effects, the acoustic energy of interest is concentrated more in the overall turbulence generated by the entire rotor wake which is close to the rotor plane, than it is with the impulsive noise generated by the discrete interaction of a blade with one or two concentrated vortices. It is believed that this observation has very meaningful implications as regards what can be done to relieve the impulsive noise due to blade vortex interactions. The basic question that must be answered is, During flight condition in which discrete blade vortex interactions occur, is the major acoustic energy associated with the discrete blade vortex interactions or with the induced turbulence generated by all of the concentrated vortex energy in the rotor wake? It is this "one man's opinion" that it is the latter. It is believed that an answer to this question can be obtained by present technology, if it is properly applied to the problem.

If the primary source of excess noise during descent is associated with the entire field of concentrated vortex energy distributed below the rotor, then there may be a limit to the reduction of blade-slap noise that can be obtained in this mode of operation. If so, then a review of the techniques being utilized to reduce blade impulsive noise during descent should be conducted in order to evaluate whether the approaches being utilized presently have the capability of reducing the impulsive noise to the degree that is desired.

As previously noted, another technique of reducing the impulsive noise due to blade vortex interaction that has been extensively investigated and which has been shown to have beneficial effects is a passive tip modification known as the Ogee tip. A strikingly similar tip shape was utilized on a helicopter rotor approximately 50 years ago. Figure 23 presents a 1930 photograph of the Curtiss-Bleeker helicopter which had low-aspect-ratio blades and a planform similar to the planform of just the Ogee tips. The reason the Curtiss-Bleeker helicopter utilized such a unique blade planform is not known, but I venture to suggest it was not to reduce the impulsive noise due to blade vortex interaction during descent or to reduce high-speed impulsive noise.

Research into the use of the Ogee tip on rotor blades to reduce the impulsive noise and dynamic loads due to blade vortex interactions in recent years is pretty well summarized by the investigations reported in references 74 through 76. The purpose of the Ogee tip is to distribute the aerodynamic loading in the tip region in a manner such that the vorticity shed at the blade tip is more like a vortex sheet than a concentrated line vortex which concentrates the vortex energy in a small compact volume. The effectiveness of the Ogee tip planform in redistributing the vortex energy is demonstrated by the data presented in figures 24 through 26. Figure 24 shows pressure isobars that were measured over a rectangular tip of an untapered rotor blade at an angle of attack of 12°. The data presented in this figure were obtained from reference 62. As can be seen from the data presented in figure 24, the formation of the concentrated vortex in the tip region generates a strong three-dimensional loading distribution having strong pressure gradients. In figure 25, pressure distributions that were measured over the Ogee tip show that approximately a two-dimensional pressure distribution is maintained over the entire tip region. This type of smooth pressure distribution (ref. 76) is also maintained over the Ogee tip when the blade is swept forward (fig. 26) as it would be in the second quadrant of the rotor azimuth. This is the rotor quadrant where the vortex is formed that intersects a following blade to generate the impulsive noise. Since the spanwise loading gradients are gradual in the region of the Ogee tip, the vorticity in the tip region would tend to be trailed as a weak unstable sheet and thus a large diameter diffuse trailed vortex would be formed.

Some preliminary results of flight tests of the Ogee concept were presented by Mantay in reference 77. Since he is presenting more detailed results of the flight test investigation in this symposium (ref. 78), I will just briefly discuss what I feel are the primary results that have been reported previously. Figure 27 presents the pertinent details of the aircraft that was used during the flight test. The aircraft was a UH-1H and the noise generated by the standard and Ogee tip blades having the same overall radius was compared over a range of flight velocities and descent rates. A brief composite summary of the data presented in reference 77, which I feel summarizes the results obtained as regards impulsive noise, is presented in figure 28. As can be seen from the results presented in this figure, which shows the acoustic signature for comparable locations within the respective impulsive noise boundaries, the pressure time histories are very similar although the peak pressures are somewhat lower for the Ogee tip than they are for the standard tip. It is suspected that, since the acoustic energy not associated with the interaction "spikes" has not been altered significantly, a dB(A) weighed spectrum might be similar to that obtained with the TAMI system (fig. 22).

It is believed that a major result of the tests discussed in reference 77 is the change in the location of the impulsive noise area as shown in figure 28. The significant increase in the descent rate at which the Ogee tip intersects the impulsive noise boundary significantly opens up the "noise-free" descent-approach corridor available to the pilot. Even if the noise within the impulsive noise boundary is not altered, the significant movement of the boundary within the flight envelope may be sufficient for commercial aircraft to make a quiet descent into a terminal area. It is believed that the large movement of the impulsive noise boundary might be related to the differences in the Mach number, loading distribution, and radial location of the formation of the tip vortex between the Ogee and standard blades. A series of wind-tunnel tests will be conducted to determine whether, in fact, this is the reason for the significant change in the impulsive noise boundary.

While the effort to date on ways to modify the impulsive noise due to blade vortex interaction has been largely experimentally oriented, it is believed that the state of the technology is such that theoretical investigations to evaluate the benefits that can be derived by various approaches can be undertaken to provide at least guidelines as to what might be expected by various vortex or blade modifications. For example, various new lifting line or lifting surface theories for helicopter rotor blades can predict the required detailed chordwise-spanwise loading distributions if the induced velocity distributions associated with the nonuniform wake and concentrated vortices are known. It is believed that the available freely deforming wake analysis, such as NUWAIVE (refs. 24 and 53), can provide these needed induced velocity distributions. Investigations are currently being conducted at RASA using the NUWAIVE program to determine the effects of various modifications to the vortex wake structure on the detailed loading distributions of highly elastic compliant rotor blades. The results obtained using the NUWAIVE program in conjunction with the Rotor Aeroelastic Response Analysis (RARA), which is an extension to the analysis procedure presented in reference 79, indicate that the analysis procedures predict the changes in loading one might expect due to changes in the vortex structure. It is believed that these or similar analysis procedures used in conjunction with a suitable rotor acoustic prediction program such as TRAMP (refs. 57 and 58) could be utilized to answer the question as to the division of acoustic energy between the discrete blade vortex interaction and that which is due to the concentrated vortex field in close proximity to the rotor during flight. It is also believed that these same or similar analyses could be utilized to establish the reason the impulsive noise boundaries for the Ogee tip are significantly different from those for the standard blade.

It is concluded, therefore, that analysis procedures which represent the state of the technology, or with slight extensions thereof, can and should be utilized to investigate various aspects of the impulsive noise due to blade vortex interactions. If this effort is undertaken, it is believed that a much more rapid advance in means of improving the noise characteristics of helicopters during descent could be accomplished.

Noise Due to Tail Rotors

Tail rotor noise, because of its higher blade passage frequency, can be a dominant source of noise in the frequency range of the so-called excess noise. On many helicopters in which the main rotor impulsive noise is not present, the noise source that draws attention to the helicopter as it is approaching is that developed by the tail rotor. It is a noise source, however, that has not received much attention in the past, particularly as regards noise reduction.

Pegg in reference 80 noted that the tail rotor developed acoustic signatures having significant high harmonic content along the flight path at frequencies of up to five times that at which the main rotor signature is lost in the background noise. A reason for the strong propagation of the tail rotor noise during these flight tests was not determined. However, acoustic data taken during flight tests of many helicopters show the same type of propagation characteristics of the tail rotor along the flight path although they may not be as severe as that reported in reference 80.

Hughes Helicopters, during the full-scale research program to develop a quiet helicopter (ref. 81), recognized the importance of the tail rotor to the overall noise signature of a helicopter and investigated the effects of various parameters on the noise characteristics of the tail rotor. It was found that by increasing the blade number from 2 to 4, reducing the rotational speed, and phasing the tail rotor blades in azimuth at 75° by 105°, a significant reduction in the noise developed by the tail rotor could be realized. This investigation was rather unique in that it was the only one, as far as is known, which had directed a significant effort towards reducing the noise output and propagation characteristics of tail rotors.

A reasonable question that might be asked at this point is, If tail rotor noise is a significant contributor to the excess noise that provides early detection and contributes to the annoyance characteristics of helicopters, why hasn't more effort been directed towards understanding and reducing this source of rotor noise? I do not think a unique answer to this question can be given, but it is suggested that the answer might lie somewhere between the following two answers:

(1) Until a solution is found to significantly reduce main rotor impulsive and rotational noise, the reduction of tail rotor noise is not going to significantly improve the noise characteristics of the helicopter.

(2) The noise associated with tail rotors is so much more difficult to analyze and understand than the noise due to main rotors because of the environment in which it is operating, it is not yet a trackable problem.

Since it is possible that answer (1) may be invalid in the near future, I would like to address my comments to the latter answer.

Is the aerodynamic environment in which the tail rotor operates complicated nd basically a real mess? Yes, without a doubt, it is one of the most, if not he most, difficult aerodynamic environment in which a lifting surface is required o perform. Figure 29 is a sketch of "simplified" representation of the velocity omponents to which a tail rotor is subjected. As is indicated in this figure, in ddition to all of the complicated aerodynamic effects to which a main rotor is subected, the tail rotor also is subjected to the periodic induced effects of the conentrated vortex wake from the main rotor. If the advance ratio is such that the oncentrated vortices from the forward and aft portions of the main rotor disk have aths such as shown in the top of figure 29, the relative velocity the tail rotor ould experience at the top of the disk would be somewhat as shown. As can be seen, he induced effect of the main rotor vortices is such that it causes a rapid variaion in the spanwise loading which can create a significant source of noise. Since he tail rotor does not operate at the same rotational speed as the main rotor, the mpulse frequency generated by these interactions would occur as the sum of the armonics of the blade passage frequencies of the main and tail rotors as on the igure. While theoretically, for every value of N, M can have values of 0 to ∞, he practical value of M generally never exceeds 3. The difference in the operatng speed of the two rotor systems, $\Omega_{TR}/\Omega_{MR} \approx 5$, allows the tail rotor blade to nteract a number of times with the same group of main rotor vortices during the ime interval it takes the vortices to cross the tail rotor disk. Because of the lifference in the orientation of the blades with respect to the vortices during ach intersection, the directivity pattern of the impulsive signature would be lifferent for each intersection.

At a lower advance ratio, when the main rotor vortices interact with the etreating blade of the tail rotor, the relative velocity the tail rotor blade night see at the bottom of the disk is shown at the bottom of figure 29. As can e seen, the velocity gradients and, thus, the loading gradients can be larger lue to the induced effects of the main rotor vortices than they were for the dvancing blade. Since the directivity pattern of the impulsive noise would be directed aft, it would not create the annoyance or detection problems that re caused by the blade vortex interactions with the advancing blade. Since he vortices are traveling in the same direction as the retreating blade of the ail rotor, the impulsive frequency due to the interactions of the main rotor ortices with the tail rotor blade would be at lower frequencies than they were or the advancing blade and would be defined by the relationship at the bottom f figure 29.

If the direction of the tail rotor was reversed, the advancing blade and etreating blades would have approximately the same perturbed velocity distriutions as indicated in figure 29 but would be reversed in their azimuth locaion in the tail rotor disk. With the direction of rotation now clockwise, the nain impulsive noise on the advancing blade due to the tail rotor/main rotor vake interaction would occur at low advance ratios when the rotor wake passed over the lower part of the tail rotor disk instead of at high advance ratios vith the tail rotor operating in the counterclockwise direction. At high advance ratios, however, when the strength of the main rotor vortices is higher and the main rotor wake is intersecting the top portion of the tail rotor disk, the tail rotor with the reversed direction would direct the impulsive noise aft and the tail rotor noise to an observer of an approaching helicopter would appear less than that of a helicopter having a tail rotor operating in the counterclockwise direction. On a subjective basis, therefore, a clockwise rotation of the tail rotor would seem to be advantageous.

In actuality, the problems associated with understanding and predicting the noise developed by tail rotors are much more complicated than just indicated. When the effects of the turbulence generated by the complete main rotor wake, the interaction of the main rotor and tail rotor wake flows, the interaction of the tail rotor wake (tractor configuration) or inflow (pusher configuration) with a lifting vertical fin, and the non-integer rotational speed ratio between the main rotor and tail rotor are considered, a more complete understanding of the complicated flow field at the tail rotor is obtained. References 82 and 83 present an excellent discussion on the effects of these various parameters on the aerodynamic characteristics of tail rotors and are recommended reading for anyone interested in understanding or predicting the noise characteristics of tail rotors.

Recently Leverton, et al., references 84 through 87, reported the results of a noise investigation conducted on the tail rotor of the Lynx helicopter at Westland Helicopters Ltd. This very interesting and intriguing study investigated the modulated noise developed by a four bladed tail rotor intersecting the main rotor concentrated vortex flow. This modulated noise, labeled a "Burbling Sound" by Leverton, was determined to be caused by a main rotor vortex being intersected four or five times by the tail rotor blades as it passed through the tail rotor disk, thus giving rise to groups of impulses as each tip vortex of the main rotor passed through the tail rotor disk. Due to this grouping effect and the variation in the amplitude of the pressure pulse, the chain of impulses is effectively modulated and the interaction noise is heard as a deep throated burbling sound. Figure 30 presents a spectrum of the noise measured by Leverton as the helicopter approached (ref. 84). The tail rotor peaks as well as the blade passage peaks developed by the main rotor vortices are clearly evident in this spectrum. Figure 31 shows the effect Leverton measured for a 130-knot flyby of the Lynx helicopter when the tail rotor direction of rotation was changed from counterclockwise to clockwise. Since, at this speed, the main rotor wake is interacting with the top portion of the tail rotor disk (fig. 29), the significant difference in the noise level during approach is explainable as the tail rotor blades are in the retreating side of the disk when they intersect the main rotor vortices. The work Leverton presented in reference 84, and the associated references, is well worth studying as it presents a great deal of information regarding an understanding of the tail rotor noise produced by the interaction of the blades of the tail rotor with the concentrated vortices of the main rotor wake.

