HOVERING IMPULSIVE NOISE

SOME MEASURED AND CALCULATED RESULTS

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SUMMARY

In-plane impulsive noise radiating from a hovering model rotor has been measured in an anechoic environment. The hover acoustic signature was compared with existing theoretical prediction models and with previous forward flight experiments using the same model rotor. These hover tests showed good experimental consistency with forward flight measurements, both in pressure level and waveform character, over the range of Mach numbers tested (0.8 to 1.0). Generally poor correlation, however, was confirmed with current linear theory prediction efforts. Failure to predict both the peak pressure levels and the shape was reported, especially with increasing tip Mach number.

INTRODUCTION

In recent years, high-speed helicopter impulsive noise has been the subject of much acoustic research, both experimentally (refs. 1 to 6) and theoretically (refs. 5 to 11). The main motivation behind the research is the eventual mitigation, by design techniques, of this intense source of noise. The military and civilian communities will both benefit. In the first case, the detection distance of an approaching helicopter will be reduced, while in the second case, quieter helicopter operations will reduce community annoyance.

High-speed helicopter impulsive noise is quite distinctive in nature. In its milder form, it consists of a sequence of thumping sounds which, because of their low frequency character and high intensity, can often be heard for many miles. As the advancing tip Mach number of the helicopter increases the subjective quality, pulse shape, and amplitude of the noise change (refs. 1 and 4). The sequence of acoustic pulses becomes harsh sounding in character and radiates large amounts of higher frequency harmonic noise. The increasing annoyance of a developing saw-toothed pulse shape is also apparent.

Prediction and measurement of this phenomena has proceeded along several different paths. Both frequency-domain and time-domain theories have been developed which emphasize the noncompactness of the problem. Most of the theoretical approaches are derived from the work of Ffowcs Williams and Hawkings (ref. 12) for bodies in high-speed flight. The application of this approach to the rotor problem was first pursued by Farassat (ref. 8) and Hawkings and Lowson (ref. 5). Today there are several basically similar ways of calculating this impulsive noise signature (refs. 5 to 7, 9, and 11) which are in general theoretical agreement and yield numerical values which agree in pulse shape and amplitude.

Unfortunately, the agreement of any of these theoretical approaches with experimental measurements is not at all certain. The issue is often clouded by the different methods of gathering impulsive noise data. For instance, ground noise measurements have shown good agreement in pulse shape and amplitude with thickness noise theories (refs. 6 and 8). Full-scale wind-tunnel measurements of rotors operating at high tip Mach number show good theoretical agreement in the peak levels of the negative pressure pulse but generally poor agreement with theory in pulse shape (ref. 3). Scale model wind-tunnel measurements of reference 4 show generally poor agreement with predicted amplitude and pulse shapes at high advancing tip Mach numbers. Also full-scale impulsive noise measurements of the UH-1H helicopter taken by an in-flight technique (ref. 1) have shown poor agreement in amplitude and in pulse shape at high advancing tip Mach numbers. All of these different measurement techniques have inherent strengths and weaknesses, but all should yield consistent results. As judged by existing theoretical models, however, they do not.

The major purpose of this paper, therefore, is to continue to pursue the comparison between test and theory by investigating a simpler problem, high tip speed hovering impulsive noise. In particular, a 1/7-scale model of the UH-1H helicopter was tested in an anechoic hover environment over a range of Mach numbers consistent with high-speed advancing flight. By avoiding major recirculation through facility design and by testing at high tip Mach numbers, the details of high tip speed noise become clearer. It will be shown from the current tests that the acoustic characteristics highlighted in the previous in-flight full-scale and wind-tunnel scale-model investigations are again present in hover. It will also be shown that existing linear theoretical approaches do not adequately describe the event of high-speed impulsive noise.

EXPERIMENTAL SETUP

The data presented in this paper were gathered in a unique anechoic hover test facility which was designed primarily to gather acoustic and aerodynamic data on hovering rotors. The test chamber has been lined with polyurethane foam and has been designed to be anechoic (without acoustic reflections) down to 110 Hz. As illustrated in figure 1, aerodynamic recirculation is avoided by allowing quiescent air to be drawn into the room through acoustically lined ducts, collecting the wake of the hovering rotor through an annular diffuser, and exhausting the wake to the outside. In its current configuration, the test chamber can accommodate rotors from 1.5 to 2.4 m in diameter.

A final acoustic and aerodynamic calibration of this facility is concurrently being conducted. Preliminary calibrations revealed the feasibility of investigating the noise generated by a high tip speed hovering rotor in the early stages of the facility checkout. Quantitative estimates of noise due to inflow turbulence, detailed measurements of thrust and power, and final verification of the free field characteristics of the chamber are not yet available. Nevertheless, preliminary measurements indicate the room is anechoic to its design frequency and that most of the rotor's wake is captured by the annular diffuser. For the tests to be reported, a minimum amount of collective pitch control (1.5 degrees at the rotor tip) was employed to exhaust the shed wake.

The rotor chosen for the test was a 1/7-scale UH-1H main rotor. The geometrically scaled rotor had a NACA 0012 airfoil section with a root-to-tip washout of 10.9° and a full-scale twist. A teetering hub was employed with cyclic controls locked out for this pure hovering test. A high-speed stroboscopic system was used for blade tracking and visual monitoring during the test. No flutter was apparent throughout the testing matrix. Thrust and torque were monitored, but data can only be considered to be qualitative at this time.

Acoustic data were gathered using 12.7-mm (0.5 in.) "free field" condensor microphones and monitored on an oscilloscope both before and after recording on an FM tape recorder. This insured that the full dynamic range of all the electronic equipment was utilized without clipping of the impulsive signature. A double extended 60 ips mode of the FM recorder gave a transient response of at least 20 000-Hz bandwidth.

EXPERIMENTAL RESULTS

The data reported in this test have been taken with a microphone located within the tip-path plane of the rotor at a distance of 1.5 rotor diameters (r/D = 1.5) from the hub. This in-plane microphone position is consistent with previous in-flight and wind-tunnel testing (refs. 1 and 4) and is in a position to measure the most intense high-speed impulsive signature. As stated in reference 11, another benefit of utilizing an in-plane microphone is that the measured signal will only theoretically depend upon acoustic dipole sources whose major axes are in the plane of the rotor disc (i.e., the thrust dipole does not contribute, only in-plane drag forces). Therefore, comparison of theory and experiment is not dependent upon detailed measurement of rotor thrust.

