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WIND TUNNEL TESTS OF A TWO BLADED MODEL ROTOR TO EVALUATE THE TAMI SYSTEM IN DESCENDING FORWARD FLIGHT

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SUMMARY

The tests of the TAMI System in the wind tunnel using a two bladed teetering rotor model indicated that this system can significantly reduce the noise output due to blade vortex interaction. In addition to this overall conclusion, the following specific observations were noted:

(1) The increased noise level in transition during steady state level flight is believed due to vortex induced wake turbulence and not to blade-vortex intersections.

(2) The vortices that are intersected by a following blade to produce blade slap are generated in the second quadrant within a $\Delta\psi$ range of approximately 30 degrees.

(3) The blade vortex intersections occur at the same azimuthal stations on the advancing side regardless of advance ratio or C_T.

(4) The total power required for TAMI, while more than desired, is approximately 15% of the installed power.

(5) Based on the results obtained it is believed that the effectiveness of the TAMI system in alleviating blade slap noise can be increased and that the power requirements can be cut at least in half.

On the basis of the results that were obtained it is recommended that a full scale flight test program be conducted to demonstrate the effectiveness of the TAMI system in a real life environment. In addition it is recommended that

(1) A brief experimental program, utilizing the existing model, be conducted to demonstrate the efficiencies of the TAMI system that are now believed possible at subsonic tip speeds.

(2) The effectiveness of the improved TAMI system be demonstrated with the existing model at high subsonic tip speeds.

(3) A theoretical study of rotor wake flows which includes the realistic wake effects as regards the acoustic effects of blade wake interactions be undertaken.

I. INTRODUCTION

A concentrated vortex trailed from a blade of a helicopter rotor interacting with another blade of the rotor can cause a sharp impulsive noise which has commonly been referred to as "blade slap". This impulsive noise signature is a great contributor to the acoustic annoyance of helicopters as it draws early attention to the presence of the helicopter. In military helicopter applications the "blade slap" type of impulsive noise provides early detection of the helicopter and thus greatly increases their vulnerability to ground fire.

Not all helicopter rotor configurations are noted for their impulsive noise signature due to blade-vortex interaction as the concentrated strength of the tip vortex and the blade-wake geometries are not of sufficient strength or are not properly oriented, particularly for multi-bladed, single-rotor systems. Notable configurations, as regards blade-vortex interaction noise, are the tandem rotor helicopters such as the CH-47 and the two-bladed rotor configurations such as the Huey UH-1 series. The impulsive acoustic footprints developed by these helicopters in descent and maneuvers which have been well documented in the past, show that the impulsive noise is rather severe (References 1 and 2).

While "blade slap" impulsive noise has generally been associated with main rotor blades, the interaction of the concentrated main rotor wake with the tail rotor also can cause a significant impulsive noise signature that can be detectable for many miles and may be an important noise source as is the main rotor impulsive noise.

Because of both the annoyance and detectability problems associated with blade-wake impulsive noise, considerable effort has been directed towards understanding the basics of the problem and in seeking ways to either alleviate or eliminate the problem by modifying the characteristics of the concentrated vortex trailed off of the tip of a rotor blade.

The attempts to modify the characteristics of the trailed tip vortex have used both passive and active approaches. Scheiman (Reference 3) investigated a number of passive and active approaches but did not find much of a beneficial effect of any of the approaches on the concentrated tip vortex. Passive approaches using a tip spoiler (Reference 4) or a porous tip (Reference 5) met with varying degrees of success as regards a beneficial modification of the vortex. The increase in power requirements to achieve these results by these techniques was prohibitive for helicopter rotors and have not been seriously considered since the initial investigations. A partial end plate has been tested and significant effects as regards vortex modifications were noted with only a small increase in power requirements (Reference 6). The use of an "Ogee tip" configuration to diffuse the concentrated energy in the tip vortex (References 7,8 and 9) seems to be successful and will soon be tested by NASA on a full-scale UH-1 rotor.

A disadvantage with any passive device is that if it draws power from the installed power plant it will degrade the performance characteristics of the aircraft in flight regimes in which blade-vortex impulsive noise is not a problem. For this and other reasons, various investigators have conducted research on active systems that utilize a directed stream of air into the vortex to modify its characteristics. The advantage of the active system is that, if it does draw power from the installed power plant, the system can be turned off for flight conditions in which blade-vortex impulsive noise is not a problem and therefore the performance characteristics of the aircraft will not be degraded for these flight conditions.

The primary active system that has been developed is that which injects an axial stream of air along the axis of the tip vortex as it leaves the rotor blade. This system has been developed and tested by the RASA Division of SRL for more than six years under the sponsorship of the Navy, Army and NASA. (References 8 and 10 through 15).

The research and development programs that have been conducted have been both theoretical and experimental investigations concerned with the basic aerodynamic characteristics of vortices forming at the tip of a lifting surface and of the viscous mixing of linear and swirling flows as well as the application of this basic knowledge to the development of a suitable injection system for the UH-1 series of helicopter blades (References 8 and 14). The results of investigations conducted in a wind tunnel with a full-scale section of the UH-1 rotor blade indicated that a significant reduction of the peak tangential velocity in the vortex core could be realized immediately behind the rotor blade with the use of a nozzle having a sonic velocity. As a result of the success that had been achieved in the wind tunnel, a proof of concept development program was conducted with NASA/Langley using a full-scale UH-1D rotor system on the Helicopter Rotor Test Facility (HRTF), (Reference 16). The results of these tests were inconclusive in that blade-vortex impulsive noise could not be obtained on the whirl tower and thus the beneficial effects of the vortex modification on the impulsive signature could not be obtained. Smoke visualization tests conducted with this rotor system indicated that a considerable change in the visualized tip vortex was obtained. While the effect of this vortex modification on the impulsive noise signature could not be determined experimentally, the noted changes in the vortex structure were utilized in the real time acoustic prediction program RASA/SRL developed for USAAMRDL (Reference 17) for a condition in which blade-vortex impulsive noise was present. The results of these calculations indicated that the noted modification of the vortex would reduce the impulsive noise by about 15dB over a reasonable range of frequencies which are significant as regards impulsive noise.

Another conclusion reached as a result of the whirl tower tests was, if the capabilities of the TAMI system in generating a beneficial effect as regards impulsive noise are to be evaluated, then tests must be conducted for rotor flight configurations in which blade-vortex impulsive noise is obtained. Since these flight conditions (descent and/or maneuvers) cannot be obtained on a whirl tower, either a flight or a wind tunnel test program must be conducted to adequately evaluate the system.