Recently an exploratory investigation of the effects of a number of configuration parameters on the noise produced by tail rotors operating in a realistic aircraft environment was conducted by the RASA Division for the NASA Langley Research Center (refs. 89 and 90). Figure 32 shows a picture of the model that was constructed specifically for these investigations. The model was approximately a 1/16-scale version of a UH-1 series helicopter. The main rotor blades had a diameter of 91.4 cm, had a chord of 4.45 cm, had a twist of -8° from the blade root to blade tip, and were hinged at the 4.2 percent blade radius. The tail rotor blades were 19.1 cm in diameter and had a chord of 1.14 cm. The tail rotor blades had an NACA 0015 airfoil section, were untwisted, and were mounted as cantilever beams to the tail rotor hub.

The helicopter model was designed to duplicate the thrust coefficient, solidity, and advance ratio of a full-scale UH-1 series helicopter. With this scaling the main and tail rotor wake flows for the model and the full-scale helicopter would retain the same location in space relative to each other and to the rotor blades. However, because of the requirement for an advance ratio of 0.30 at a tunnel velocity of 30 m/sec, the scaled rotor was designed to operate at a lower tip speed than the full-scale rotor. Because of this scaling the effects of compressibility could not be tested with the model in its present form. The manner in which the model was designed and constructed provided the capability for variations in many of the main rotor wake/tail rotor parameters of primary importance. The following table lists these parameters and the range over which they could be varied:

Main rotor collective pitch angle	0° to 20°
Tail rotor collective pitch angle	0° to 8°
Tail rotor/fin offset spacing	0.20R to 0.31R
Shaft tilt angle	0° to 15° nose down
Tail rotor/main rotor disk longitudinal spacing	$0.50c_{MR}$ to $4.05c_{MR}$
Tail rotor/main rotor hub vertical spacing	-1.85R to 1.62R
Main rotor rotational speed	0 to 4100 rpm
Tail rotor rotational speed	0 to 13,000 rpm
Tail rotor direction of rotation	Clockwise or counterclockwise
Tail rotor thrust mode	Tractor or pusher
Fin blockage area	12% to 25% tail rotor disk area

Figure 33 presents a typical spectrum of the noise measured for the helicopter model at an advance ratio of 0.09. The similarity between the characteristics of the model and full-scale spectrum presented in figure 30 are apparent and although not marked in figure 30, the peaks at $2N\Omega_{TR} \pm 2M\Omega_{MR}$ due to the main rotor wake are very pronounced.

The following table presents the general effect of various parameters on the noise produced by the tail rotors that were noted during the brief exploratory investigation that was conducted:

Parameter	Effect		
	Large	Moderate	Slight
Advance ratio	X		
Longitudinal spacing			X
Lateral spacing	X		
Fin blockage area		X	
Operating mode (pusher, tractor)	X		
Direction of rotation*	X		
Tip speed		X	
Tail rotor/main rotor speed ratio			X
Main rotor thrust			X

*Since the investigation did not extensively investigate the effects of directivity, the effect of the direction of rotation may be somewhat overstated.

Pegg, et al. (ref. 88) have reported during this symposium, the results of an additional research investigation that NASA has recently conducted with the same model. These results tend to confirm and expand upon the results obtained during the brief exploratory investigation reported in references 89 and 90.

Since a rather systematic set of acoustic data on the effects of various parameters on the noise produced by a tail rotor operating in a realistic environment had been obtained (refs. 89 and 90), an effort was undertaken by the RASA Division to determine whether, by utilizing existing theoretical programs, the tail rotor noise characteristics measured for the model could be predicted. The analyses that were used in this investigation were the advance version of NUWAIVE free wake analysis developed from the analysis presented in reference 53 and the rotor noise prediction program discussed in reference 58. The relative paths of the main rotor concentrated vortices as they pass through the tail rotor disk, as predicted by the NUWAIVE program at an advance ratio of 0.20, are presented in figure 34.

As can be seen from figure 34 the concentrated vortex generated in the forward portion of the main rotor disk passes through the tail rotor plane clos to the path of the concentrated vortex trailed from the aft portion of the roto disk. While the proximity of the two vortex paths might be surprising, it is as would be expected when the strong upwash induced on the forward vortex filaments by the rotor wake as it passes across the rotor disk is recognized. To obtain a realistic understanding of the real effects of the main rotor wake on the noise produced by a tail rotor, it is believed that knowledge of the force-free wake positions is of paramount importance, particularly in the transition flight regime where the wake induced effects can be dominant. The solid outline is the tail rotor position for which a comparison of experimental and theo retical results will be presented. The dashed outline of the tail rotor disk are positions for which experimental data are also presented in reference 89.

In the following discussion, various degrees of sophistication in the prediction of the aerodynamic flow field will be used in order to demonstrate the characteristics that need to be modeled in order to predict the noise output of a tail rotor operating in a realistic aerodynamic flow field. The first results that will be presented are based on the following assumptions:

(1) The downwash of the tail rotor is represented by a uniform flow field.

(2) The main rotor wake is represented by a uniform downwash containing only the concentrated vortices from the blade tips.

(3) The tail rotor rotational speed is an integral harmonic of the main rotor rotational speed $\Omega_{TR}/\Omega_{MR} = 5$.

The predicted pressure time history of the tail rotor noise at a microphon upstream of the model is presented in figure 35. The intersections of the two blades with the various vortices are indicated and the obvious periodicity of the pressure time history is apparent. As time continues, because of the assumed integral relationship between the rotational speeds of the main and tail rotors, the pressure time history shown in this figure would be repeated.

The spectrum of the periodic pressure time history is presented in figure 36. As can be seen, the correlation with the experimental data is very poor. Except for the pressure at the first blade passage frequency, the predicted dB level is almost the same throughout the frequency range. Using the actual value $\Omega_{TR}/\Omega_{MR} = 5.09$ instead of the previous assumption of an integral relationship between the main rotor and the tail rotor rotational speeds, $\Omega_{TR}/\Omega_{MR} = 5$, gives the pressure time history shown in figure 37. A significant difference in the wave form of the primary pressure peaks can be seen when this pressure time history is compared with that presented in figure 35. If time was allowed to continue, each of the following wave forms would be different from its predecessor in the same time interval. The spectrum of this pressure time history (fig. 37) is presented in figure 38. Comparison of this spectrum with that presented in figure 36 shows that, while the pressure peak of the first blade passage frequency is predicted fairly well and a small pressure peak associated with the second blade passage frequency is now apparent, the correlation of the predicted and measured spectrum has not been improved significantly over that which is predicted on the basis of an integral relationship between the rotational speeds of the two rotor systems. The removal of the integral relationship between the rotational speeds does remove the repetitive characteristic of the spectrum in the frequency range of 0.5 to 4.0 Hz. When the complete nonuniform unsteady downwash characteristics of the main rotor wake are considered in addition to the concentrated tip vortices, the spectrum presented in figure 39 is obtained. As can be seen, the correlation between predicted and measured results is rather good with the remaining differences probably due to the nonuniform wake effects of the tail rotor.

On the basis of the results presented in the last series of figures, it is concluded that, while some characteristics of the tail rotor noise may be evaluated considering only the interaction of the concentrated main rotor blade tip vortices with the tail rotor blades, the tail rotor noise is dominated by the total unsteady induced velocity characteristics of the main rotor wake. In addition, it is apparent that the significant aerodynamic parameters which need to be included in an analysis which can be used to predict the noise characteristics of tail rotors must be at the least the following:

(1) Definition of the deformed spatial positions of main rotor wake over the tail rotor disk

(2) The induced velocity of the wake of the main rotor on the tail rotor inflow and downwash

(3) Representation of the nonperiodicity and arbitrary phasing of the interaction phenomenon

(4) Nonuniform wake effects of the tail rotor

(5) The aerodynamic interference effects of the tail fin on the tail rotor inflow and downwash

It is believed that by extending and modifying deformed wake analysis procedures, such as that represented by NUWAIVE (ref. 53), the wake-induced aerodynamic flow field over the tail rotor disk can be predicted to the required degree of accuracy for inclusion in the rotor noise prediction theory of reference 58 so that the effect of various aerodynamic and geometric parameters on the noise output of tail rotors can be realistically investigated. It is believed that, through the proper application of the above noted predictive techniques, a meaningful investigation of ways of reducing tail rotor noise can be undertaken.

As indicated, while the unsteady loadings developed by a tail rotor operating in a realistic environment are rather complex and result in a rather complex noise signature, it is believed that existing technology, or a relatively minor extension of existing technology, if properly utilized, can form the basis of analyzing means of relieving the tail rotor noise problem.

Another unique rotor system that has demonstrated superior performance dynamic loads and acoustic characteristics is called the VGR. The concept upon which the rotor design is based is to determine the most favorable relationship between the location of the rotor system with respect to the wake it develops. The rotor system parameters that have been utilized to investigate the most beneficial relationship are the relative vertical spacing and the azimuthal spacing of the rotor blades with respect to each other. Some results of a full-scale test of a VGR system on a whirl tower are presented in figure 40. These results were obtained for a six-bladed rotor in which the blades were coplanar and spaced 60° in azimuth and also for a corotating rotor system in which alternate blades were moved 1 chord below the other three blades. In this configuration there are basically two three-bladed rotor systems in which the blade phasing between the blades is 120° while maintaining 60° between the blades of the total rotor system. The significant reduction in the noise output of the rotor system when the separated rotor planes was incorporated is obvious from the data presented. The effects of both rotor separation and blade azimuth phasing on the performance, dynamic loads, and acoustics of the VGR concept were investigated theoretically using the deformed wake, loads analysis, and noise prediction programs (refs. 53 and 58). The investigation (ref. 55) indicated results similar to those obtained in hover as it was shown that, with the proper blade spacing and phasing, the performance and blade dynamic characteristics could be enhanced and the acoustic signature altered significantly.

It is believed that these investigations have shown that although, at first glance, the problems associated with predicting the noise characteristics of rotor systems operating in a complex flow field seem beyond the scope of reality, the proper and knowledgeable application of existing state of the technology techniques can be used successfully to investigate the effects of various rotor and aerodynamic parameters on the noise characteristics of rotors operating in such an environment.

CONCLUDING REMARKS

In reviewing the efforts of many investigators over the last 10 to 15 years, it became obvious that a significant advance has been made in the area of Rotor Noise Technology. This is particularly true as regards the understanding and predictability of the basic aerodynamic mechanisms associated with the generation of rotor noise, i.e., noise due to steady loadings, compressibility, and thickness. However, it also became apparent that the real helicopter environmental effects, such as free-stream turbulence, the induced effects of the rotor wakes, lifting surfaces, and fuselages have not been adequately considered in the development of these techniques. It is these self-induced and configuration effects which can be the primary reason the basic aerodynamic mechanisms result in the undesirable amplification of excess noise. It is important therefore to develop prediction programs to include these environmental effects so that the helicopter rotor noise that must be reduced to acceptable levels can be understood and investigated. The following table sets forth my evaluation of the status of Rotor Noise Technology as regards both the basic mechanisms and the application of the basic mechanisms to the real helicopter environment. It is readily apparent, after studying my evaluation, that

EXCESS NOISE SOURCE	IMPACT ON NOISE			UNDERSTANDING OF BASICS						UNDERSTANDING IN REAL ENVIRONMENT						HOPE FOR NEEDED CORRECTION					
				EXPERIMENTAL			THEORETICAL			EXPERIMENTAL			THEORETICAL			EXPERIMENTAL			THEORETICAL		
	L	M	H	P	F	G	P	F	G	P	F	G	P	F	G	P	F	G	P	F	G
MAIN ROTORS		X—	X			X		X—	X		X			X			X—	X		X—	X
Structural Vibrations	X—	X				X			X	X—	X			X—	X		X			X—	X
Stall	X—	X			X		X—	X		X—	X		X			X—	X		X—	X	
Free Stream Turbulence	X—	X			X			X			X			X		X—	X			X	
Compressibility Effects M<0.85		X—	X			X			X		X		X—	X			X—	X		X—	X
Compressibility Effects M>0.85			X			X		X—	X		X—	X		X—	X		X—	X		X—	X
Blade Vortex Interactions		X—	X		X—	X		X			X		X—	X			X—	X		X—	X
Wake Turbulence		X			X			X			X		X—	X			X			X	
Body-Lifting Surface Interference	X—	X		X—	X		X—	X		X—	X		X—	X			X—	X		X—	X
TAIL ROTORS		X—	X		X		X—	X		X—	X		X—	X			X—	X		X—	X
Structural Vibrations		X		X—	X			X—	X	X				X			X—	X		X—	X
Stall	X—	X			X		X—	X		X—	X		X			X—	X		X—	X	
Free Stream Turbulence	X				X			X			X			X		X—	X		X		
Compressibility Effects M<0.85			X			X		X—	X	X—	X		X—	X			X—	X		X—	X
Compressibility Effects M>0.85			X			X			X	X—	X		X—	X			X—	X		X—	X
Blade Vortex Interactions		X—	X		X—	X		X		X—	X		X—	X			X—	X		X—	X
Wake Turbulence																					
Main		X—	X		X			X			X			X			X			X—	X
Tail		X		X—	X			X		X—	X		X—	X			X			X	
Body-Lifting Surface Interference		X			X			X		X—	X		X—	X			X			X—	X

I do not believe our understanding and prediction of the helicopter self-induced environmental effects on rotor noise are on as solid a foundation as that which has been developed for the basic noise mechanisms. This is particularly true as regards the environment in which the tail rotor operates. As indicated by my relatively high rating of the possibility of reducing the excess noise to acceptable levels, I believe there should be a reasonable degree of confidence in our ability to analyze and predict the self-induced environmental effects. To either verify or negate these confidence ratings, I would recommend that state-of-the-art predictive techniques (or a minor extension thereof) of the helicopter self-induced environmental characteristics be utilized in conjunction with the basic noise predictive analysis techniques to determine the true State of Rotor Noise Technology. It is believed that such a study is warranted particularly for tail rotors. The results of such an investigation should bring to light possible deficiencies in either the understanding of or in the parameters required for the prediction of the basic types of noise mechanisms. In addition, the results of such an investigation would also establish limitations in the analyses used for the prediction of the true helicopter rotor environment (i.e., rotor wake analyses, rotor body interference analyses). Since it is the rotor noise produced in the real-world environment that must be reduced and not that produced within the "laboratory" type environment, the initial steps in this direction should be undertaken soon.