Figure 2 presents the measured acoustic signature at a hover tip Mach number M_T of 0.8. Two time scales are presented. Part (a) depicts two blade passages, approximately one-half of a complete rotor revolution. Part (b) of the same figure is an expanded scale of the first acoustic pulse. The latter is used to emphasize the detailed waveform characteristics of the measured pulse.

The waveform presented in this figure (and throughout this report for all test conditions) is unaveraged and exhibits some degree of unsteadiness both in peak negative pressure level and in the finer waveform shape structure. An investigation into the sources of this unsteadiness is one of the purposes of ongoing tests. The most striking feature of the waveform at M_T = 0.8 is its almost symmetrical character. This same character has been observed in full-scale and scale-model forward flight testing. The only real difference when the same model rotor is tested in forward flight is that the peak negative amplitude of the pulse is higher in the hovering condition, as would be predicted.

Figure 3 illustrates the pressure time history at a hovering tip Mach number of 0.9. The peak negative amplitude of the measured pulse has increased dramatically and the pulse shape has now lost its symmetry. The resulting saw-toothed waveform is known to generate large amounts of high intensity, higher frequency noise. Again, this same type of waveform was measured on the same rotor system operating in forward flight at an advancing tip Mach number of 0.9. In this previous test, schlieren photographs were used to correlate the discontinuous increase in pressure with a radiating shock wave. It is apparent that a similar phenomenon is occurring in this controlled hover test.

At a hover tip Mach number of 0.962 (fig. 4), the saw-toothed pulse shape is firmly established and the negative peak pressure level has doubled from the $M_T = 0.9$ condition. The large, discontinuous rise in pressure resulting from a radiating shock wave exhibits some variability from blade to blade. One particularly interesting aspect of the waveform shown in figure 4(b) is the pulse width. At lower hover tip Mach numbers, the pulse width was observed to narrow with increasing rotor tip speed up to the point of waveform transition from symmetrical to sawtooth. Above this transition point, for example at $M_T = 0.962$, the pulse width has become larger. Figure 4 also shows that a positive pressure wave (bow wave) begins to form. At still higher tip Mach numbers, the classical N-wave of sonic boom research is likely.

In addition to the general increase of peak negative pressure level with increasing hover tip Mach number, the waveform transition from symmetrical to sawtooth dominates the changing acoustic signature. Figure 5 illustrates the development of the radiating waveform discontinuity as measured in hover at $r/D = 1.5$. The sequence of waveforms in figure 5 shows that transition occurs over a very small range in hover tip Mach number from 0.88 to 0.90, with $M_T = 0.89$ being the point of transition for the test rotor. Transition was found to be characterized by a simultaneous increase in negative peak pressure level and the following rapid pressure rise. Both events were observed to be highly unsteady even under controlled rotor test conditions.

It is also instructive to compare the peak negative amplitude of the measured waveform versus hover tip Mach number (fig. 6). A very rapid increase in level is noticed as M_T approaches 0.9. However, as M_T increases beyond 0.9 to $M_T = 1.0$, the increase in peak level is less or the rate of increase of this peak negative pressure level with Mach number becomes smaller. As noted on figure 6, the shaded area depicts the degree of unsteadiness (mentioned previously) in the measured data. The vertical solid bars reflect data taken with the UH-1H model twisted blades, and the "dashed" vertical bars are for the same dimension model rotor using untwisted blades. The correlation between twisted and untwisted results is good with the exception of unsteadiness at $M_T \approx 1.0$ for the twisted blades. The reason for this difference is not known at this time. No apparent flutter was visualized for either set of blades.

The peak levels versus advancing tip Mach number measured on the same model rotor in forward flight (ref. 4) are also shown in figure 6. Although the rate of increase is similar, the peak levels are much smaller in amplitude as would be expected.

COMPARISON WITH THEORY

Figures 7 to 10 compare the measured hover results with the linear non-compact acoustic models developed in the literature. In this case, the methods of reference 11 were used to calculate the pulse shape. Only monopole thickness terms were included, because local forces in the in-plane direction do little to affect the calculated signature. The linear dipole and quadrupole refinements as well as details of the often transonic flow field have been neglected. An "acoustic planform" approach (ref. 7) was used to calculate the waveform time history at or near $M_T = 1.0$ and to check theoretical computations at lower Mach numbers. There is nothing really new in these computations of "thickness noise" at the current time as there are many existing programs which could produce similar results. In the following comparisons, no exact attempt was made to phase match the theoretical and experimental acoustic signatures.

The striking features of the comparison between theory and experiment in hover at $M_T = 0.8$ (fig. 7) are the similarity in pulse shape and the discrepancy in peak pressure levels. As in forward flight (ref. 4) at advancing tip Mach numbers below 0.9, thickness noise theory misses the measured negative peak levels by a factor of approximately two.

The comparison of theory and experiment as M_T is increased to 0.88 (fig. 8) remains similar to that made at $M_T = 0.8$. The waveform shape is still generally symmetrical but the peak negative pressure level is underpredicted by slightly more than a factor of two. As was noted previously, $M_T = 0.88$ is slightly less than the critical hover tip Mach number for waveform transition, at least as measured at r/D = 1.5 with the test rotor.

At a hover tip Mach number of 0.9, the situation becomes even worse (fig. 9). The amplitude of the peak negative pressure pulse is again underpredicted by a factor of approximately two. However, as indicated previously, there is also a dramatic change in the waveform of the experimental data which is not predicted by the linear theory.

The comparison becomes even more intriguing at a hover tip Mach number of 0.962 (fig. 10). The theoretical waveform is still symmetrical and generally smooth in shape and thus does not compare favorably with the measured data. In addition, theory now only slightly under-predicts the peak negative pressure amplitude of the pulse. Also as previously noted, the measured pulse width is becoming wider; whereas, the linear theory predicts a more narrow pulse width with increasing hover tip Mach number. In fact, the experimental pulse width (measured at zero pressure) exceeds by at least 50% the width expected (by linear theory) from an airfoil of chord equal to the model rotor tested and traveling at sonic velocity. This pulse widening effect suggests that aerodynamic events off the rotor blade are contributing to the measured acoustic signature.