As a wind tunnel test program is generally less expensive to conduct than a flight test evaluation, this means of evaluating the TAMI system was utilized.

This report will describe the model and the tests that were conducted as well as presenting a brief summary of the results that were obtained.

II TECHNICAL DISCUSSION

A. DESCRIPTION OF MODEL & TEST EQUIPMENT

The model that was designed, constructed and tested was a two bladed 7 foot diameter teetering rotor driven at 1365 RPM by a remotely located variable frequency motor. Collective pitch control was obtained by a simple swashplate system whose position was changed by a remotely controlled linear actuator. Effective lateral cyclic pitch control was obtained by shaft tilt which was also controlled remotely.

Figure 1 presents a photograph of the rotor model in the University of Maryland 7.75 X 11 foot subsonic wind tunnel. The three microphones used to record the rotor noise can be seen in the photograph and Figure 2 shows a sketch of the relative locations of microphones with respect to the helicopter that was being simulated. Microphone 1 was located upstream below the rotor tip path plane, microphone 2 was located at the scaled fuselage door location and microphone 3 was located at the scaled location of the horizontal stabilizer.

The analog signals from the three microphones, the teetering potentimeter and a two per rev RPM blip were recorded on an FM tape recorder at 60 inches per second. On line spectrum analysis of the microphone signals was carried out and analyzed from an oscilloscope plot to evaluate the fidelity and characteristics of the acoustic signals in order to properly conduct a realistic test program in terms of not overlooking important and unexpected results. Off line hard copies of the spectrums and pressure time histories were developed from the taped signals.

B. DESCRIPTION OF TEST PROCEDURES

As discussed previously the purpose of the subject research program was to evaluate the effectiveness of the TAMI system in flight modes in which blade vortex interaction occurs. Typical of such a flight condition is descending flight at low advance ratio in which the rotor wake is traversing through the plane of the rotor. During this flight condition the noise output of the rotor is increased markedly due to these interactions. The documentation of this noise source is usually presented as a footprint which categorizes the noise as a function of advance ratio and descent rate such as shown in Figure 3. While these type of plots generally categorize the noise in terms of type and intensity they are generally generated by subjective testing and therefore the level of the noise on the various boundaries are not identified in terms of dB or other types of scientic measure. For the tests reported on herein such a footprint was partially defined in terms of dB levels for the basic rotor system by measuring the rotor noise in steady state level flight at various forward velocities and then at various forward velocities over a range of steady state trimmed descent rates. With these results obtained, similar tests were conducted to determine how these boundaries were altered by the TAMI system. While these tests could not evaluate the effects of compressibility on the impulsive noise, these types of effects are generally not of prime importance in this flight regime of a full scale helicopter. It was believed therefore that a conservative estimate of the benefits of the TAMI system in reducing blade-interaction noise at subsonic tip speeds could be obtained by testing the model in this flight regime.

In order to conduct the desired descent tests, means of properly developing the proper trim descent flight conditions in the wind tunnel had to be determined.

In steady state flight the effect of gravity has, other than determining the lift, no effect on the rotor forces. In descending flight however, there is a component of gravity that acts along the longitudinal axis of the helicopter which has significant effect on the aerodynamic forces that the rotor must develop to obtain aircraft trim. Figure 4 presents vector diagrams of the force developed by a helicopter in both steady state level and descending forward flight as well as the equations for the wind tunnel balance forces when the aircraft is in trim flight. For steady state level flight simulation, the thrust must equal the simulated weight and the drag balance readings should equal the drag of the fuselage that is being simulated at each forward velocity. In descending flight however, the drag of the fuselage will be reduced by the component of aircraft weight along the longitudinal axis of the helicopter. As the descent rate is increased the angle ϕ increases and the effect of the component of helicopter weight gets larger and cannot be ignored if proper descent trim is to be simulated in the wind tunnel. Utilizing these types of relationships and assuming an effective scaled fuselage cross sectional area, the drag balance readings for various rates of descent were determined prior to the tests. The actual drag readings used to trim the rotor in the wind tunnel were the calculated values corrected for the system drag forces obtained without the rotor attached.

C. DISCUSSION OF RESULTS

Figure 5 presents a spectrum of the background noise measured in the wind tunnel at a velocity of 70 ft/sec with the drive system operating but without the blade attached. The frequency scale is labeled with both the real frequencies (upper numbers) and those scaled to a full scale rotor system of the same type. The weak peaks in the spectrum were not specifically identified but are believed to be associated with various components of the drive system. In comparison with the spectrums that will be presented later for the rotor in various operating conditions, the background noise is at least 15dB less than the noise generated by the rotor over the entire frequency range and thus it is believed that the rotor noise characteristics that were measured during the tests are truly representative of the noise generated by a rotor during the interaction of the rotor blades with the rotor wake.

(1). Effect of Wake During Transition To Steady State Level Forward Flight

Figures 6a through 6e present the spectrums and pressure time histories for the simulated cabin microphone for advance ratios from 0.04 to 0.08. As the advance ratio is increased from 0.04 to 0.05 the pressure time history shows an increase in the small distinct spikes and the spectrum shows an increase in the acoustic energy in the frequency range of 500 to 3000Hz (note change in dB scale). While the humping of the noise spectrum is a characteristic normally associated with blade vortex interactions, a characteristic that is also normally associated with spectrums of rotor noise containing blade vortex interaction noise is the existence of very sharp and distinct peaks at harmonics of the blade passage frequency. ≈ 50 Hz is only slightly apparent in the spectrums presented in Figures 6a through 6e. It is believed that if the increase in noise is due to the induced effects of the "randomly unstable" rotor vortex wake and not due to the discrete blade-vortex interaction then the spectral characteristics shown in these figures would be obtained. It was concluded therefore that the increase in the noise level as the advance ratio was increased from 0.04 to 0.05 is due to the induced effects of the concentrated vortices in a turbulent wake and not due to discrete blade-vortex interactions.

As the advance ratio was increased to an advance ratio of 0.08 the acoustic energy in the spectrum decreased and the frequency range in which the energy was contained also decreased.