One final thought that came to mind during the preparation of this review paper is associated with the validation of theoretical predictive techniques. It is rather universally assumed that, if the results of a predictive analysis do not match a measured result, the theory must be incorrect. I believe that this assumption can be totally inaccurate and it may be responsible for unnecessarily limiting our confidence in the capabilities of various predictive techniques. A simplified analysis can demonstrate that, if the basic parameters which define the equipment and test conditions are not adequately defined, the measured results may be extremely misleading and useless as a correlative data base. It is recommended that state-of-the-art sensitivity analyses be undertaken to evaluate the degree to which the basic system and test parameters must be defined in order that the measured noise data have a confidence level to within ±20 percent. To some, the results of such analyses might be surprising as it was to me when I undertook such a study to evaluate parameter input requirements for establishing a reliable correlative data base for rotor dynamic loads. It is believed important to undertake such an evaluation in order that present "data banks" can be analyzed as to their applicability in assessing the adequacy of predictive techniques. It may be found that there are no adequate data and that a directed effort will be required to develop a suitable data bank.

It is suggested that we can be successful in reducing the excess noise to acceptable levels if we are bold enough to open the laboratory door and accept the challenge to further the development of the capabilities to understand, analyze, and predict the noise developed by rotors in their true operating environment.

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Figure 1.- CH-53E helicopter.

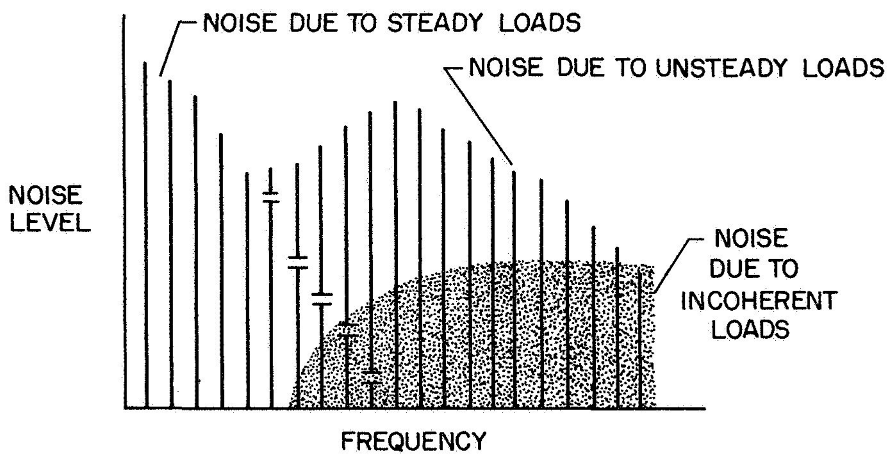


Figure 2.- Generalized acoustic spectrum for rotors.

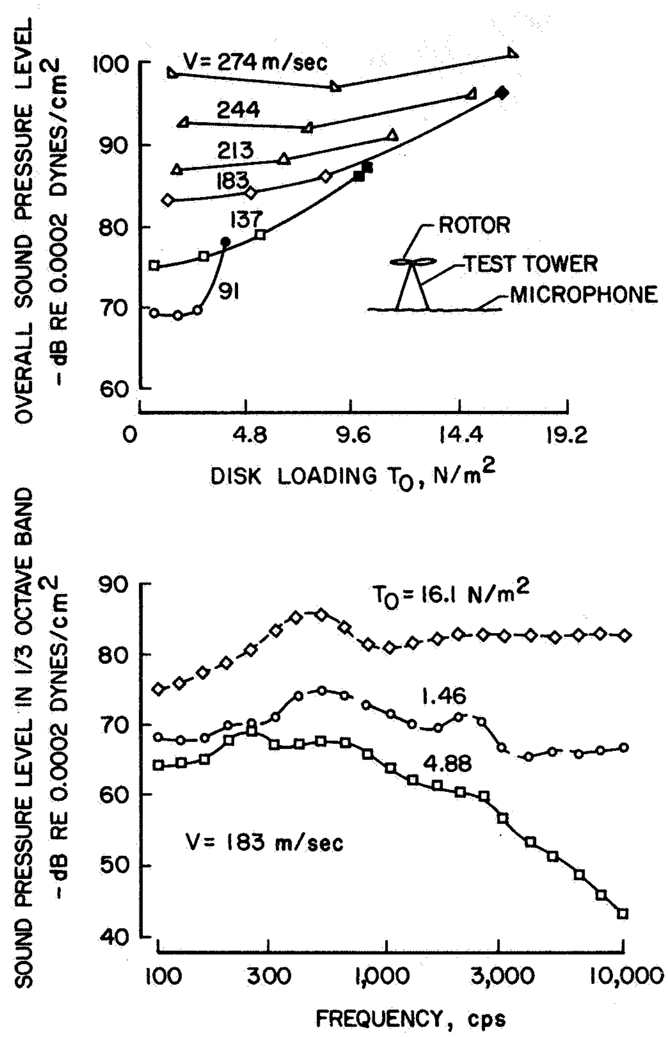


Figure 3.- Effect of stall on rotor noise.

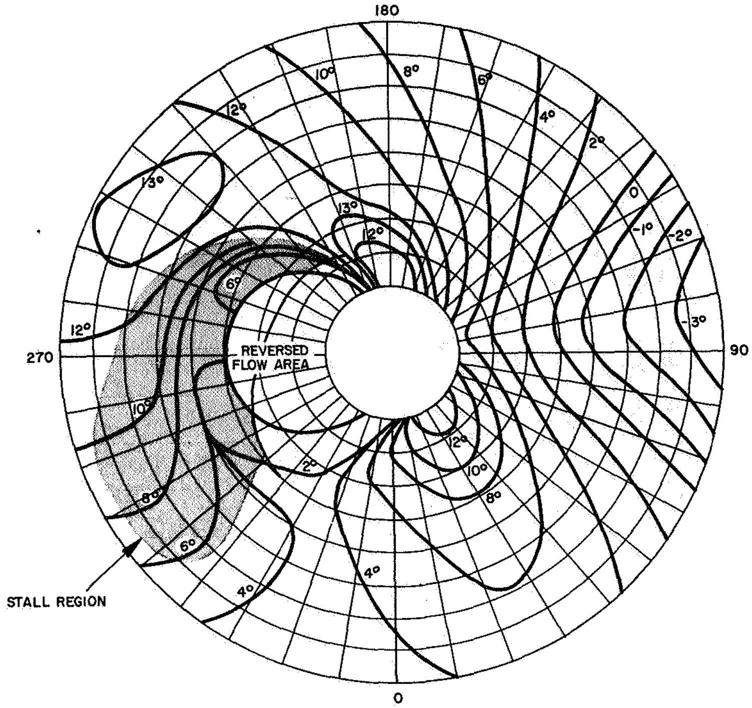


Figure 4.- Angle of attack contours.
UH-2, $\mu \cong 0.48$.

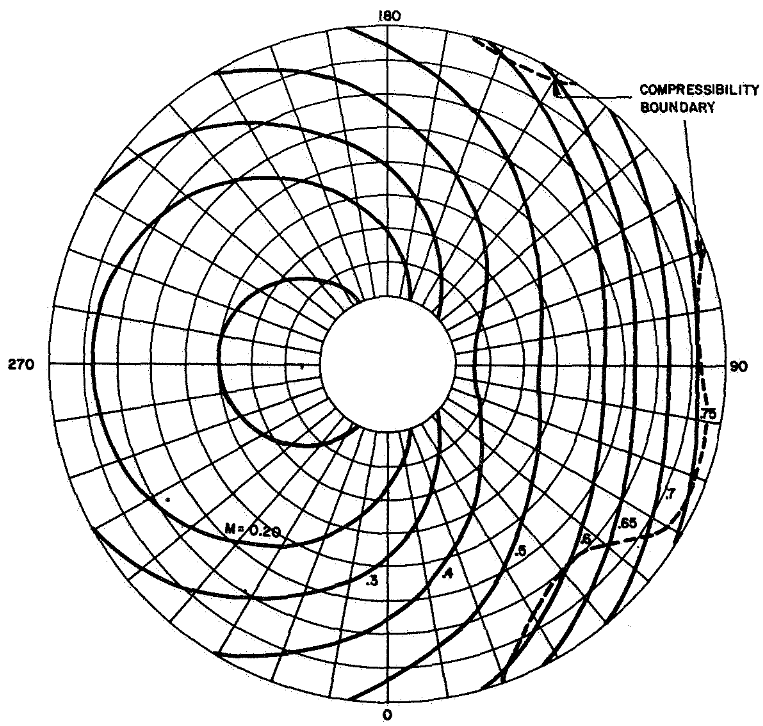


Figure 5.- Mach number contours.
UH-2, $\mu \cong 0.48$.

Figure 6.- RPV propeller in the test facility.

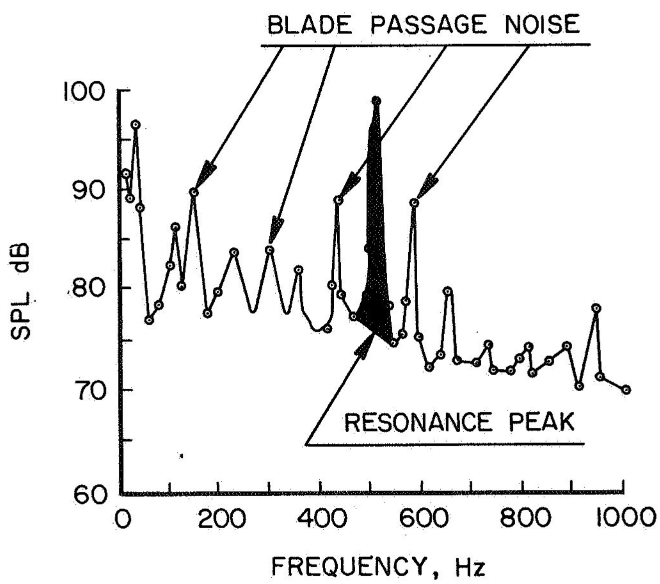


Figure 7.- Noise due to propeller resonance.

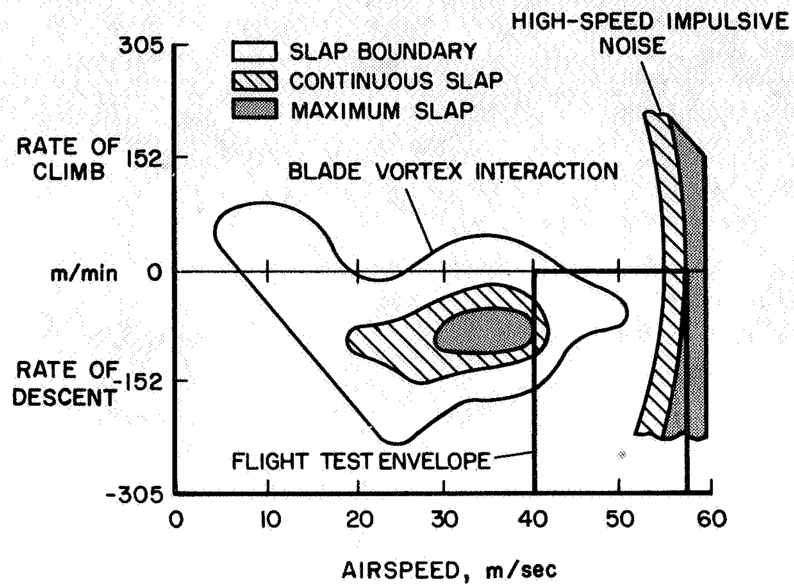


Figure 8.- Impulsive noise boundaries for UH-1 series helicopters.

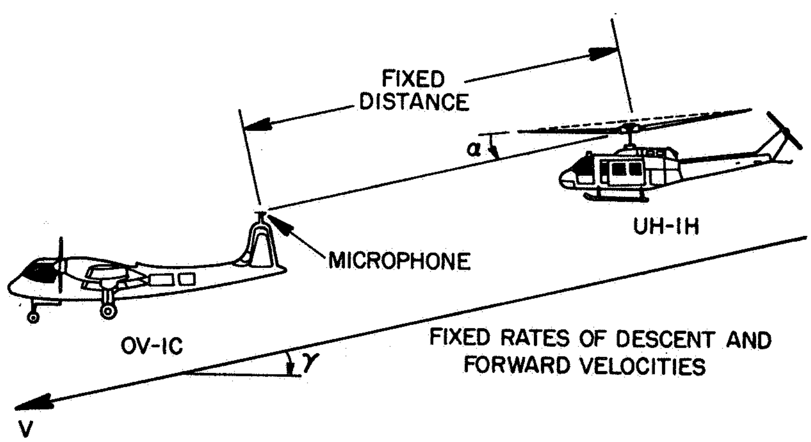


Figure 9.- Schematic of in-flight far field measurement technique.