The difference in peak negative pressure levels between linear monopole theory and experiment can be seen more clearly in figure 11. Clearly, the theoretical model does not predict the rate of increase of the peak negative pressure level. However, at a hover tip Mach number of 0.97, the two curves cross. This fact may be a partial explanation for the generally good correlations between theory and experiment reported in reference 3.

The theoretical predictions utilized in this paper have only considered the linear monopole source contributions. It was shown in references 2 and 4 that forward flight in-plane impulsive noise was not (to first order) dependent upon thrust. Similarly, in this test, no first-order dependence of thrust (and therefore drag) was observed. The inclusion of in-plane quadrupoles will tend to improve the correlation. However, as pointed out in reference 11, a more sophisticated treatment of rotor transonic aerodynamics is undoubtedly necessary. For quantitative comparisons, it may be necessary to reformulate the basic acoustic equations to capture these transonic effects.

CONCLUDING REMARKS

The preliminary data taken in a controlled aerodynamic and acoustic environment on model rotors have shown that there are many fundamentals of rotor noise that are on the verge of discovery. The development of the anechoic rotor test facility is a valuable asset in that direction.

It is apparent that there exists a major discrepancy between existing linear theoretical approaches to the high-speed noise problem and experimental measurements. The use of monopole thickness and dipole drag terms in the theoretical expressions does not predict the trend of increasing noise levels with Mach number. It also only predicts the correct waveform below hover tip Mach numbers of 0.89 for the test rotor. It is concluded that the use of these theoretical approaches in the design of rotors at high tip speed is somewhat premature, their applicability has yet to be quantitively demonstrated.

The underprediction of impulsive noise at low hovering Mach numbers (0.8) is not understood. An additional large source of noise appears to be present which has been omitted from the theoretical analysis to date. The overprediction of impulsive noise at high Mach numbers ($M_T \approx 1.0$) is not quantitatively describable. However, as in transonic fixed-wing aerodynamics, this might qualitatively be explained by arguing that local transonic effects weaken the radiating sound wave.

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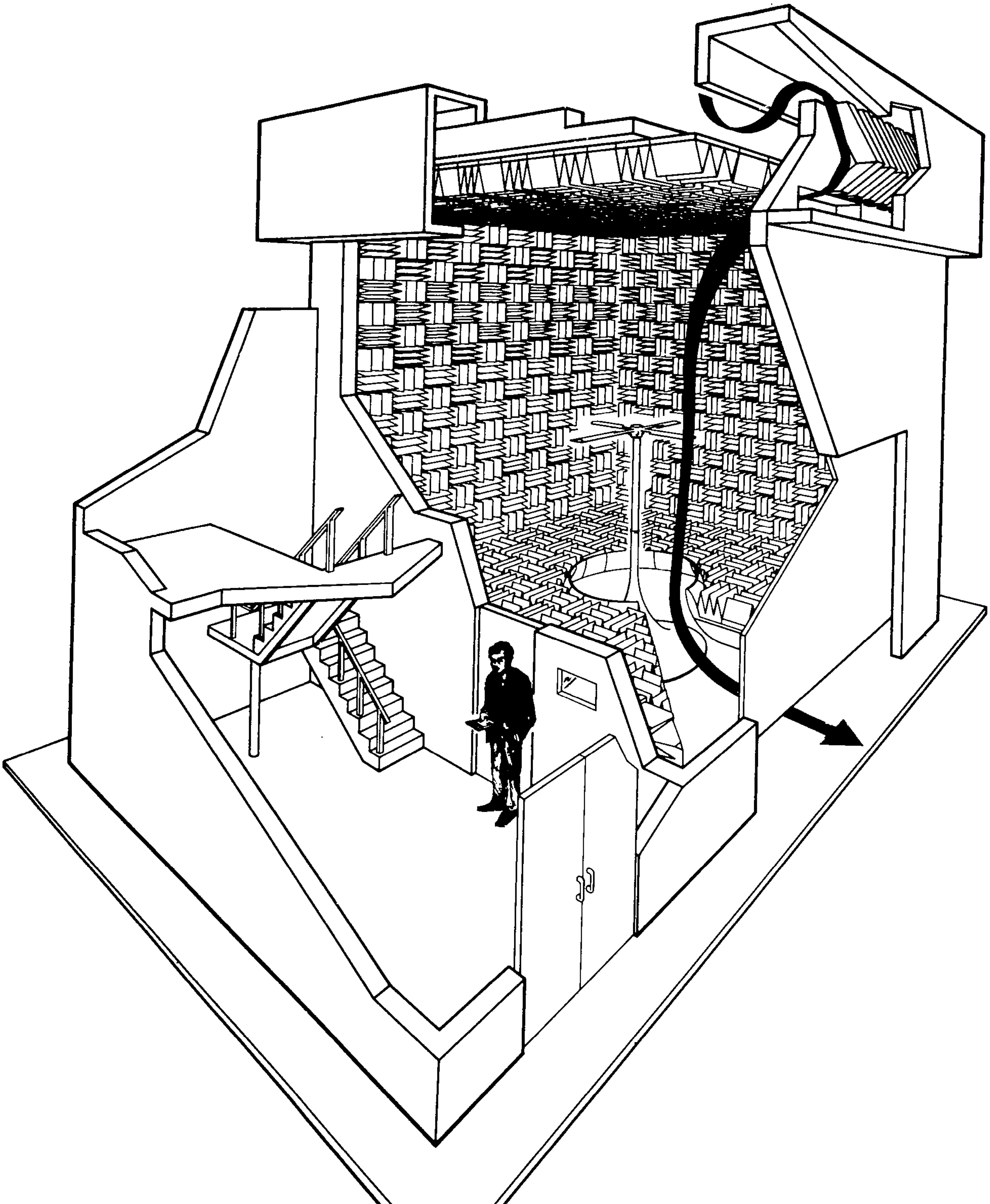


Figure 1.- Anechoic rotor hover test facility.