(2). Effect of Descent Rate On Noise Pressure Time Histories

The effect of descent rate at an advance ratio $\mu = 0.14$ on the pressure time histories at the cabin microphone are presented in Figures 7 and 8 for two different simulated gross weight conditions of the helicopter. The data shown in Figure 7 would correspond to a full scale helicopter gross weight of approximately 9000 lbs and that in Figure 8 for a gross weight of approximately 17,000 lbs. At a descent velocity of 7 feet per second for the low gross weight configuration (Figure 7) a single slap peak is noted at a one per rev spacing in the pressure time history. Using the blade passage markers at the bottom of the picture, it was determined that this impulse was generated when the interacting blade was at $\psi \simeq 82$ degrees. Since at this descent rate the impulsive noise, while relatively weak, was obtained on a consistant basis it was assumed that this descent rate corresponded to the boundary of "Continuous Slap" noted on the footprint shown in Figure 3. As the descent rate is increased the single spike increases in magnitude, and at a descent rate of 12 feet/sec, a second and a weak third spike begin to appear in the pressure time histories. The second and third spikes in the pressure time history occurred at $\psi \simeq 106$ and 120 degrees respectively. It is believed that the noise output for this type of signature is that which is classified as "Loud Slap" in Figure 3. At a descent rate of 12 feet/sec, believed to be at the center of the footprint, the three spikes are very predominate and as the descent rate is increased further the predominate spikes get smaller. For the higher gross weight condition, Figure 8, a very similar pressure time history pattern is obtained at the same descent rate except the magnitude of the predominate spikes relative to the rest of the noise is larger. It is of interest to note that the azimuthal position of the spikes are the same for both gross weight conditions which indicates that the wake position relative to the rotor plane is independent of the gross weight when the effect of the gravity is taken into account during descending forward flight.

Another feature of the pressure time histories for both gross weight conditions, is that as the blade-vortex interaction spikes are obtained there is obviously an increase in the noise level at all other azimuth locations that is not due to blade-vortex interaction. It is believed that this increased noise is due to all of the other concentrated vortices in the rotor wake that are close to the plane of the rotor.

(3) Effect Of TAMI System On Noise During Blade Vortex Interaction

(a) Analysis Based On Pressure Time Histories And Narrow Band Spectrums

Before discussing the specific results that were obtained during the present program, it is important to fully understand the desired goal of the evaluation that was undertaken. Figure 9 presents a typical envelope of a noise spectrum for a rotor when blade-vortex interactions are present. In the low frequency range the typical rapid drop off of the rotational noise with increasing frequency is depicted. For higher frequencies the increase in the dB level with frequency is due to the acoustic energy associated with blade vortex interaction. As the frequency content of the interaction noise becomes limited the dB level then decreases with increasing frequency. Because the rotational noise is not dependent upon the wake it would not be expected to be affected by a change in the wake vortex structure. Since the "spectrum humping" is related to the concentrated vortex energy in the wake interacting with the rotor, it would be expected that the greatest change in the noise spectrum resulting from the modification of the vortex structure by TAMI would be apparent in a change in the "spectrum humping". However, because the additional acoustic energy associated with rotor-wake interactions is not only due to the discrete blade vortex interaction but is also due to the general interaction of the entire wake with the rotor, the change in the acoustic energy associated with a significant reduction in the blade slap noise created by discrete blade vortex interactions would probably alter the spectrum only as shown in Figure 9. As the frequency range of the "spectrum humping" is that in the range of maximum ear sensitivity, a significant reduction in the dB level in this range of frequencies would be of great practical benefit even though the entire noise due to blade-wake interactions had not been eliminated. The evaluation of the success of the TAMI system, therefore was made on the basis of the decrease in the dB level in the area of "spectrum humping".

Figure 10 presents the spectrum of the noise measured at the simulated fuselage door location for the rotor at an advance ratio of $\mu = 0.14$ in steady state trim level flight. The pressure time history, shown in the insert, indicated that the primary pressure pulse is due to rotational effects which is also indicated by the rapid drop off of the rotational noise with increasing frequency as shown in this spectrum. In contrast, for a descent rate of 8 ft/sec and $\mu = 0.14$ as depicted in Figure 11a the pressure time history shows a significant interaction spike at the azimuthal angle of $\psi \approx 80$ degrees. The presence of this interaction spike is also apparent in the spectrum in the frequency range of 250 to 2000 Hz. The peak dB of the "spectrum humping" at a frequency of 1000 HZ for this descent rate is approximately 28 dB above the dB level occurring for a descent rate of zero. This spectrum also shows distinct spikes associated with the rotor blade passage frequency up to the 40th harmonic.

When the TAMI system was activated, the distinct interaction spike at $\psi \approx 80$ degrees was almost completely eliminated, Figure 11b, and the peak dB was decreased by 5 and the harmonic range by approximately 25%.

For a descent rate of 10 ft/sec, it can be seen in Figure 12a that the discrete blade-vortex interaction at $\psi \approx 80°$ is evident as well as the weaker ones at $\psi \approx 108$ and 120 degrees. It can also be seen that there is a great deal of acoustic energy present throughout the blade passage that is not associated with discrete blade-vortex interactions. As was seen in Figure 10, the non-discrete blade-wake interaction acoustic energy was nil at a zero descent rate in forward flight. It is concluded therefore that this acoustic energy is associated with the concentrated vortex energy of the rotor wake that is in the vicinity of the rotor plane but not being intersected by the blades.

The effect of the TAMI system on the acoustic output of the rotor during blade vortex interaction at a descent rate of 10 feet per second can be seen by comparing the spectrums presented in Figures 12b and 12c with that presented in Figure 12a. In comparing the spectrums presented in 12a and 12b it can be seen that the TAMI system, operating at a design pressure, not only reduces the peak dB in the "spectrum humping" region but also reduces the frequency range over which the "humping" occurs. Operating the TAMI system at 1.25 design pressure (Figure 12c) further reduces the peak dB and frequency range so the peak dB has been reduced by approximately 8 dB and the frequency range of significance by approximately 600 Hz. The results obtained at a descent rate of 12 feet per second at an advance ratio of $\mu = 0.14$, which are the flight parameters at the center of the footprint, are presented in Figures 13a through 13c and are similar to those obtained at a descent rate of 10 feet per second. While the peak dB was not reduced as much as at a descent rate of 10 feet per second it can be seen from the pressure time histories that a significant reduction in the impulsive peaks was obtained.

(b) Analysis Based on a dB(A) Weighted Approach

While the analysis of the narrow band spectrums and pressure time histories can provide much useful information, the results cannot be utilized directly to provide an evaluation of the subjective human response. In order to obtain an estimate as to what the results of a subjective evaluation of the TAMI system would be, the results obtained were compared on a dB(A) basis.