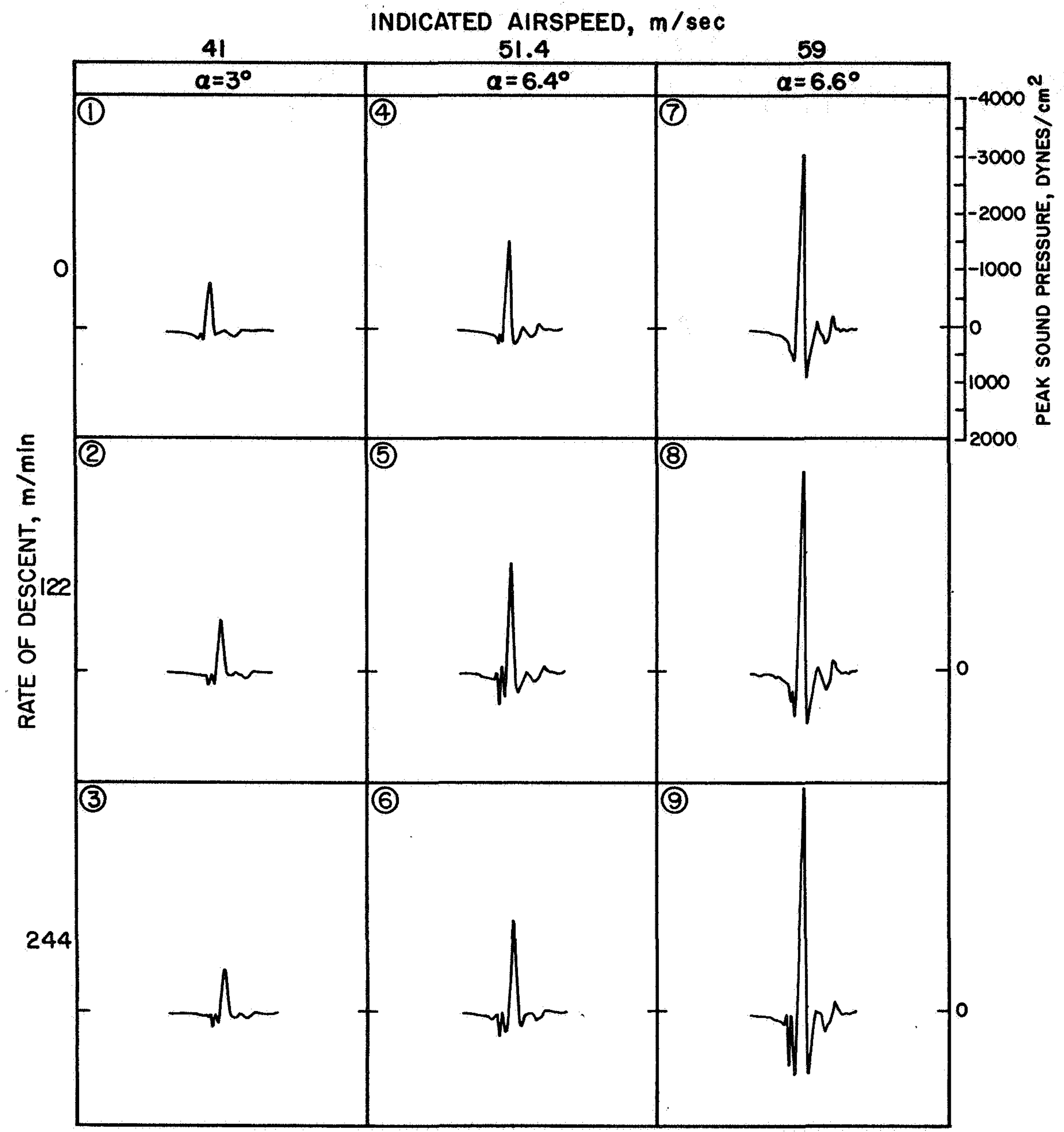
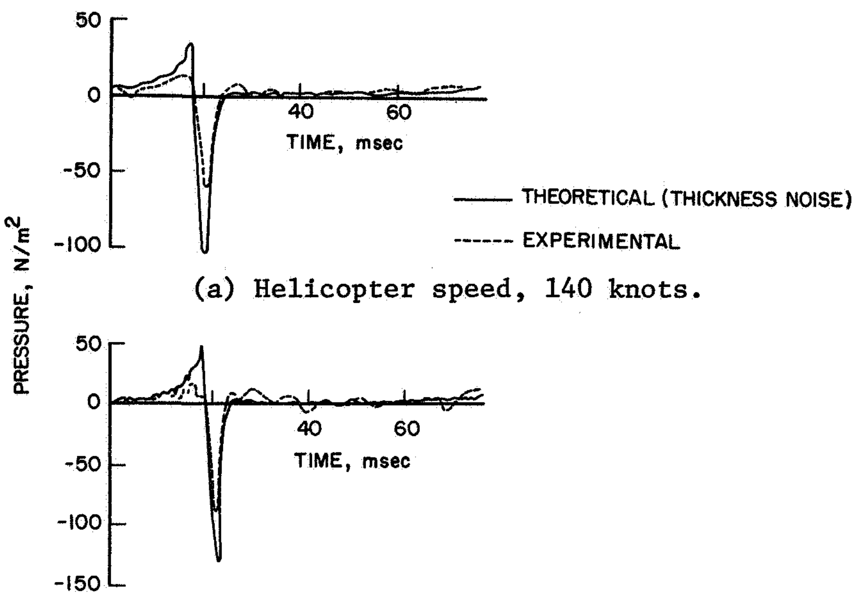


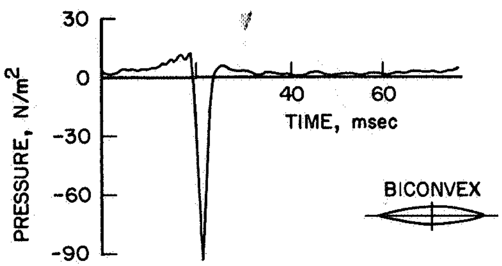
Figure 10.- Averaged acoustic signature of UH-1H impulsive noise versus forward airspeed and rate of descent.



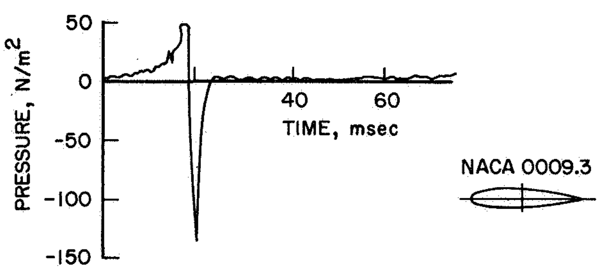
(a) Helicopter speed, 140 knots.

(b) Helicopter speed, 170 knots.

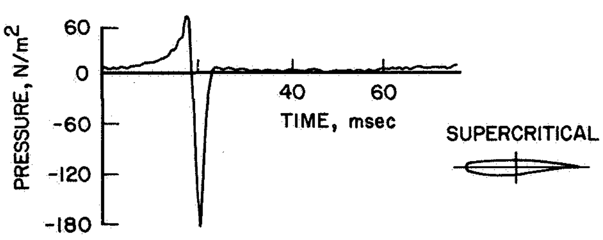
Figure 11.- Comparison of experimental and predicted thickness noise.



(a) Biconvex airfoil (parabolic arc).



(b) NACA four-digit airfoil.



(c) Supercritical airfoil.

Figure 12.- Theoretical effect of change in thickness distribution on acoustic pressure signature. Thickness noise only. (Note change of scale.)

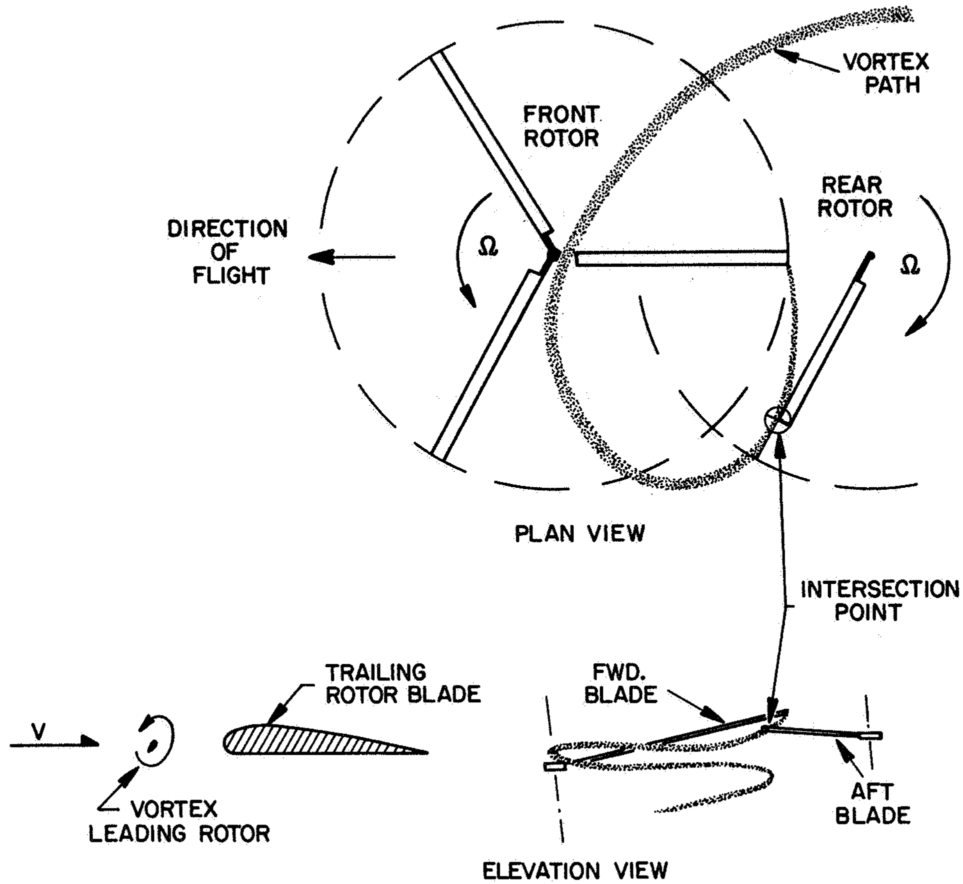


Figure 13.- Diagram of tandem rotor wake geometry in forward flight during impulsive noise.

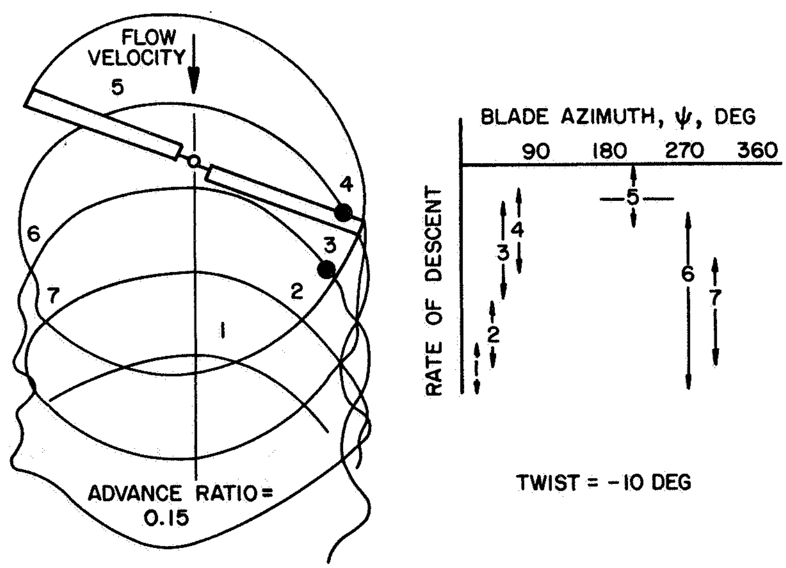


Figure 14.- Blade/vortex intersections during partial power descent.

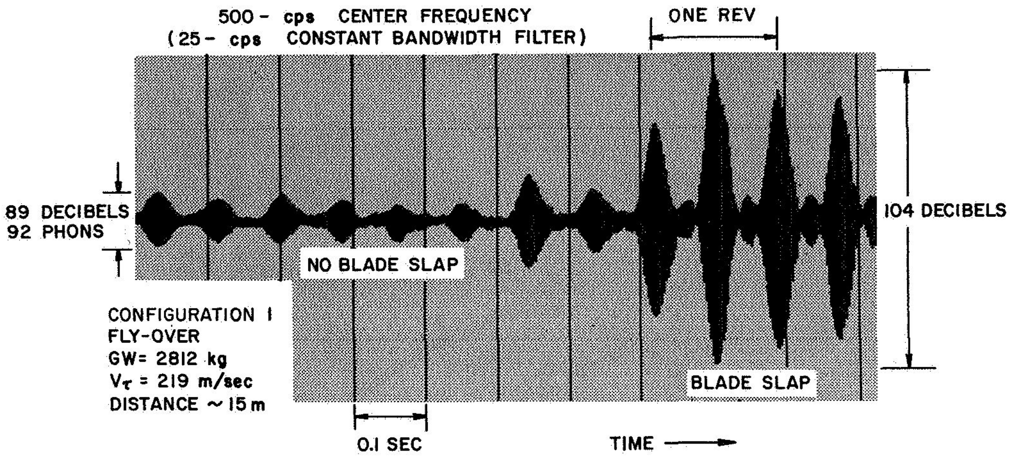


Figure 15.- Single rotor helicopter blade slap during turn at a frequency of 500 cps.