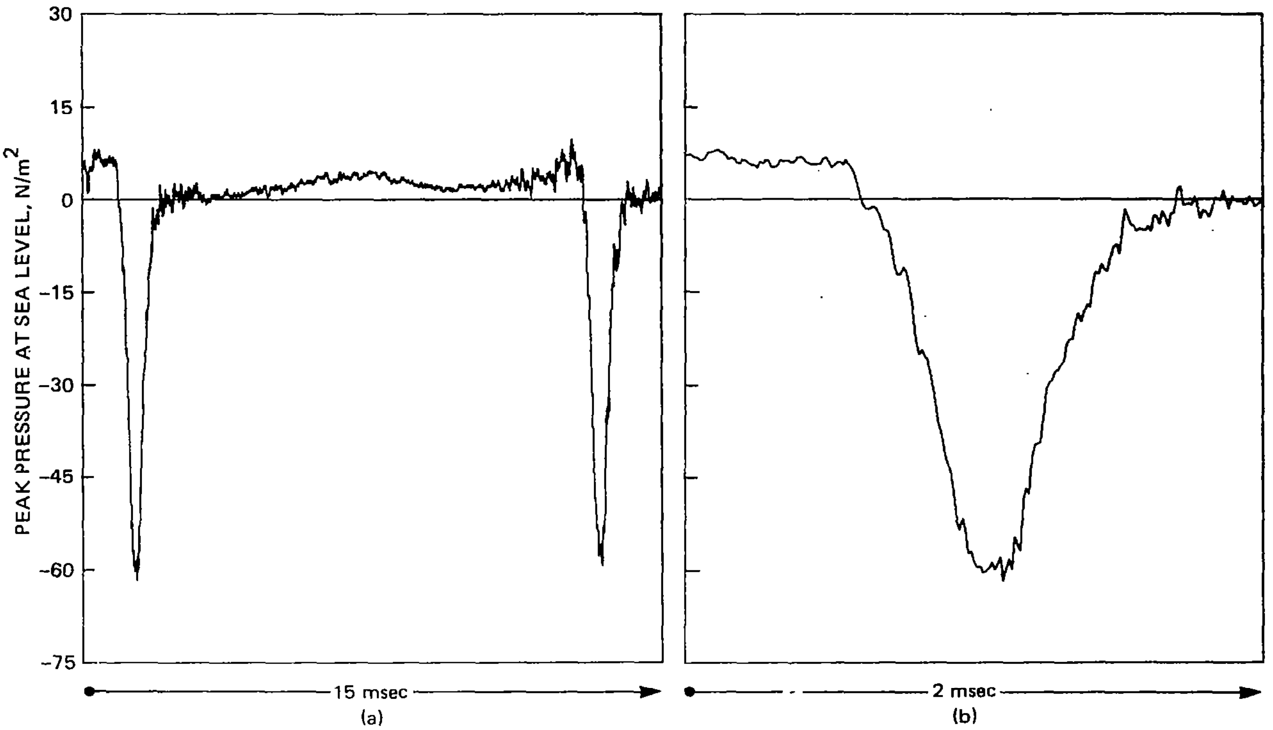


Figure 2.- Acoustic pressure-time history, in-plane, $r/D = 1.5$, $M_T = 0.8$.

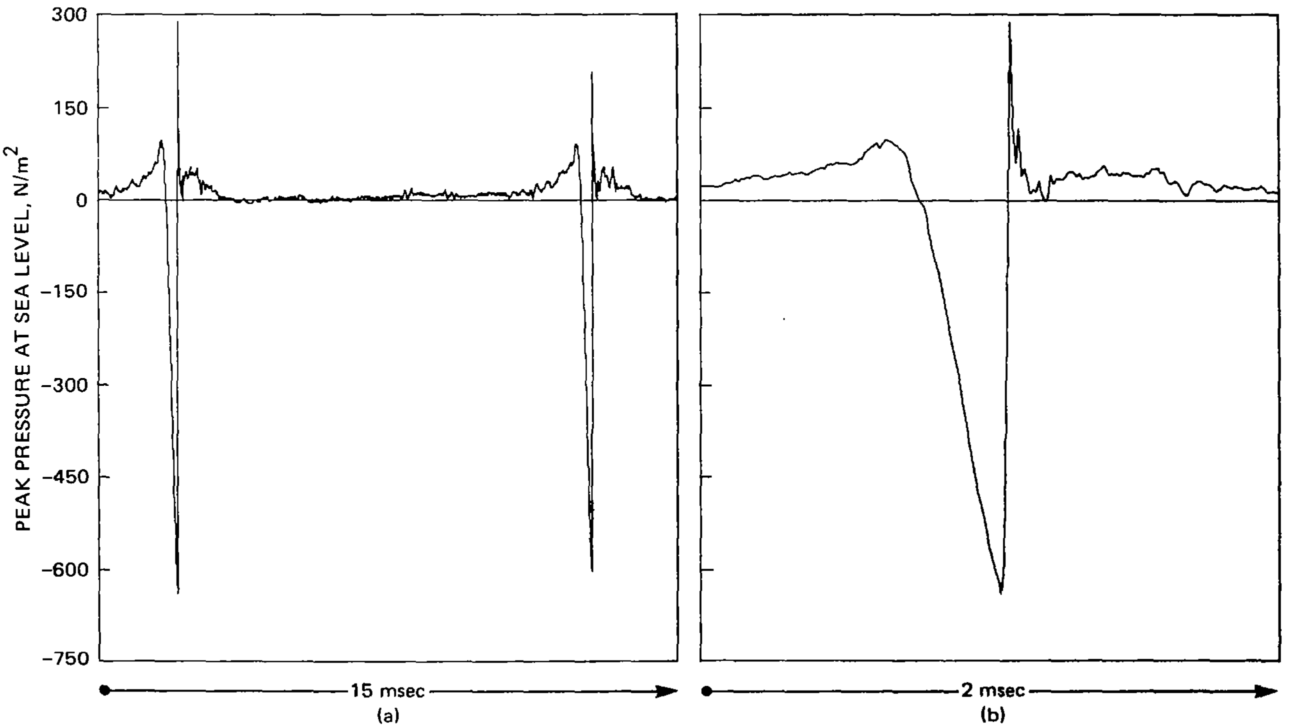


Figure 3.- Acoustic pressure-time history, in-plane, $r/D = 1.5$, $M_T = 0.9$.