Figure 14 presents the spectrums, based on a dB(A) analysis, for the rotor (without TAMI) in various rates of descent at an advance ratio of $\mu = 0.14$. It can be seen from the results presented that the peak dB(A) value increases from 67 to a maximum of 98.5 as the descent rate increased from zero to 12 feet per second. It is also of interest to note that as the descent rate is increased from zero to 12 feet per second that the peak dB(A) occurs at a higher frequency (approximately 60 Hz per foot of descent rate). Figures 15 and 16 present the effect of TAMI, based on a dB(A) analysis, for three different descent rates for an advance ratio $\mu = 0.14$. The results for a descent rate of 8 feet/sec are presented in Figure 15. As previously noted the rotor noise for these flight conditions have been classified as "Continuous Slap". It can be seen from the results presented in this figure that not only did TAMI, operating at design pressure, significantly reduce the overall dB(A) but also shifted the dB(A) peak to a lower frequency.

Figure 16 presents similar data for descent rates of 10 and 12 feet per second. For a descent rate of 10 feet per second, previously designated subjectively as an area of "Loud Slap", the TAMI system operating at design pressure reduced the overall dB(A) by only 3 dB but at 1.25 design pressure a 7.5 dB drop in the overall dB(A) was obtained which subjectively reduced the noise level from "Loud Slap" to "Continuous Slap". In addition the frequency at which the peak dB(A) was obtained dropped from 300 to approximately 200 Hz (full scale frequency) corresponding to a "Continuous Slap" designation. At a descent rate of 12 feet per second the TAMI system operating at 1.25 design pressure again reduced the overall dB(A) as well as the frequency at which it occurred. On the basis of the previously noted subjective classification, TAMI reduced the noise at a descent rate of 12 feet per second, from "Maximum Slap" to "Loud Slap".

On the basis of the results that have been presented it is concluded, that on both a "scientific" and "subjective" basis, TAMI had a significant benefit on blade slap noise in that it not only reduced the overall dB(A) but also the frequency at which the peak dB(A) was obtained.

(4) Power Requirements Of The TAMI System

Measured power vs advance ratio curves for the rotor in steady state level flight and for a descent rate of 12 feet per second (descent rate for maximum noise) are presented in Figure 17. Also presented in this figure is the rotor power required by the TAMI system when it is operating during descent.

It can be seen that the power required for the present TAMI system is approximately 14% of the power required by the rotor in hover. Also noted, on this figure is the power required for a modified TAMI system which is approximately one half that of the present system. It is believed, based on the results of the present and previous tests, that the same beneficial effects of the TAMI system can be obtained with this reduced power requirement. The results obtained during the present tests that indicate that this reduced power requirement can be achieved are the following:

(a) The effectiveness of TAMI was dependent upon the spanwise location of the jet.

and (b) The blade tip vortices being intersected by a following blade are generated in a relatively small azimuthal range within the second quadrant of rotor azimuth.

During the present tests the model blade tips were constructed such that the injection nozzle could be moved spanwise $\pm$ 2% of the blade radius. On the basis of smoke tests using a strobe light with the rotor, in hover, it was observed that the nozzle was lined up with the core of the vortex and was thus in an optimum location. While similar tests could not be conducted as effectively in forward flight, the observed results indicated that in the second rotor quadrant the tip vortex left the blade section at the same radial location but at an angle with respect to the trailing edge.

Since nozzle angularity with the trailing edge could not be accomplished, the nozzle was moved inward so that the jet would more strongly interact with trailing vortex. While this was obviously not the optimum location, it did improve significantly the reduction in the noise output of the rotor during blade vortex interaction with TAMI operating.

The flow picture shown at the top of Figure 18 illustrates the point that was being discussed. While this flow picture was taken with a fixed lifting surface swept forward approximately 15 degrees, it is believed that the rotor blade would experience, to various degrees, the same type of swept flow characteristics in the second rotor quadrant during forward flight. On the basis of the path of the vortex core shown in this photograph it can be seen that if the TAMI system was directed perpendicular to the trailing edge at the centerline of the vortex core on the lifting surface, the nozzle flow would be directed away from the vortex centerline. If the nozzle were moved inward, the directed nozzle flow would more strongly interact with the vortex and cause a larger modification of the concentrated vortex energy as was indicated during the rotor tests.

The data plot shown in the center of Figure 18 is data taken from Reference 14 in which the effect of the angle of injection with respect to the vortex core was investigated. The model that was tested during this program was a full scale UH-1D blade section in a fixed reference frame with the nozzle angle fixed at 9.5 degrees with respect to the chordplane. When the blade section was at an angle of attack of 9.5 degrees the trailed vortex path was lined up with the centerline of the vortex core. When the angle of attack of the wing section was reduced by 4 degrees so that the path of the injection system was traversing the path of the vortex as it left the wing, the maximum swirl velocity of the modified vortex was approximately twice that of modified vortex with the nozzle in the optimum location. This change is similar to the effect one might expect by moving the nozzle inward on the rotor blade. When the angle of attack of the wing was increased by 4 degrees so that the centerline of the injection system was at an angle less than the path of the vortex, the maximum swirl velocity of the modified vortex was approximately three times that of the maximum swirl velocity of the modified vortex with the nozzle in the optimum location. While sufficient backup data was not obtained to determine what the change in the power would be if the optimum injection could be realized, it is estimated that to achieve the same benefit that was achieved during the present tests, only 70% of the power used in the present test would be required.

An additional aspect of potential power savings is that associated with scale effects. The data plot presented at the bottom of Figure 18 was obtained during a research program conducted for the Office of Naval Research (Reference 12) in which possible scale effects of the TAMI system were being investigated. The results indicate that for a given nondimensional thrust parameter the modification of the vortex generated by a full scale rotor blade section was significantly greater than that obtained with a 1/4 scale model. Unfortunately the reason for this noted scaling effect was not determined although it was obtained consistently.

On the basis of the data presented in Figure 18 it has been concluded that if the nozzle can be lined up properly with the tip vortex generated in the second rotor quadrant that the same benefits achieved during the present program could be achieved on a full scale rotor with approximately one half of the power that was utilized during the present tests.

III CONCLUDING REMARKS AND RECOMMENDATION

It is believed that the research investigation that was conducted provided a realistic assessment of the potential of the TAMI system in reducing the noise output during blade-vortex interaction in descending low speed flight. In general it was concluded that the noise output due to blade-vortex interaction could be reduced by 4 to 6 dB with an equivalent power expenditure of approximately 14% of installed power. On the basis of the analysis of the results it was also concluded that the same benefits could probably be obtained with approximately one half of the present power by altering the nozzle location and angularity to more closely align it with the axis of the vortex core in the azimuthal range in which the critical vortices are generated. Specifically the following conclusions were drawn:

(1) The increased noise level in transition during steady state level flight is believed due to general vortex induced wake turbulence and not due to discrete blade-vortex interactions.