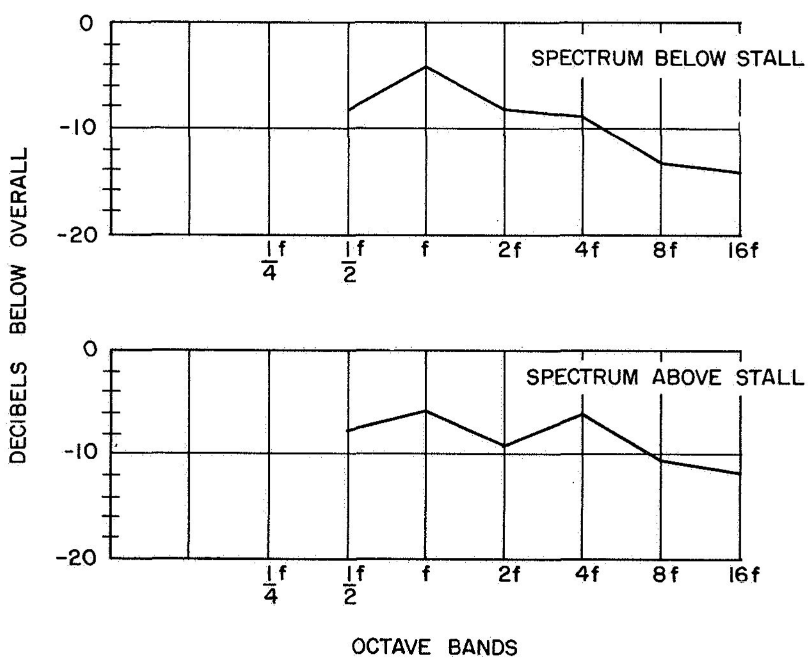


Figure 16.- Changes in spectrum due to retreating blade slap.

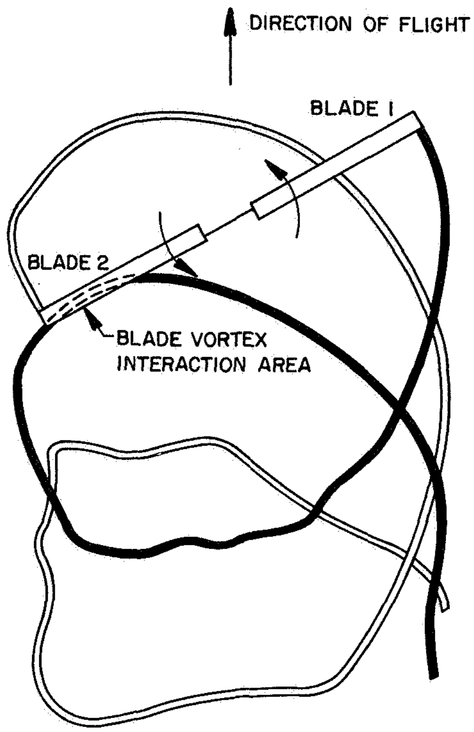


Figure 17.- Predicted deformed wake position of UH-1 helicopter in a 1.5g climbing left turn.

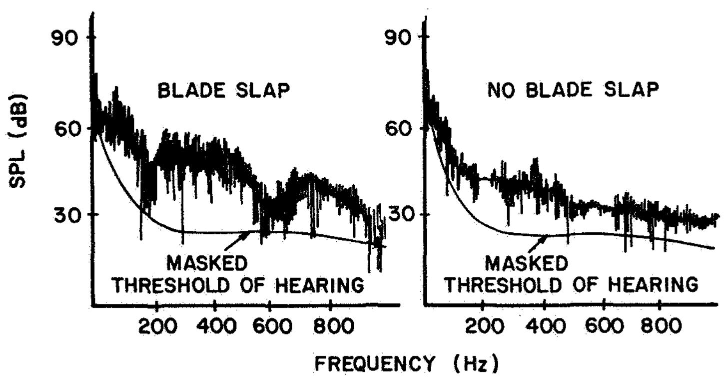


Figure 18.- Spectra of UH-1 in a 1.5g climbing left turn.

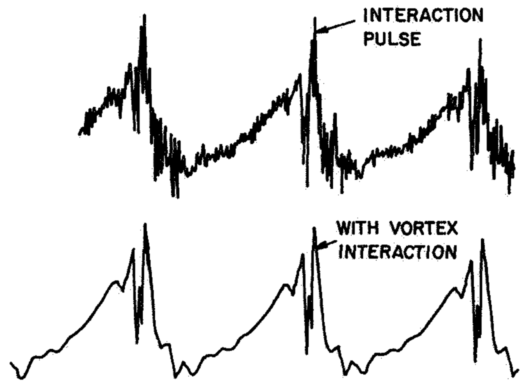


Figure 19.- Pressure time histories of a UH-1 in a 1.5g climbing left turn.

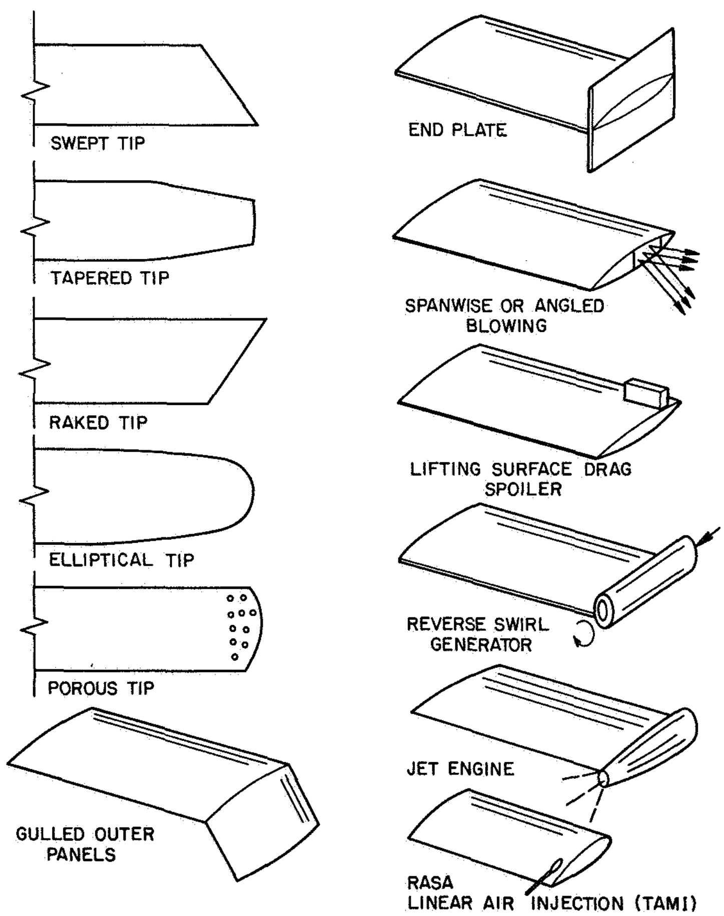


Figure 20.- Configurations for distributing, dissipating, or relocating tip vortices.

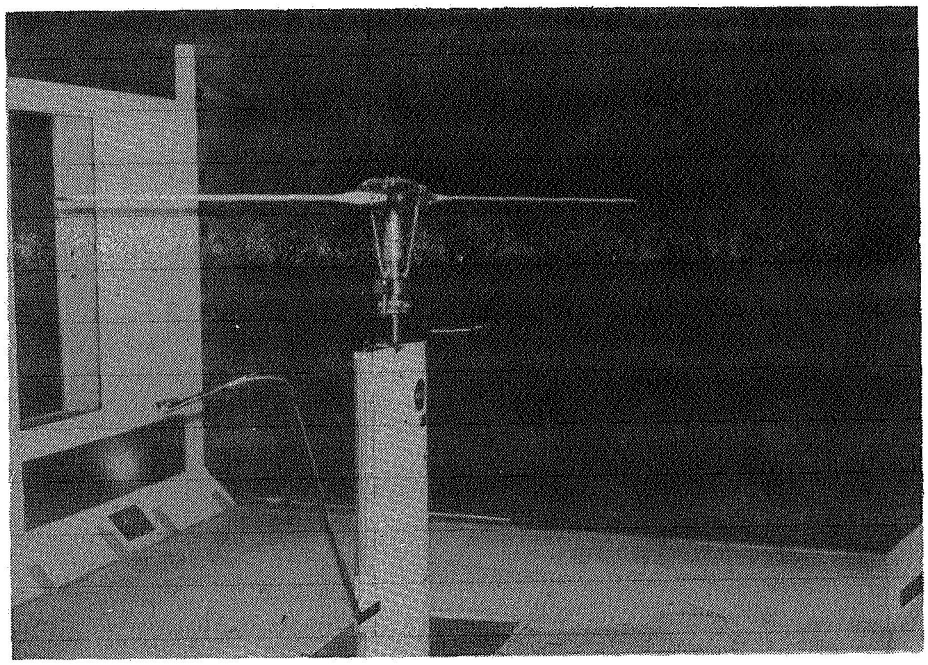


Figure 21.- TAMI model in wind tunnel.

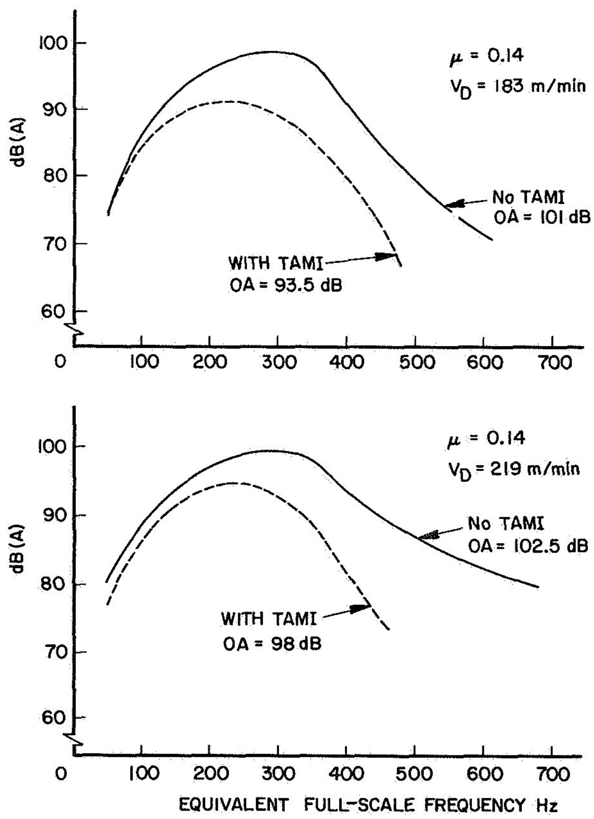


Figure 22.- Effect of TAMI on dB(A) weighted spectrum.

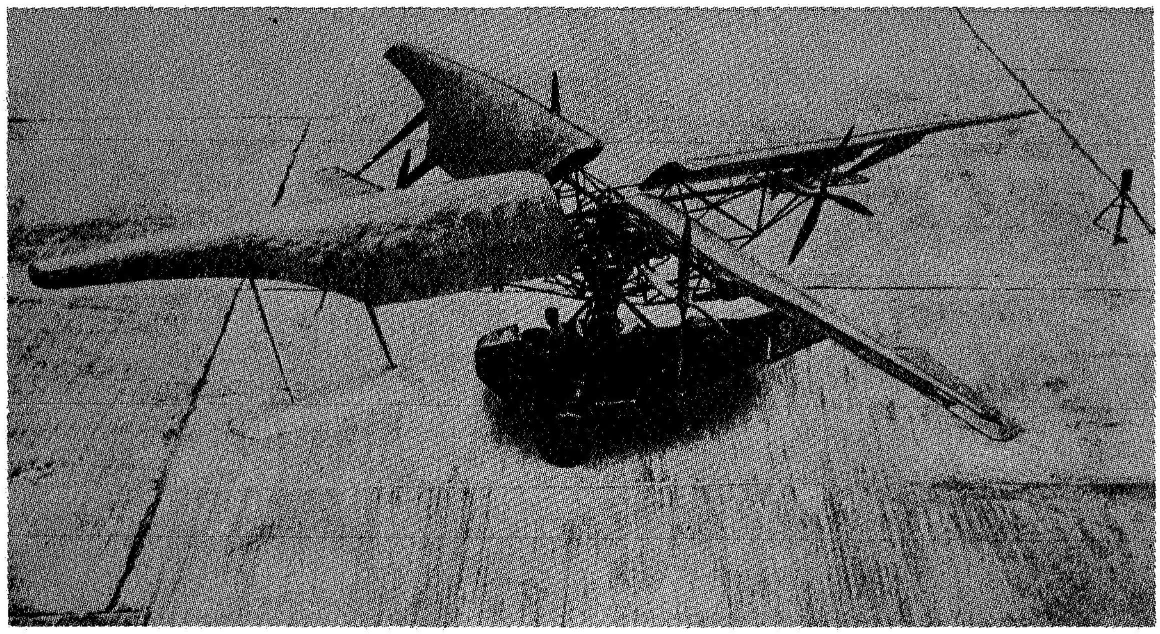


Figure 23.- Curtiss-Bleeker helicopter - 1930.