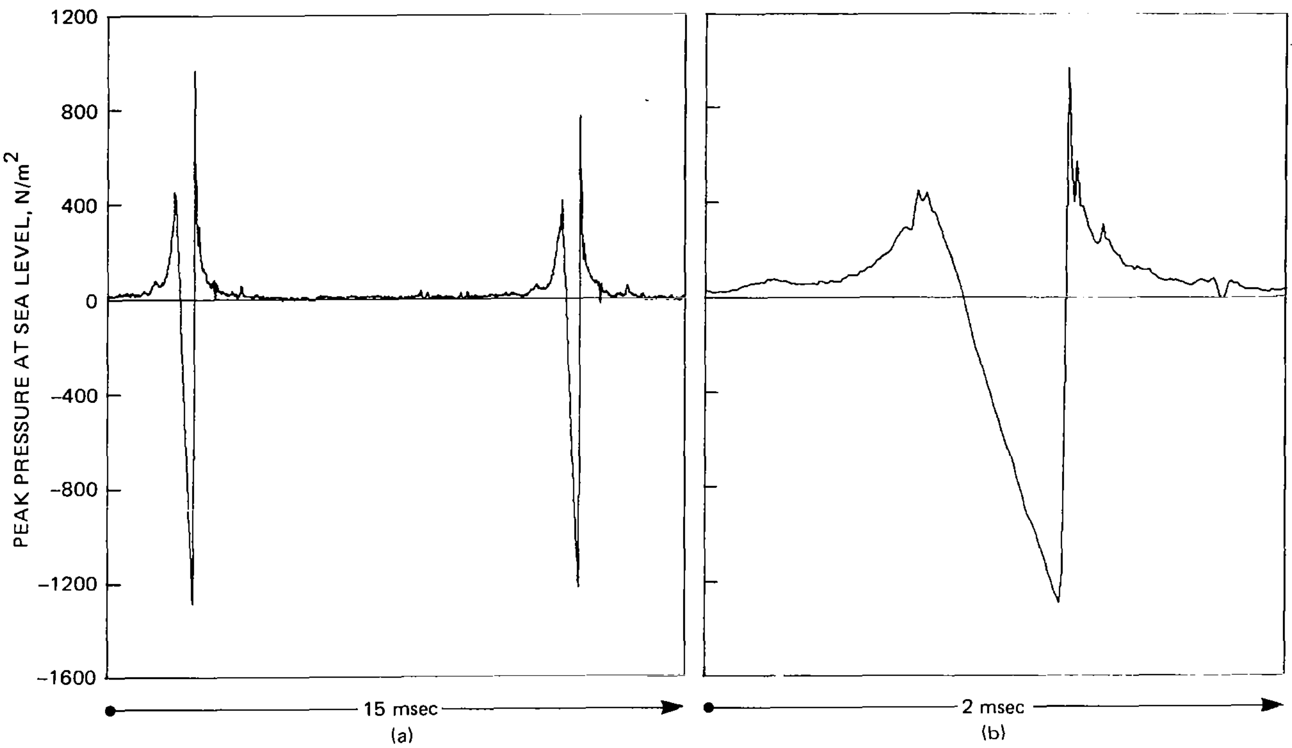
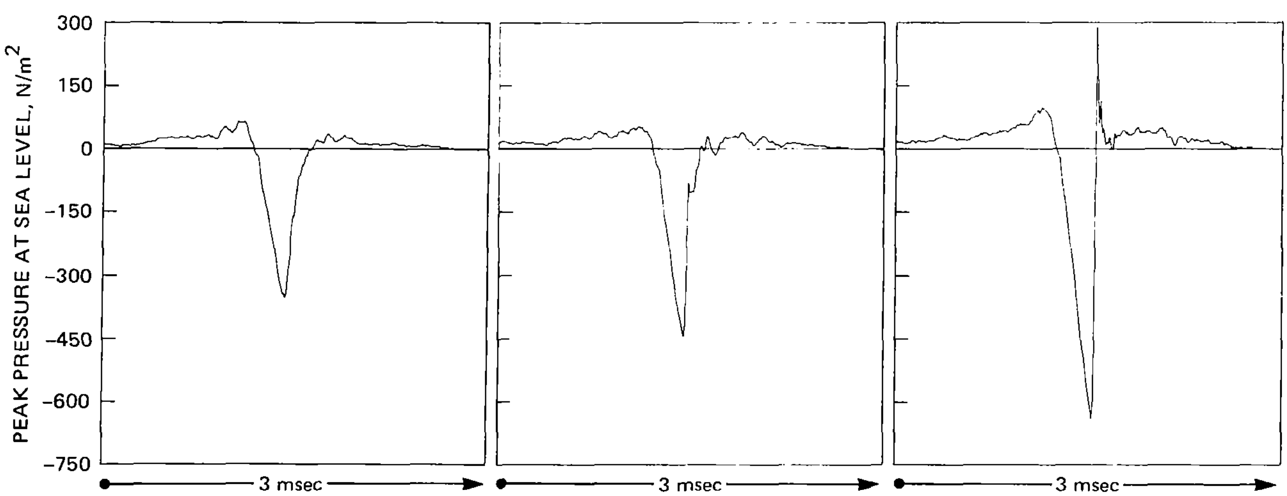


Figure 4.- Acoustic pressure-time history, in-plane, r/D = 1.5, M_T = 0.962.



(a) M_T = 0.88. (b) M_T = 0.89. (c) M_T = 0.90.

Figure 5.- Waveform transition — the development of a radiating discontinuity, in-plane, r/D = 1.5, M_T = 0.88 to 0.90.