(2) The vortices that are intersected by a following blade to produce blade slap are generated in the second quadrant within a $\Delta\psi$ range of approximately 30 degrees.

(3) The blade vortex intersection occur at the same azimuthal locations on the advancing side regardless of advance ratio or C_T.

and (4) Based on the results obtained it is believed that the effectiveness of the TAMI system in alleviating blade slap noise can be increased and that the power requirements can be cut in half by proper design.

On the basis of the results obtained and the success achieved during the present program the following recommendations are made:

(1) Conduct a brief experimental program to demonstrate the improved efficiencies of the TAMI system that are now believed possible.

(2) Using the present TAMI rotor model evaluate the effectiveness of the TAMI system at high subsonic speeds.

(3) Conduct a full scale flight test program to demonstrate the effectiveness of the TAMI system in a real life operating environment.

(4) Undertake a theoretical study of rotor wake flows which includes the realistic wake effects as regards the acoustic effects of blade-vortex interactions.

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TABLE I

CONVERSION OF BRITISH ENGINEERING UNITS TO SI UNITS

to convert	to	multiply by
feet/sec	meter/sec	3.28
inches/sec	meter/sec	39.37
feet	meter	3.28
pounds	kilograms	2.20
horsepower	kilowatts	1.34

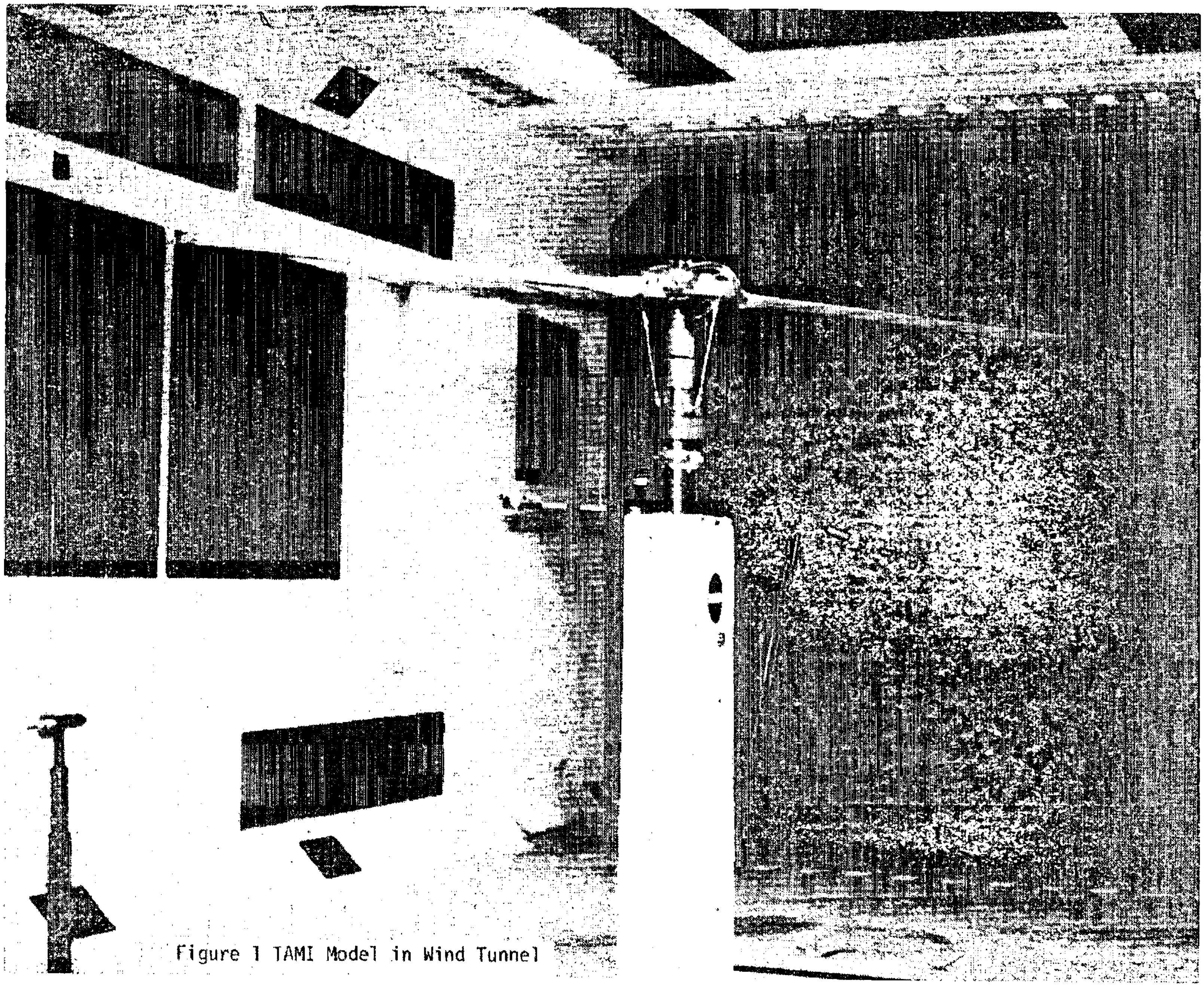


Figure 1 TAMI Model in Wind Tunnel

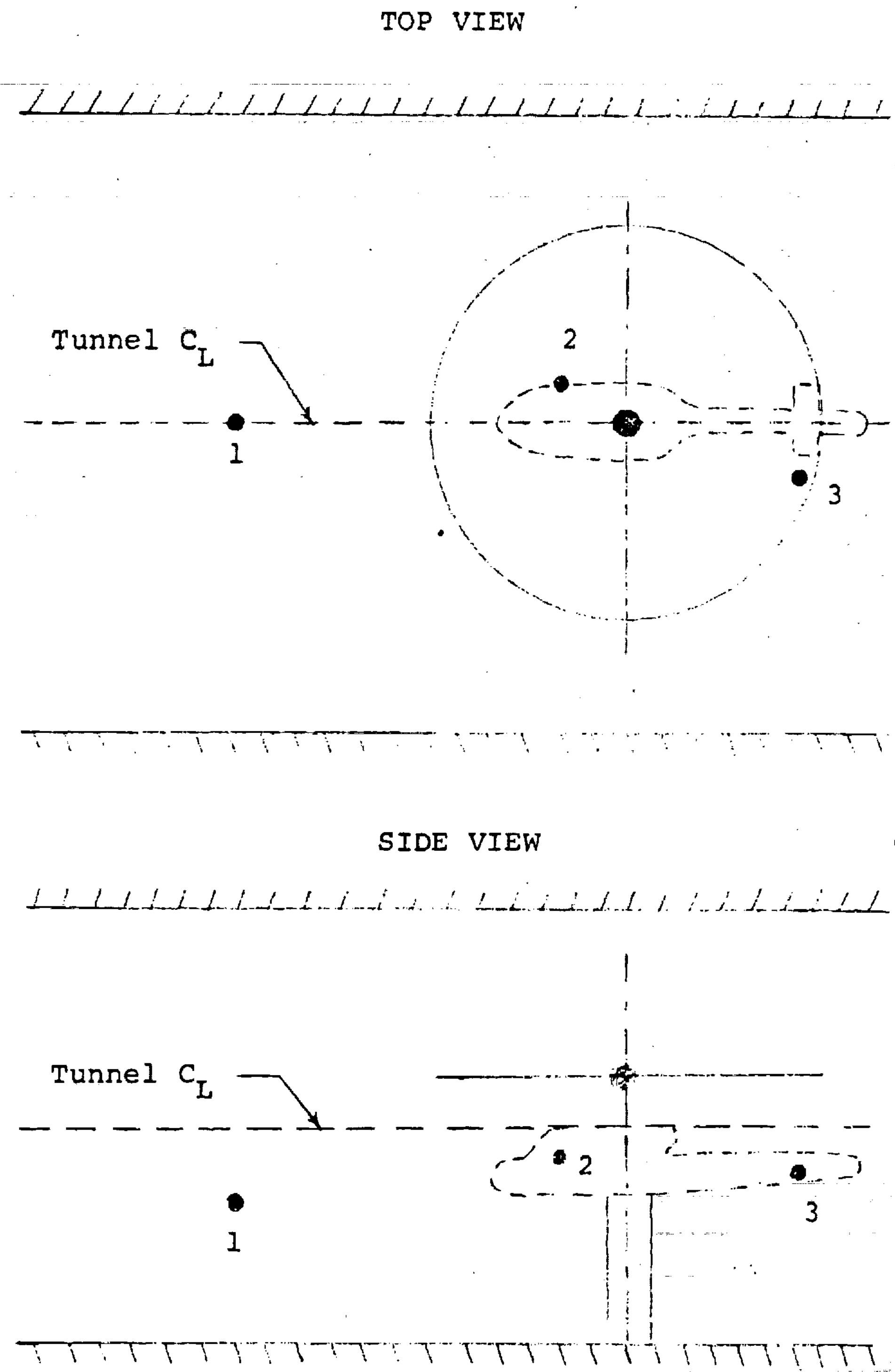


Figure 2 Microphone Locations (Note: Mikes 2 and 3 move with rotor support structure)

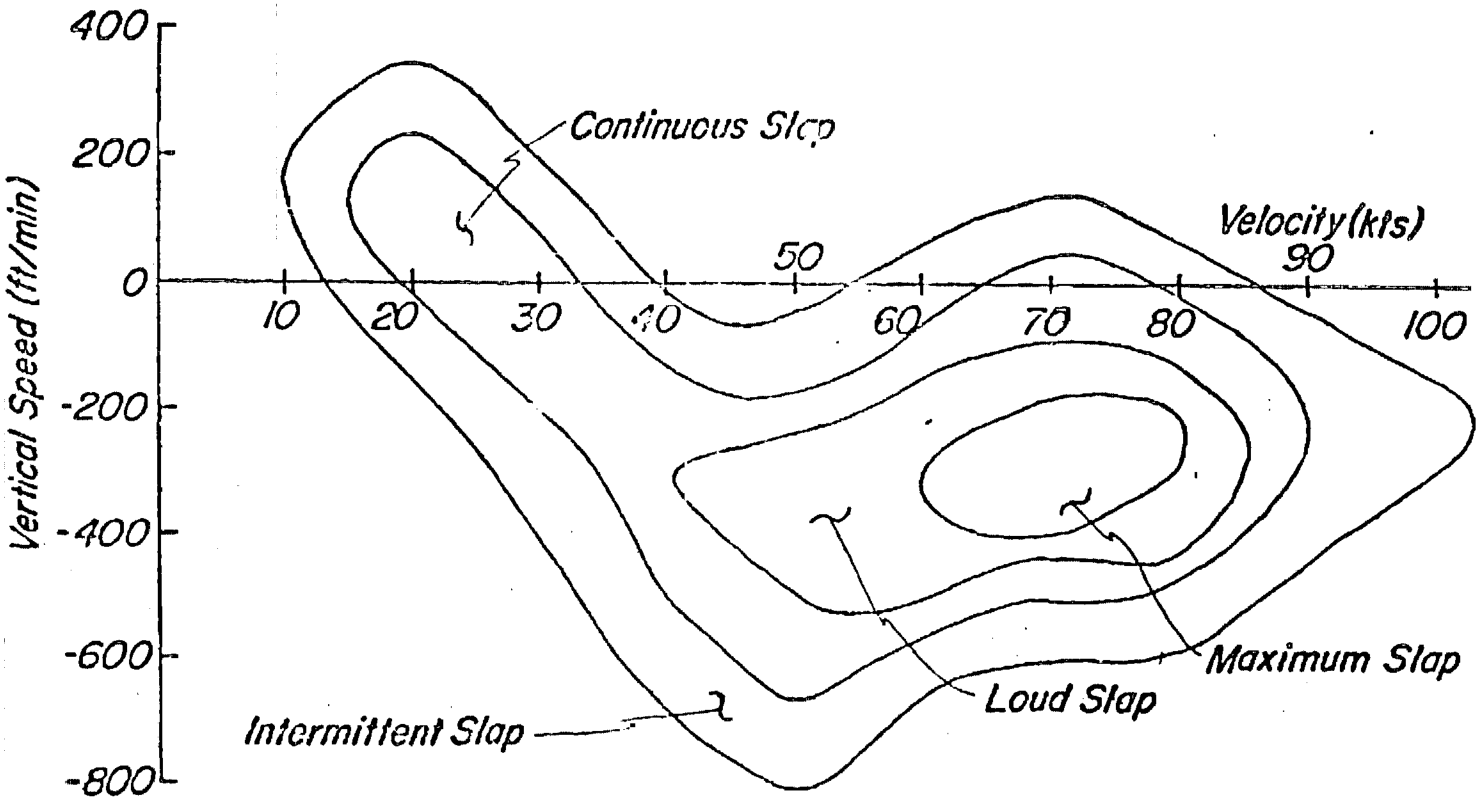
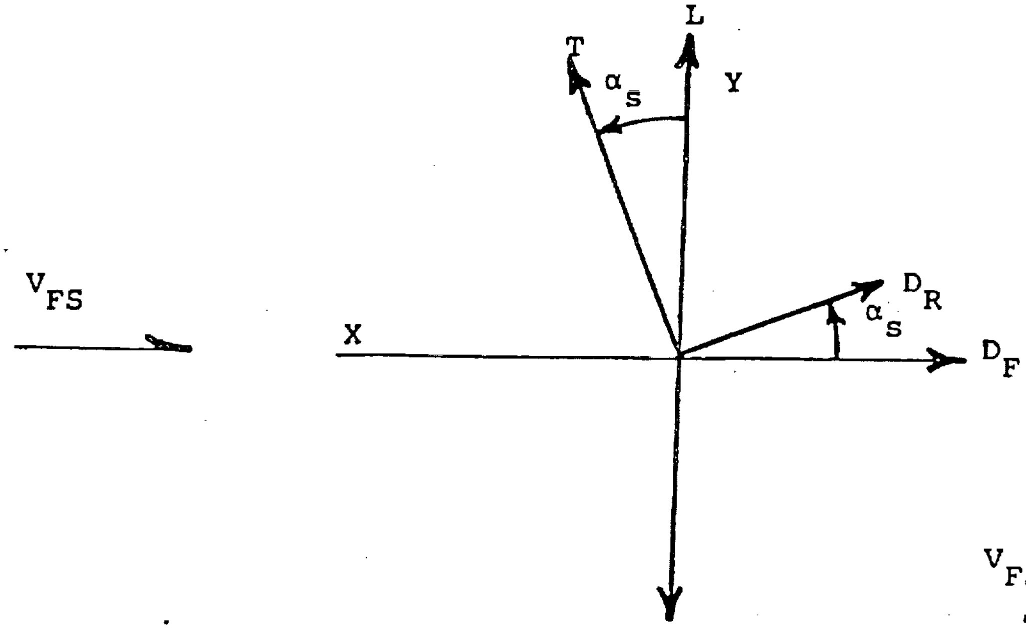