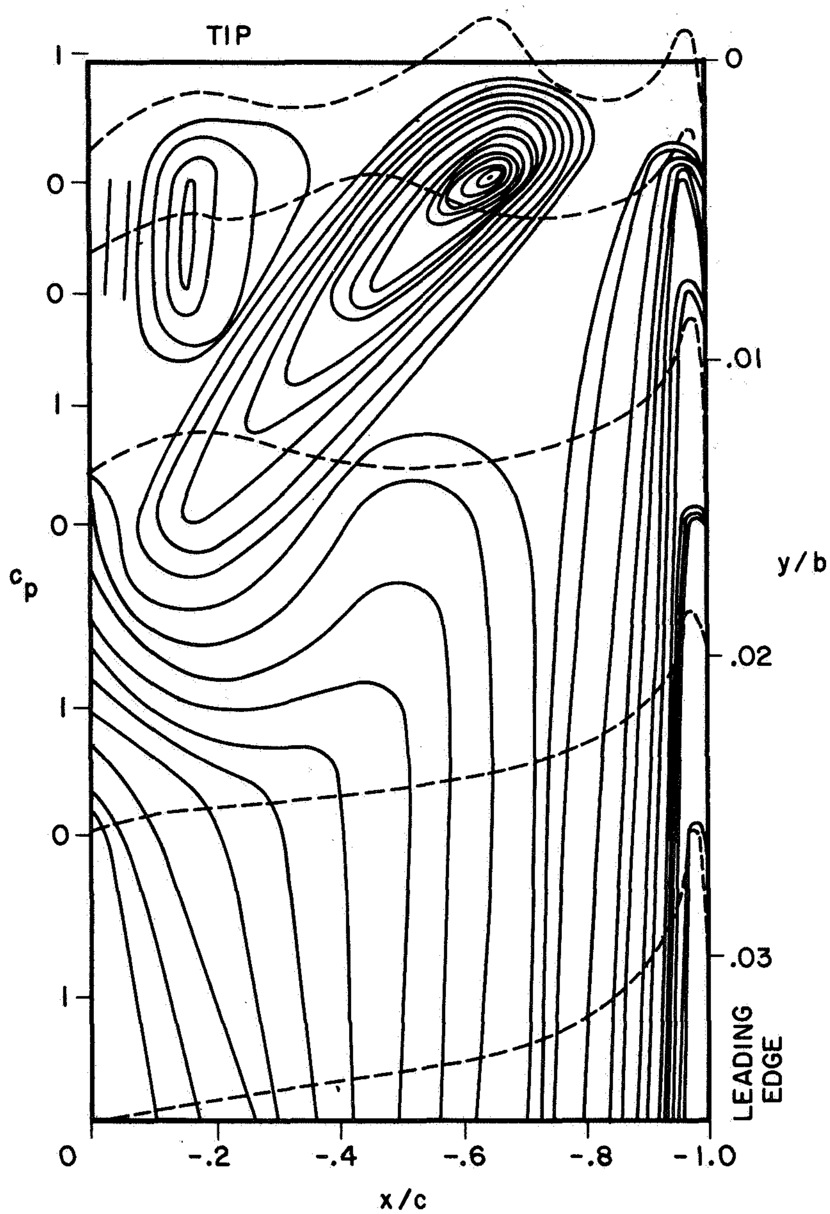


Figure 24.- Pressure isobars on top surface of blade tip at $\alpha = 12^{\circ}$ and $\lambda = 0$.

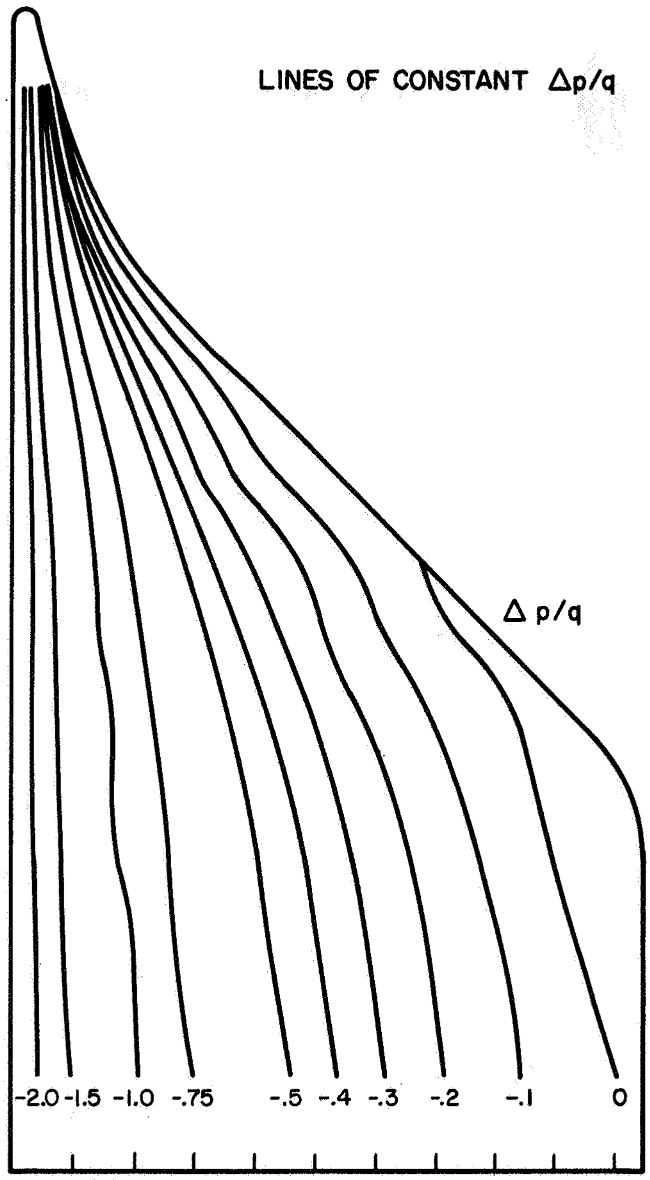


Figure 25.- Contour pressure plot of the Ogee-tip section at $\alpha = 8^{\circ}$ and $\lambda = 0^{\circ}$.

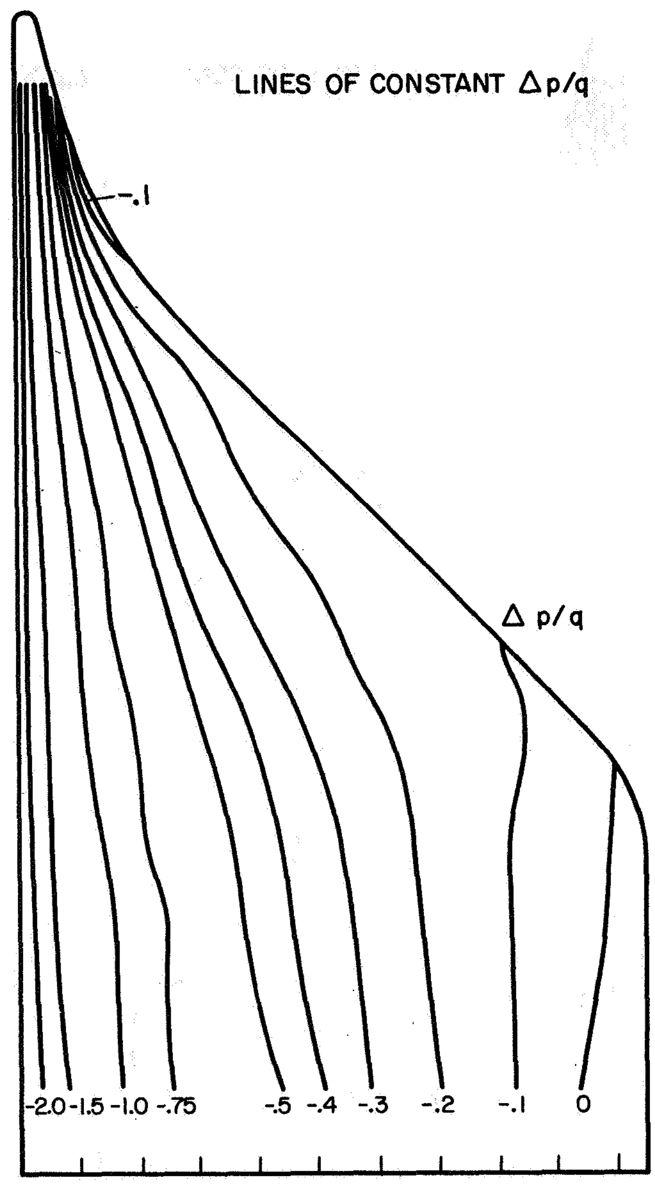


Figure 26.- Contour pressure plot of the Ogee-tip section at $\alpha = 8^{o}$ and $\lambda = -20^{o}$.

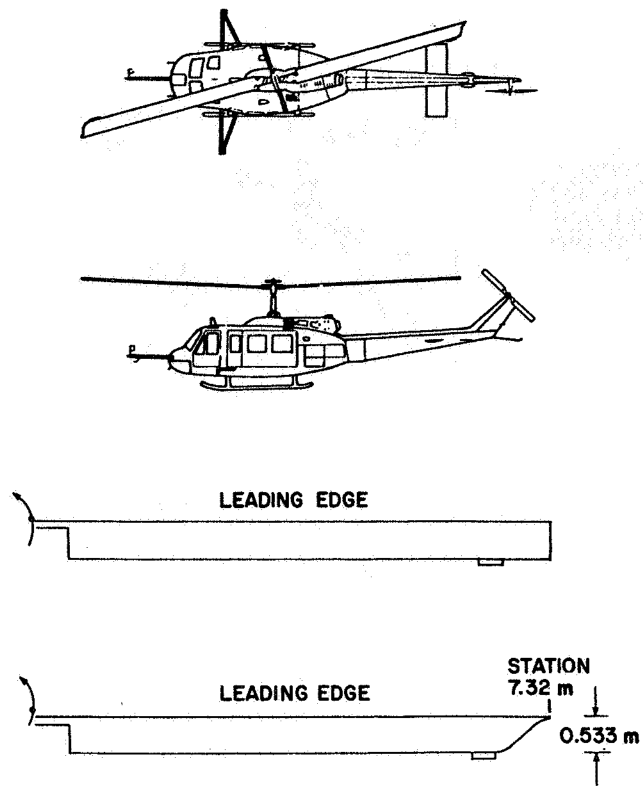


Figure 27.- UH-1H test helicopter.

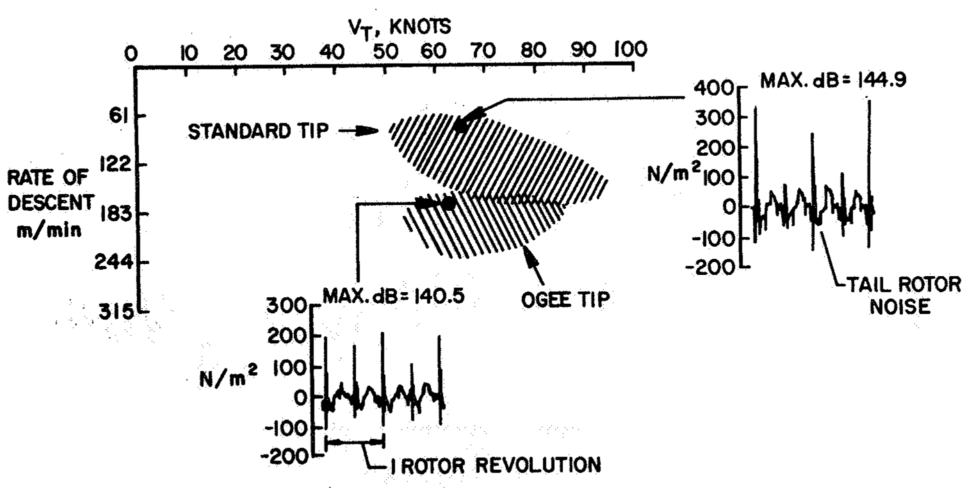


Figure 28.- Peak levels of near-field impulsive noise as measured by IFAMS.

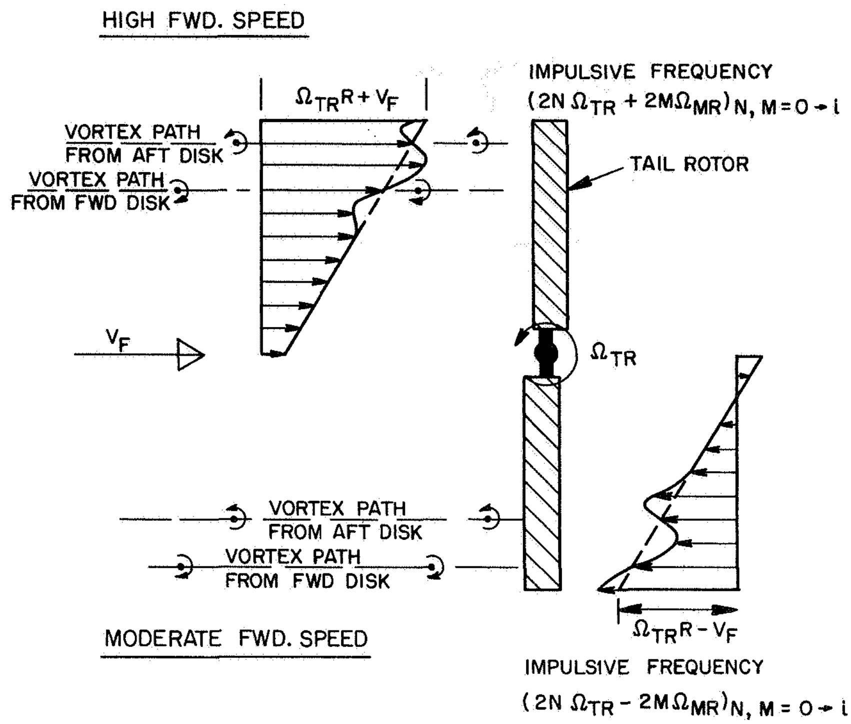


Figure 29.- Schematic diagram of main rotor vortex interactions with tail rotor.