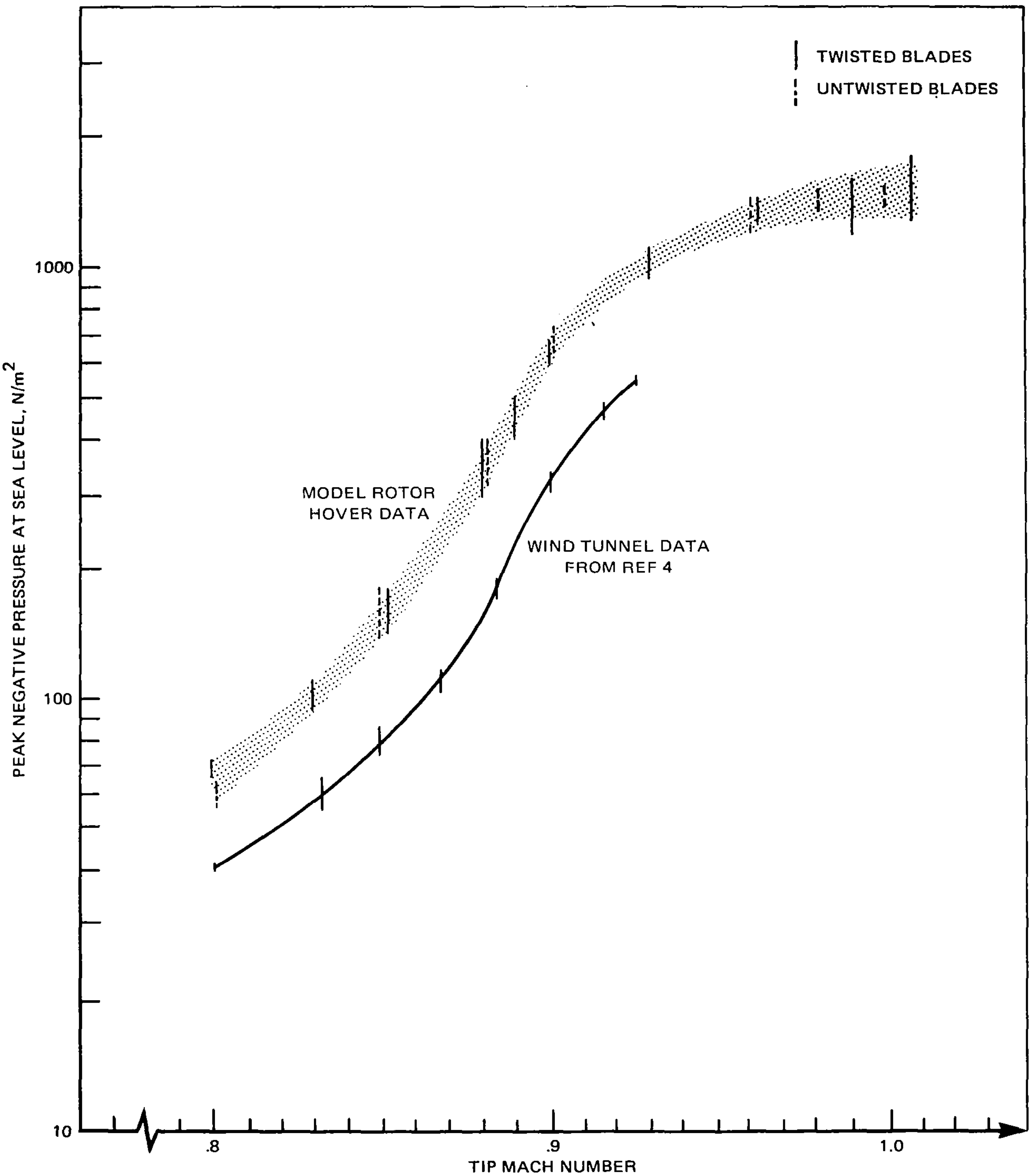


Figure 6.- Peak negative pressure level vs. tip Mach number for model rotor in hover and forward flight, in-plane, r/D = 1.5.

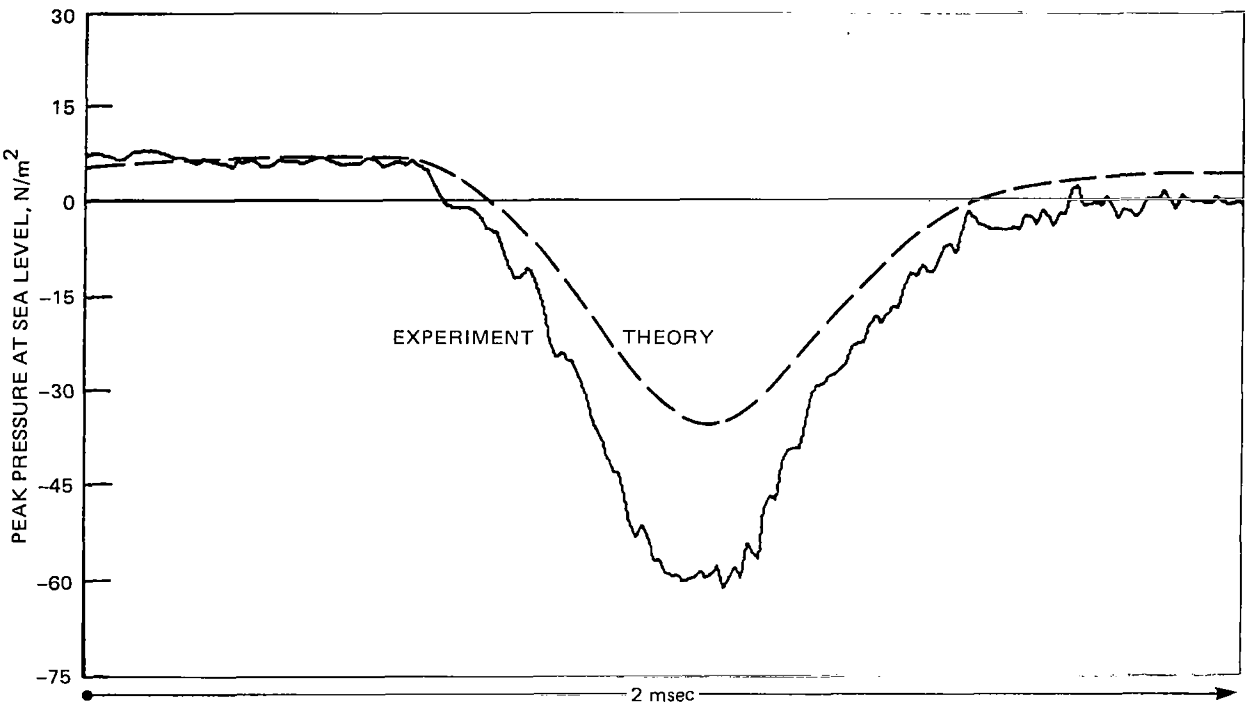


Figure 7.- Comparison of theory with experimental pressure-time history, in-plane, r/D = 1.5, M_T = 0.8.