Figure 3 Full Scale Flight Conditions Associated with Main Rotor Blade Slap

LEVEL FLIGHT



V_{FS} = Freestream Velocity
T = Thrust
D_R = Rotor Drag
D_F = Fuselage Drage
W_A = Aircraft Weight
α_s = Shaft Tilt Angle
L = Lift

For Force Balance:

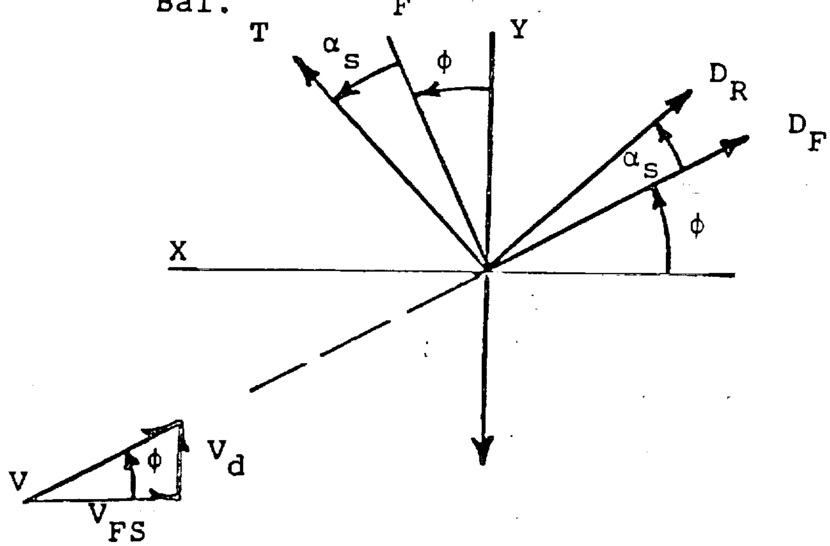
$$T \cos \alpha_s + D_R \sin \alpha_s = W_A$$