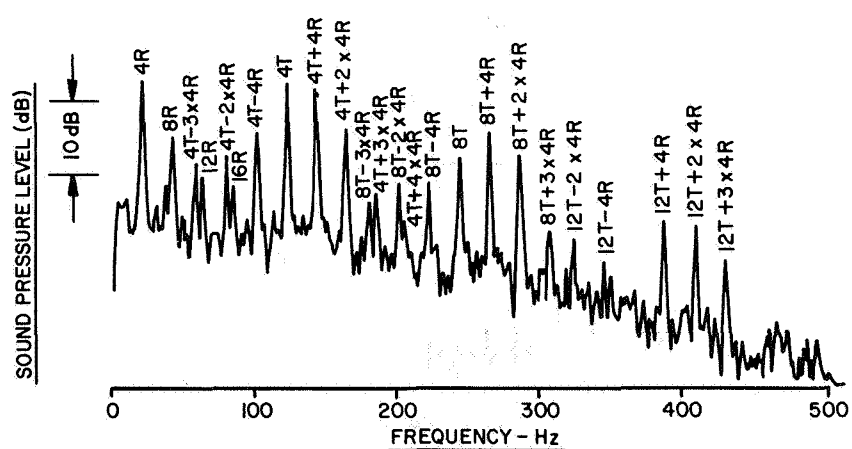


Figure 30.- Tail rotor noise as measured by Westland Helicopters Limited for the Lynx.

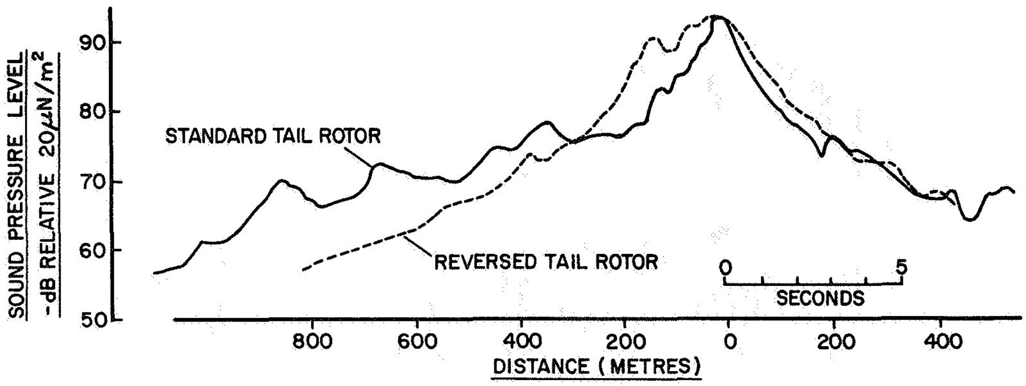


Figure 31.- Comparison of dB(A) time histories for standard and reversed tail rotors at a flyover condition of 50-m altitude and 130 knots.

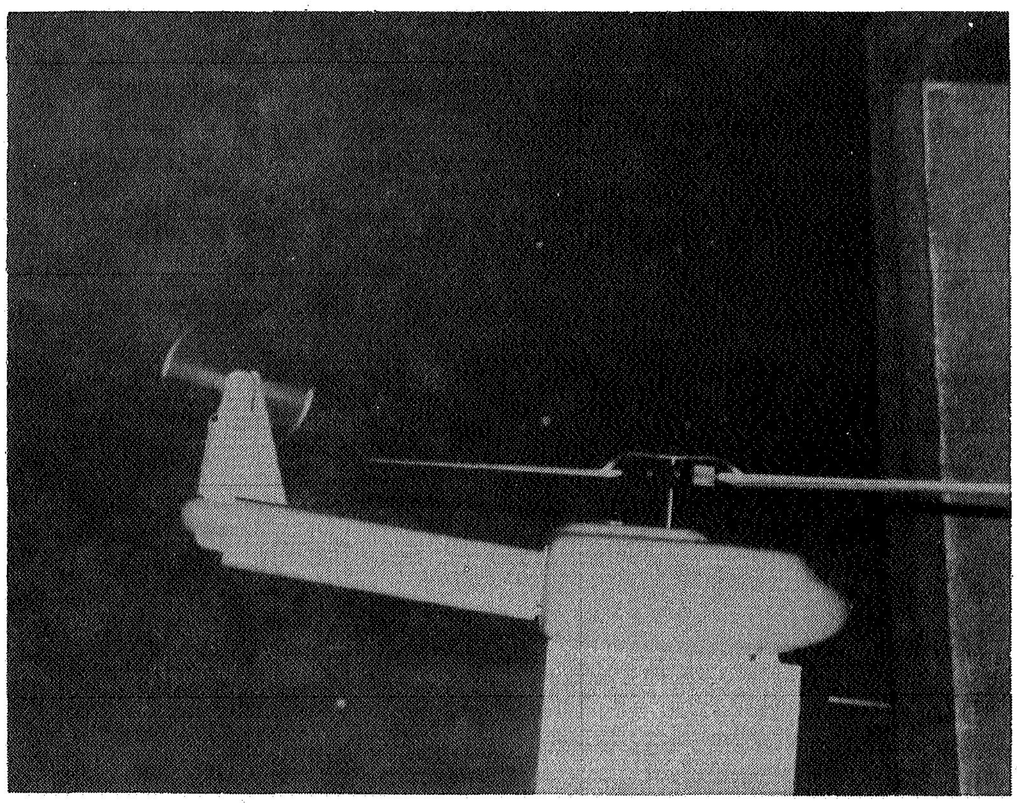


Figure 32.- Main rotor/tail rotor test model.

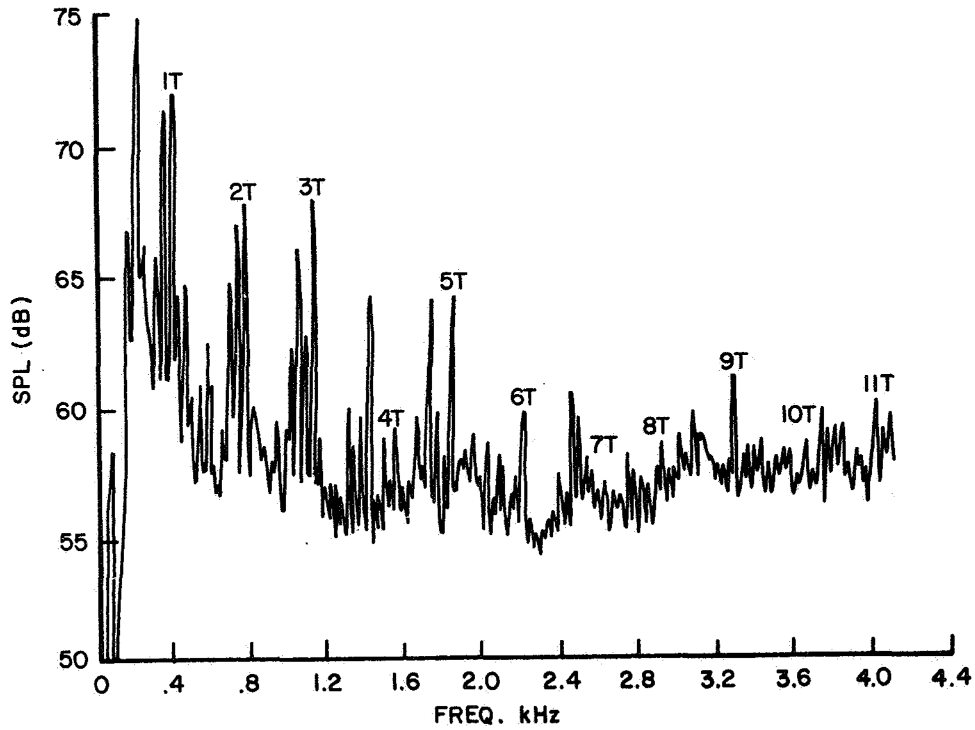


Figure 33.- Noise spectra at $\mu = 0.09$.

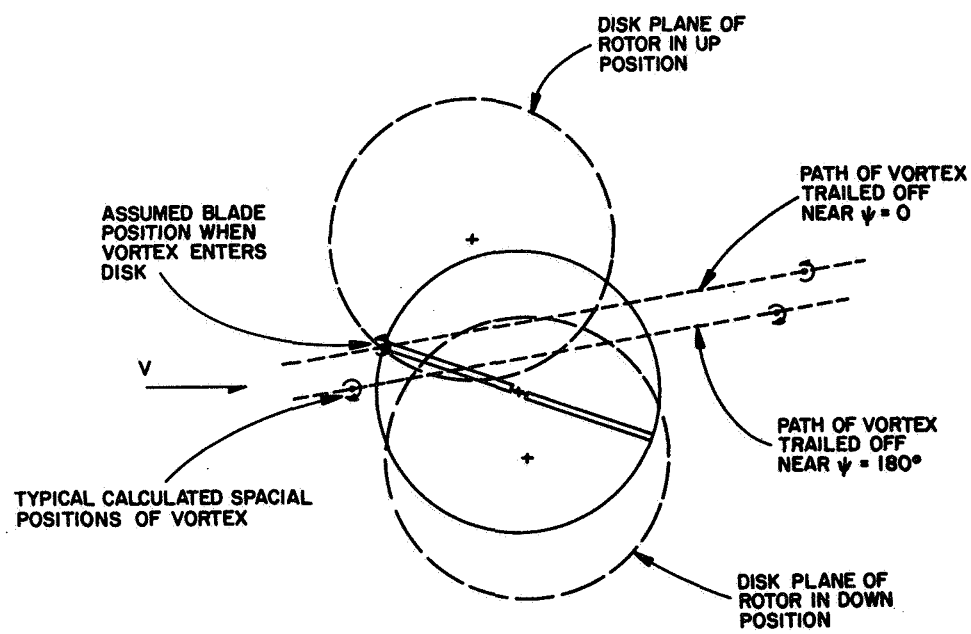


Figure 34.- Main rotor wake/tail rotor interaction at $\mu = 0.20$.

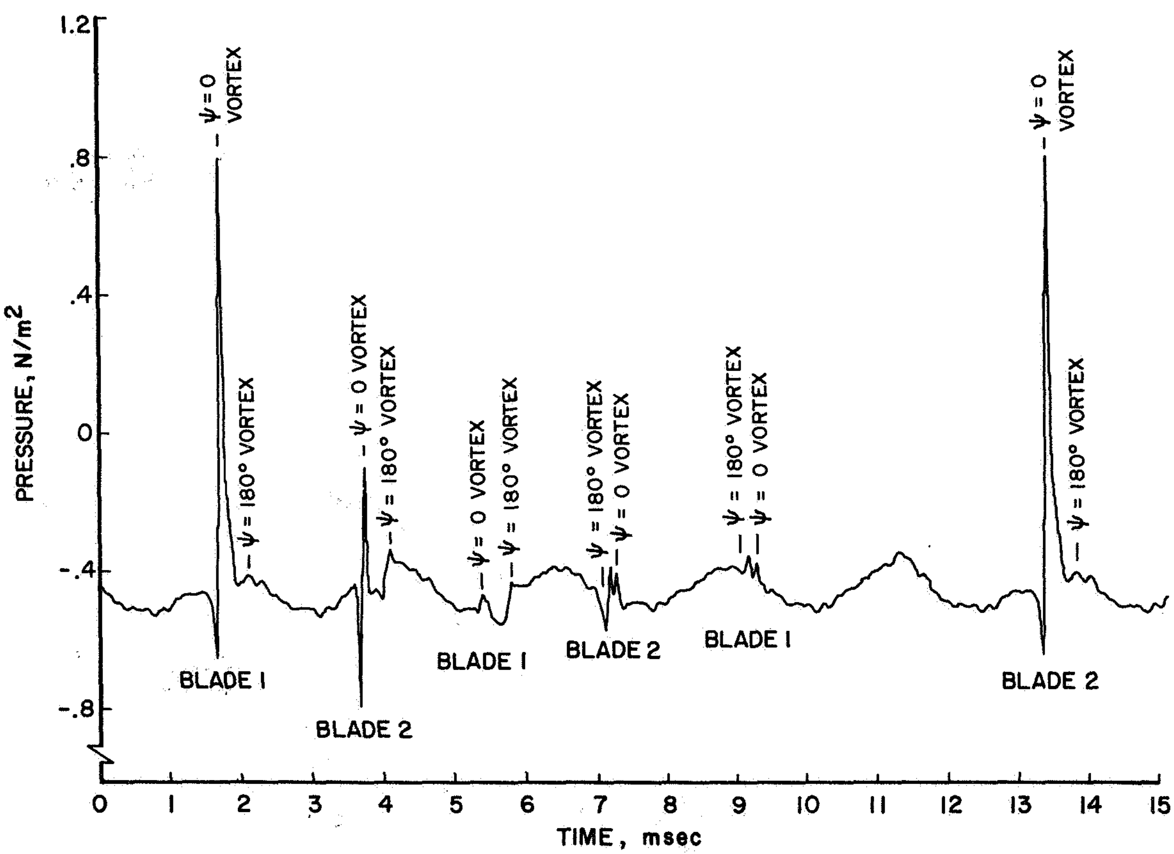


Figure 35.- Calculated sound pressure versus time with periodic main rotor wake interaction at $\mu = 0.20$.