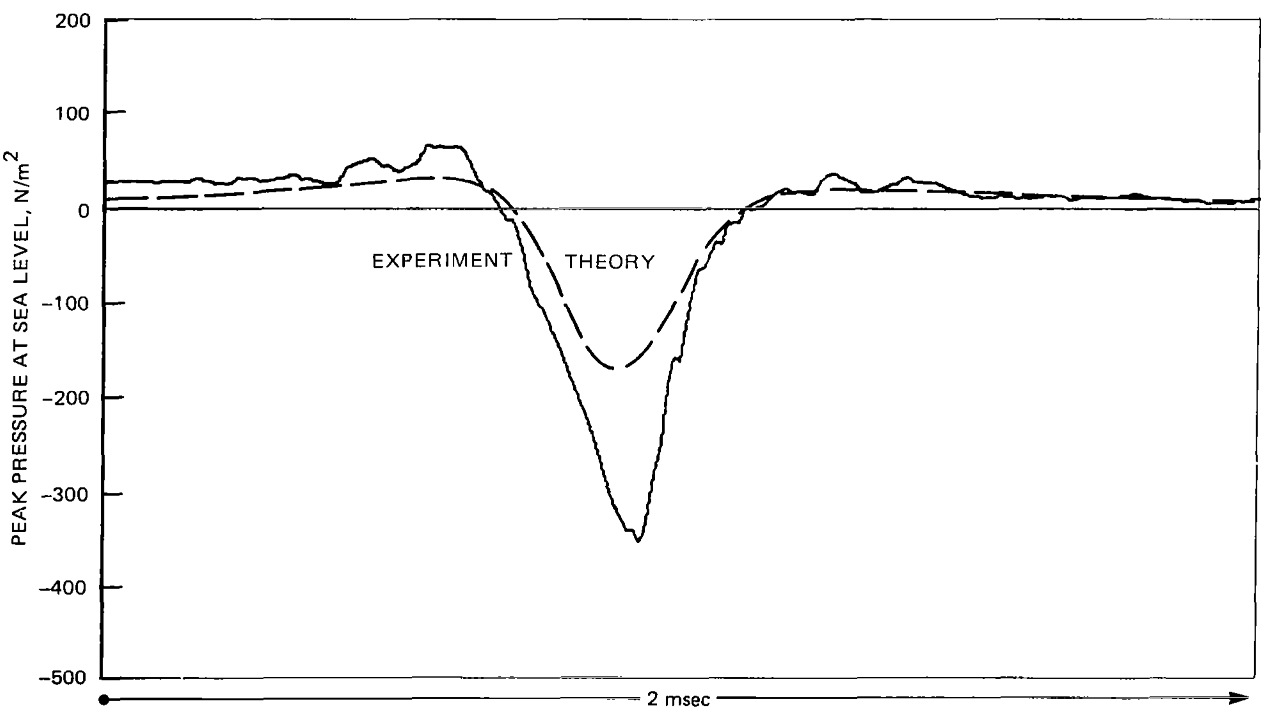


Figure 8.- Comparison of theory and experimental pressure-time history, in-plane, r/D = 1.5, M_T = 0.88.

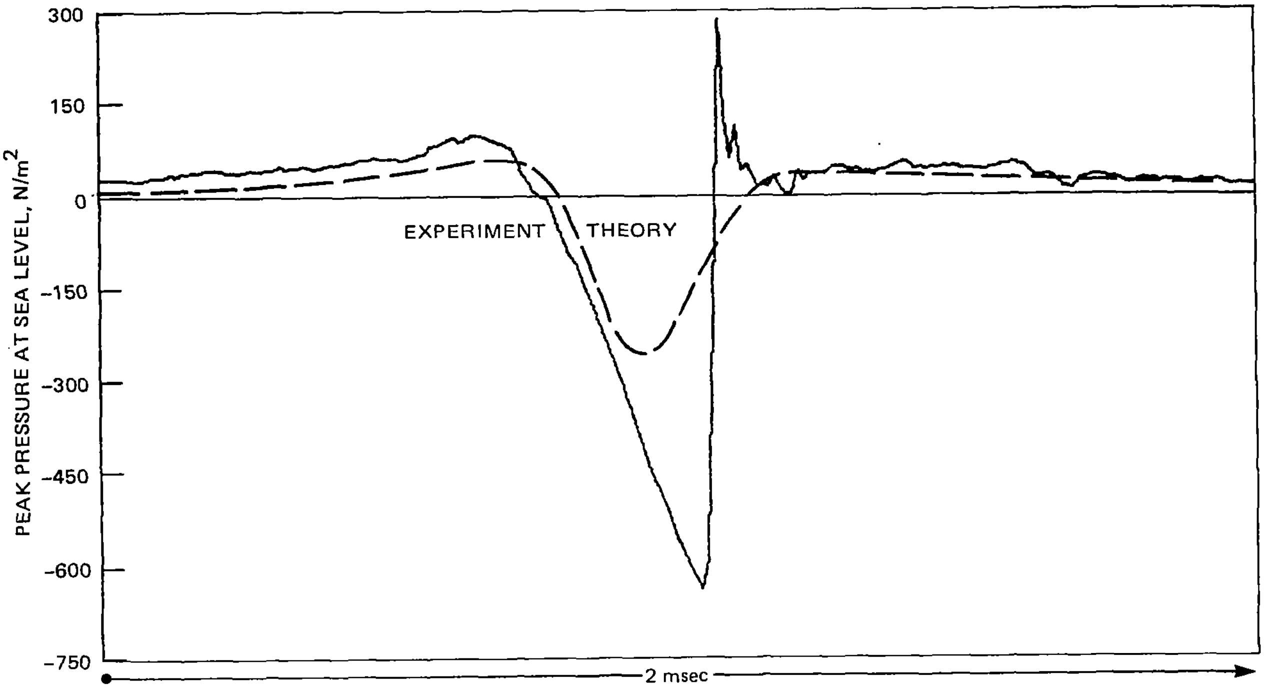


Figure 9.- Comparison of theory and experimental pressure-time history, in-plane, r/D = 1.5, M_T = 0.9.