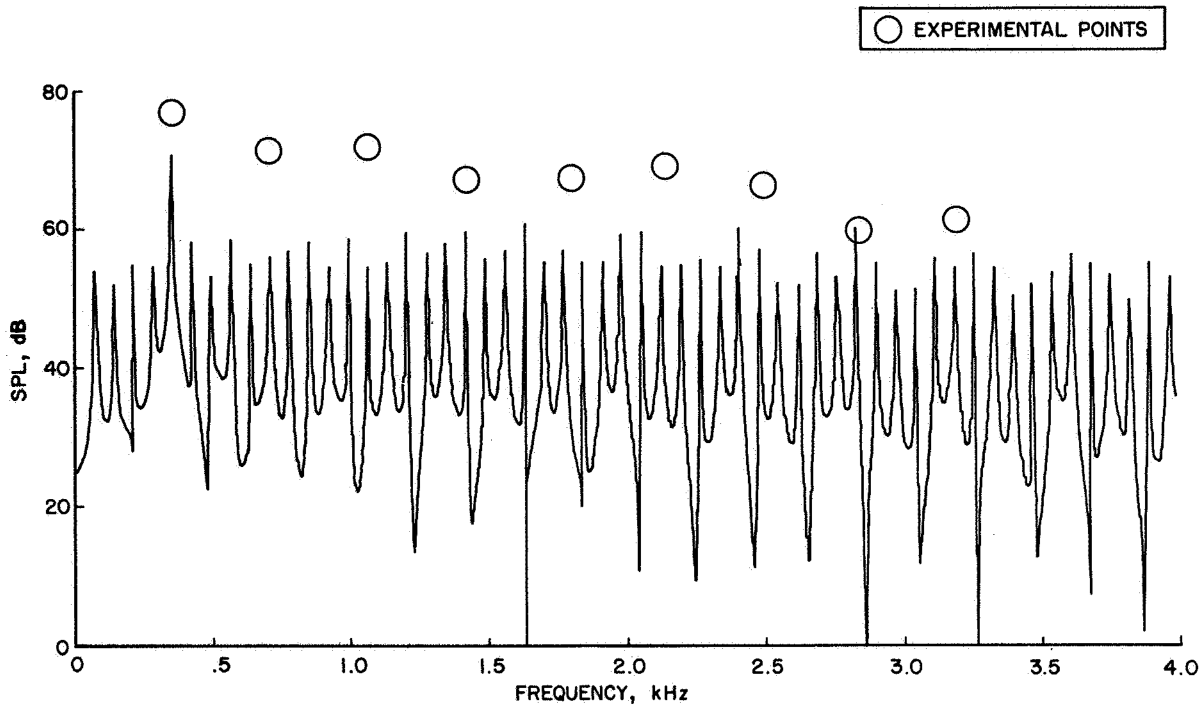


Figure 36.- Calculated noise spectrum with periodic main rotor wake interaction at $\mu = 0.02$.

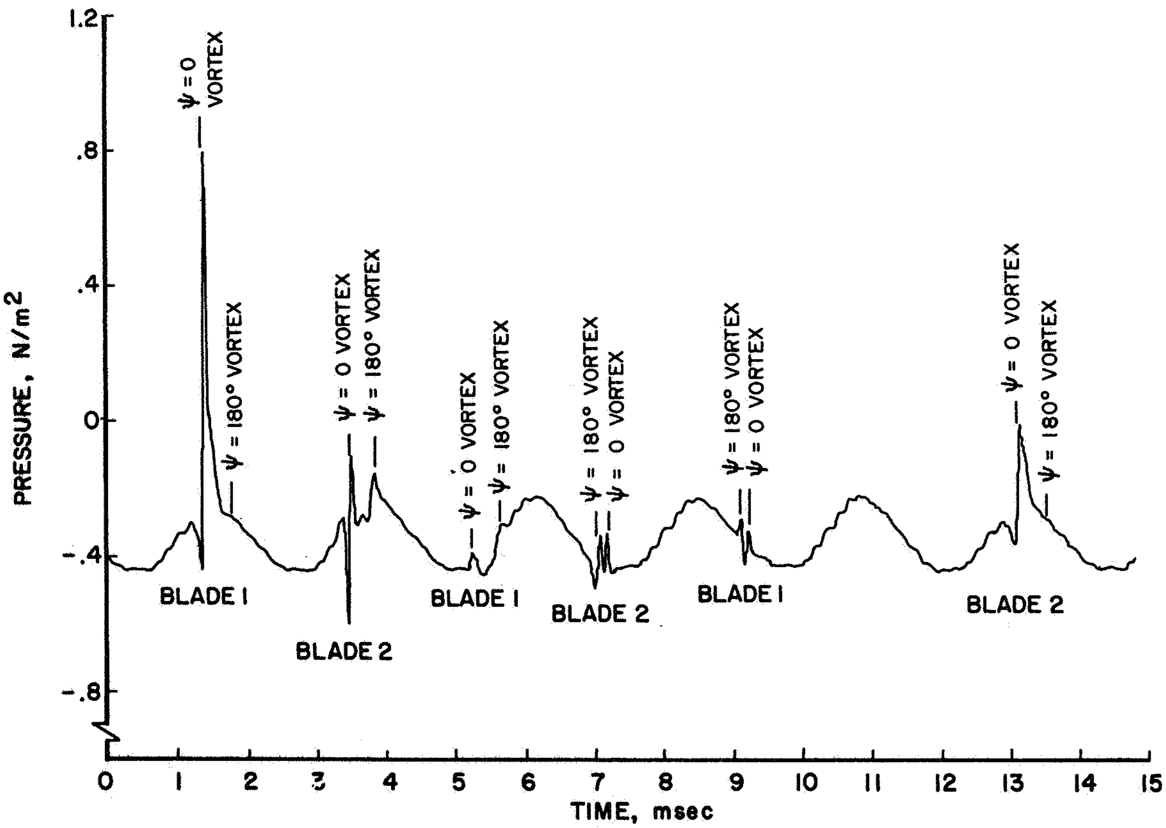


Figure 37.- Calculated sound pressure versus time with nonperiodic main rotor wake interaction at $\mu = 0.20$.

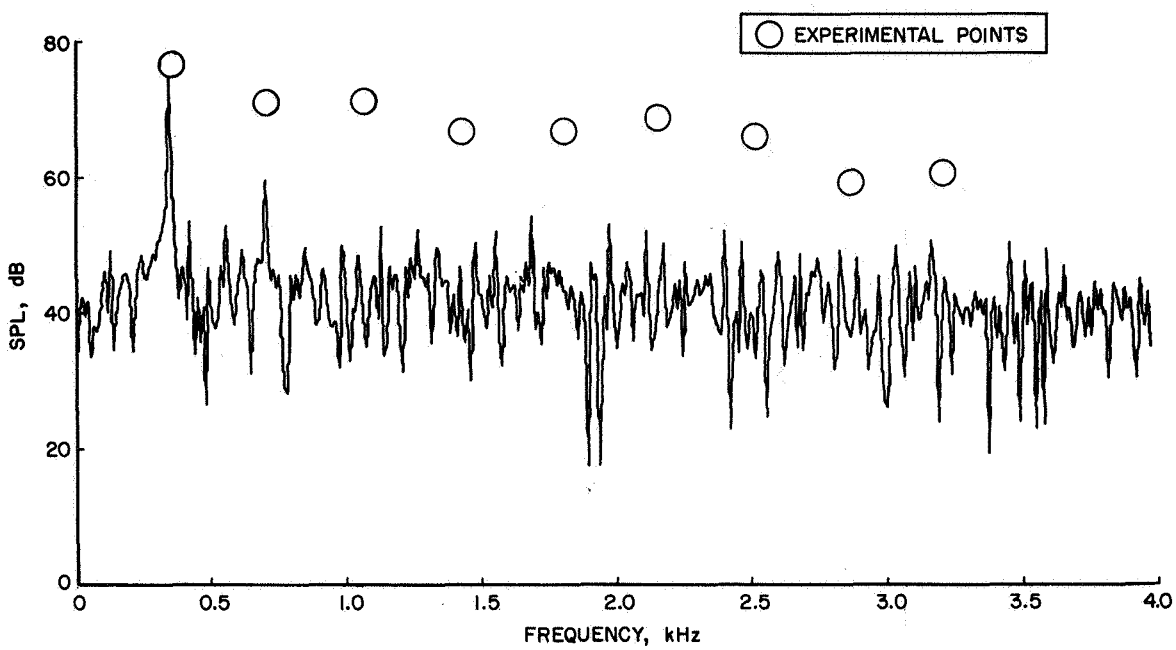


Figure 38.- Calculated noise spectrum with nonperiodic main rotor wake interaction at $\mu = 0.20$.

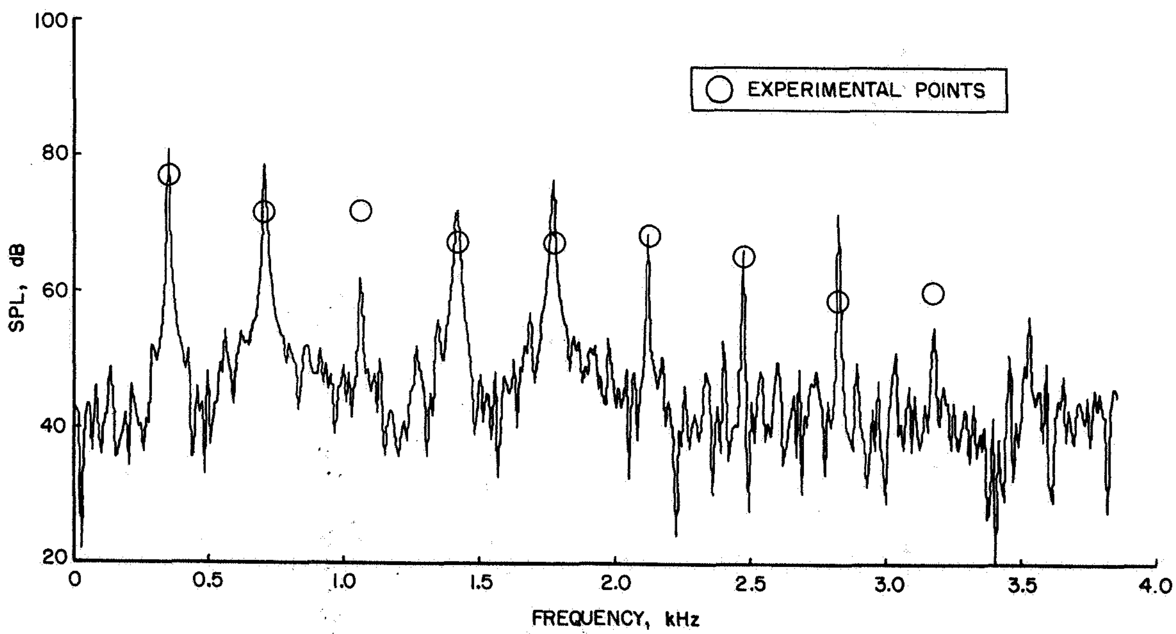


Figure 39.- Calculated noise spectrum with nonperiodic main rotor wake interaction and nonuniform tail rotor wake at $\mu = 0.20$.

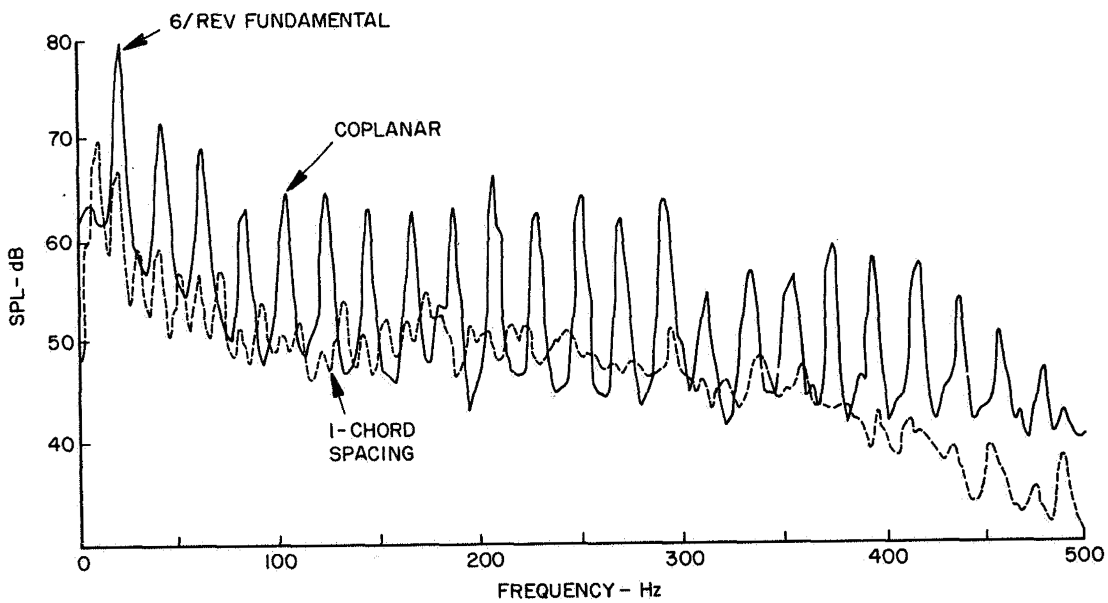


Figure 40.- Effect of rotor dissymmetry on noise spectrum of a six-bladed rotor in hover.