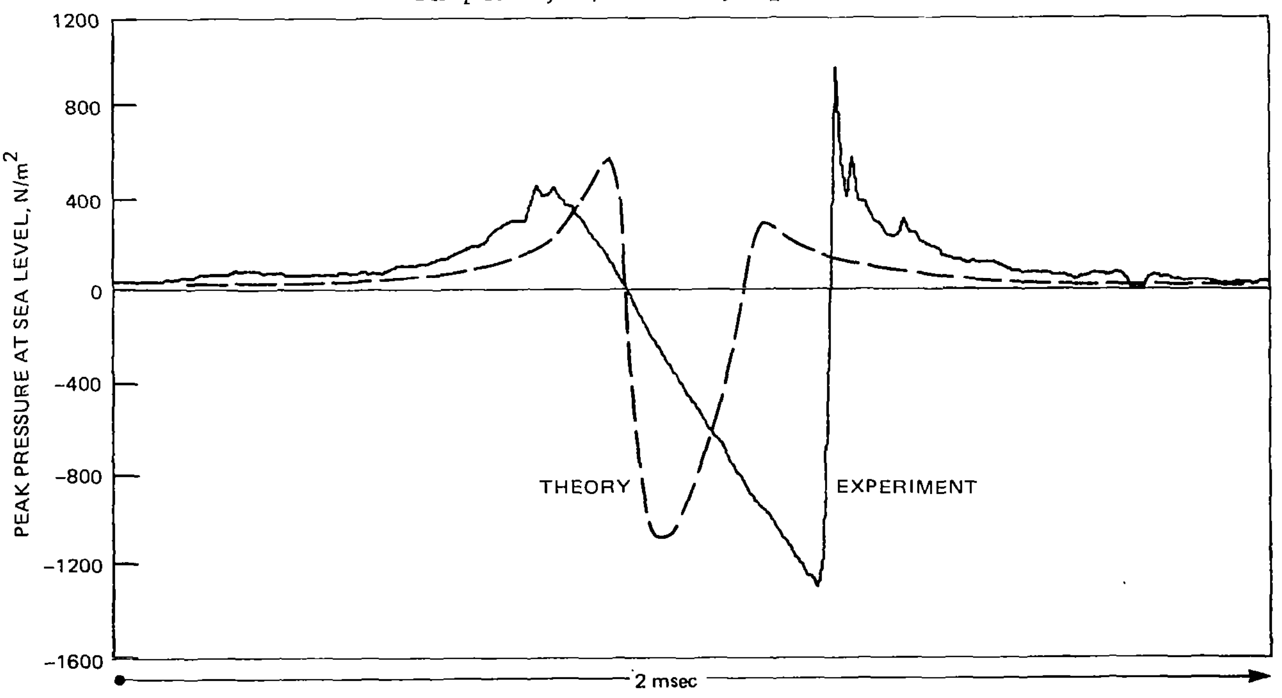


Figure 10.- Comparison of theory and experimental pressure-time history, in-plane, r/D = 1.5, M_T = 0.962.

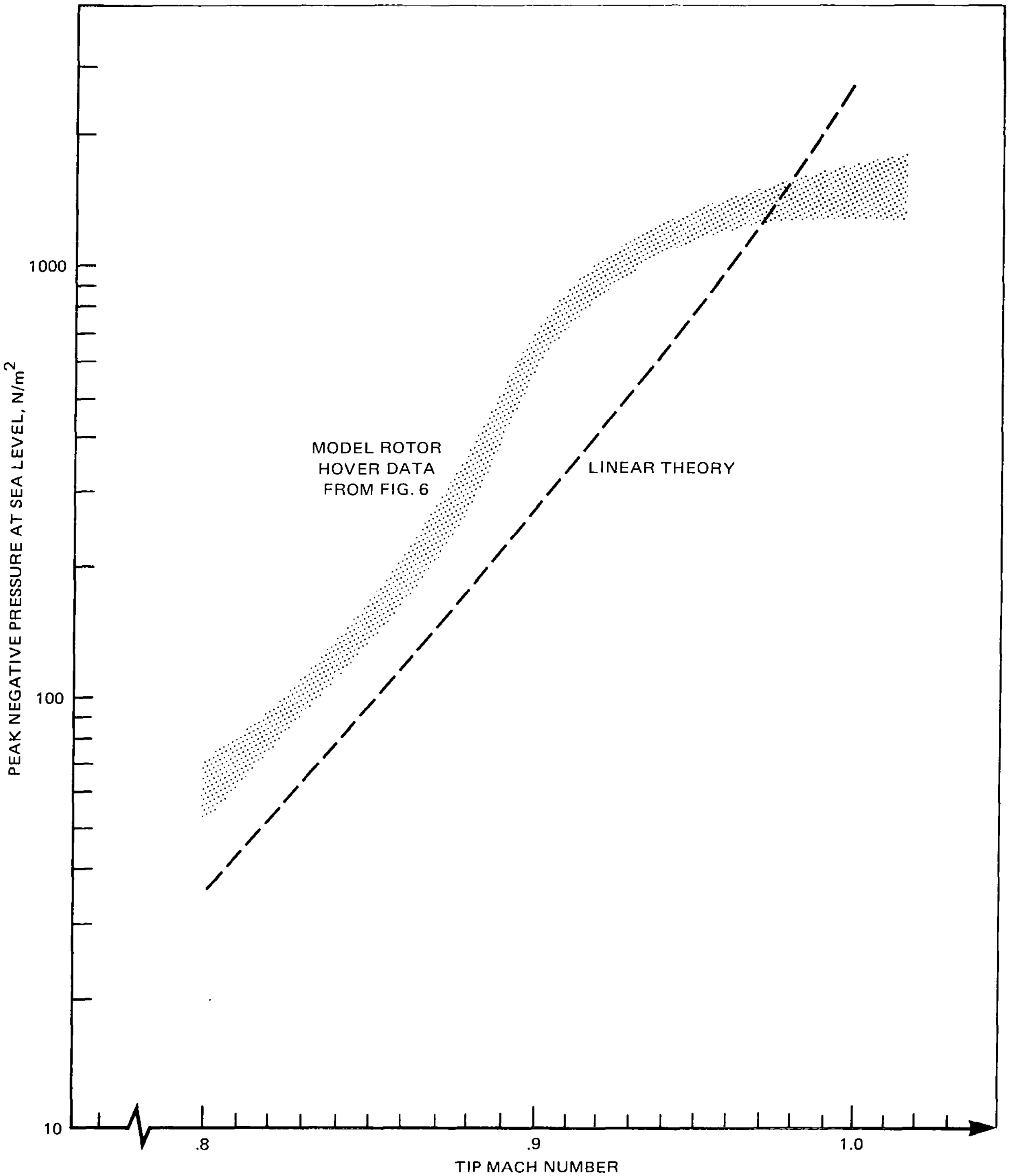


Figure 11.- Comparison of theory and experiment in hover, in-plane, r/D = 1.5.