$$T \sin \alpha_s - D_R \cos \alpha_s - D_F = 0$$

$$T \cos \alpha_s = W_A - D_R \sin \alpha_s$$

$$L_{Bal.} = W_A - D_R \sin \alpha_s \simeq W_A$$

$$D_{Bal.} = - D_F$$



DESCENDING FLIGHT

$$L_{Bal.} = W_A \cos \phi - D_R \sin \alpha_s \simeq W_A$$

$$D_{Bal.} = - D_F + W_A \sin \phi$$

FIGURE 4. Balance Force Equations

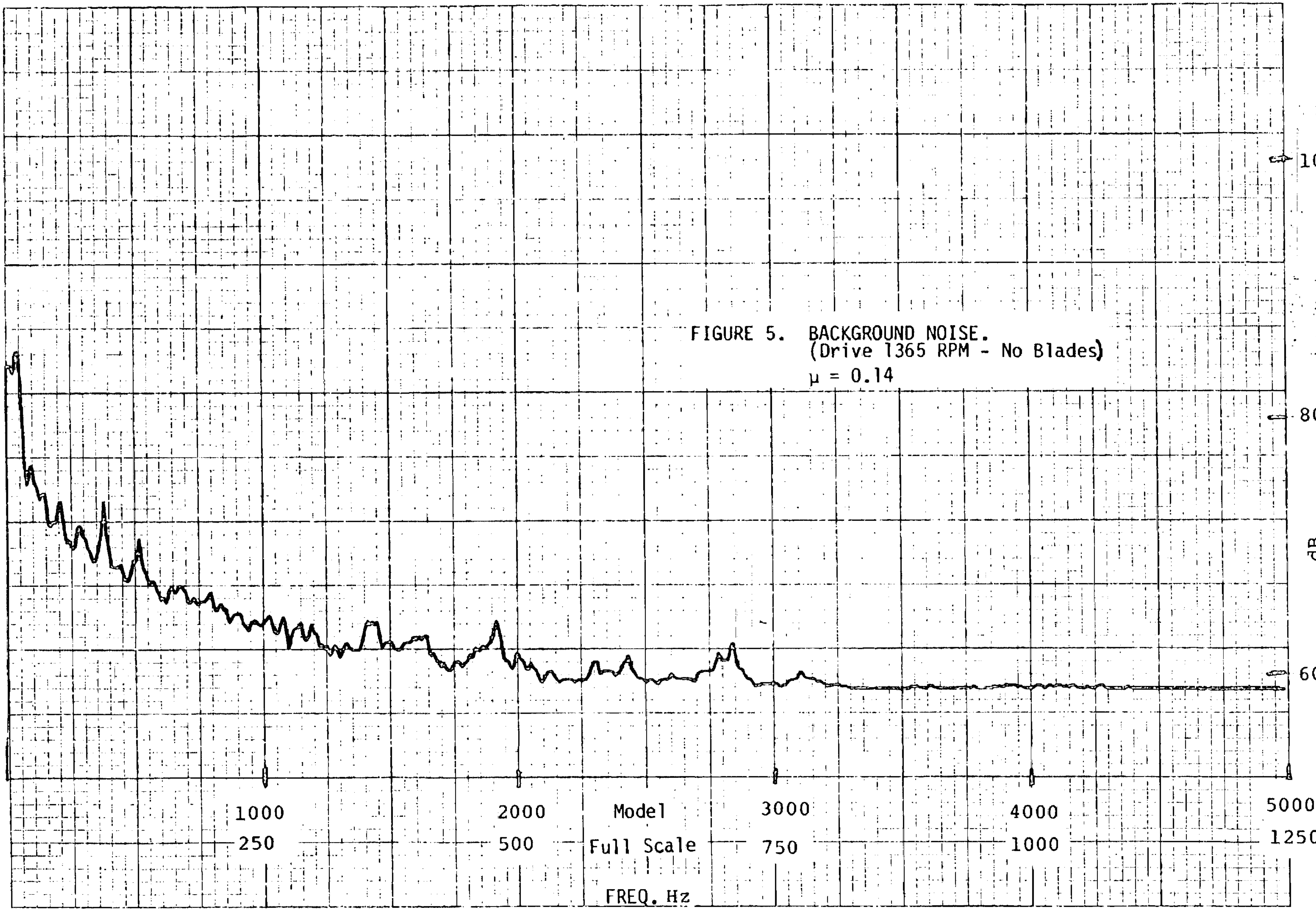


FIGURE 5. BACKGROUND NOISE.
(Drive 1365 RPM - No Blades)
μ = 0.14
100
80
60
dB
1000
2000
3000
4000
5000
Model
250
500
750
1000
1250
Full Scale
FREQ. Hz

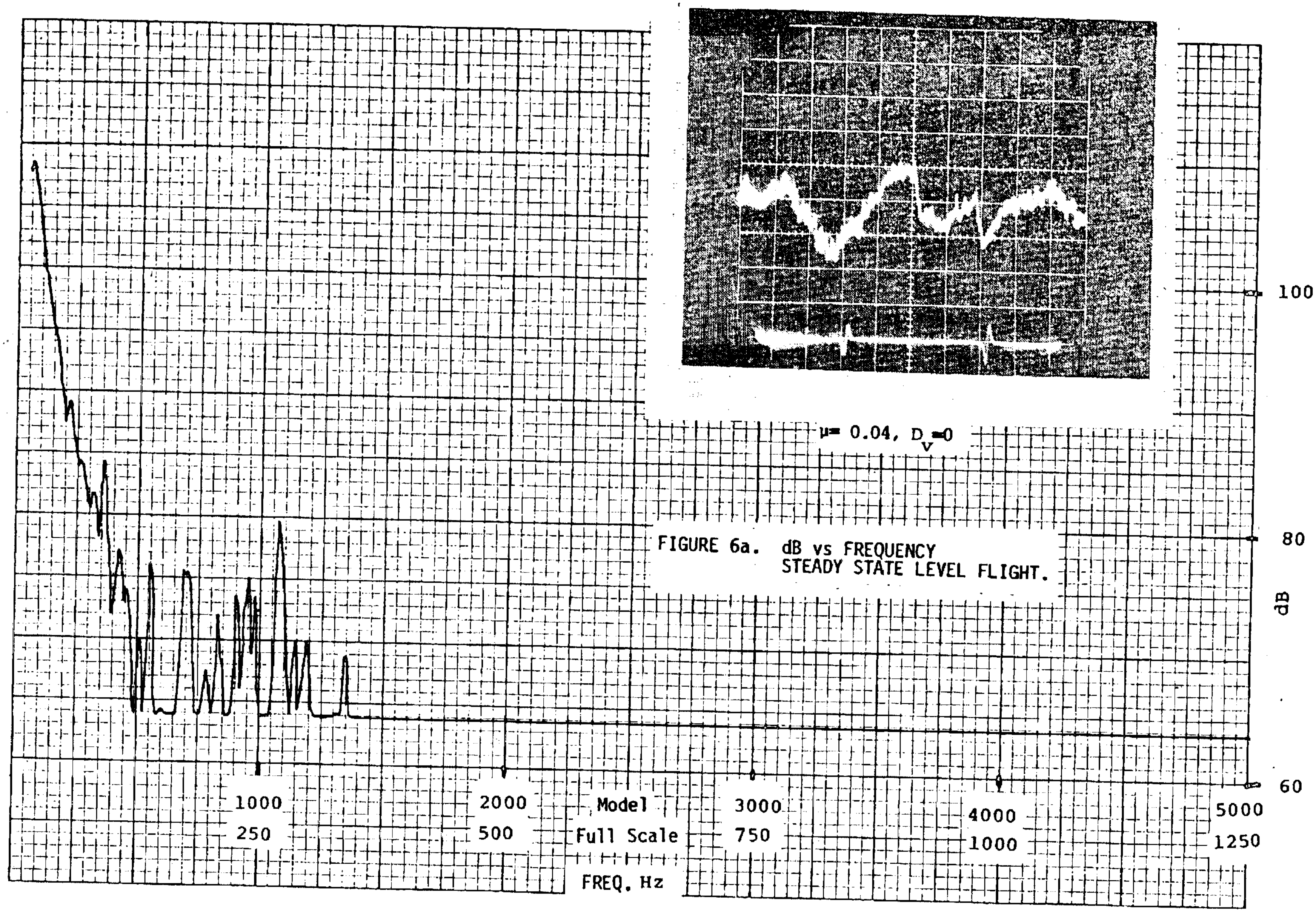


FIGURE 6a. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT.

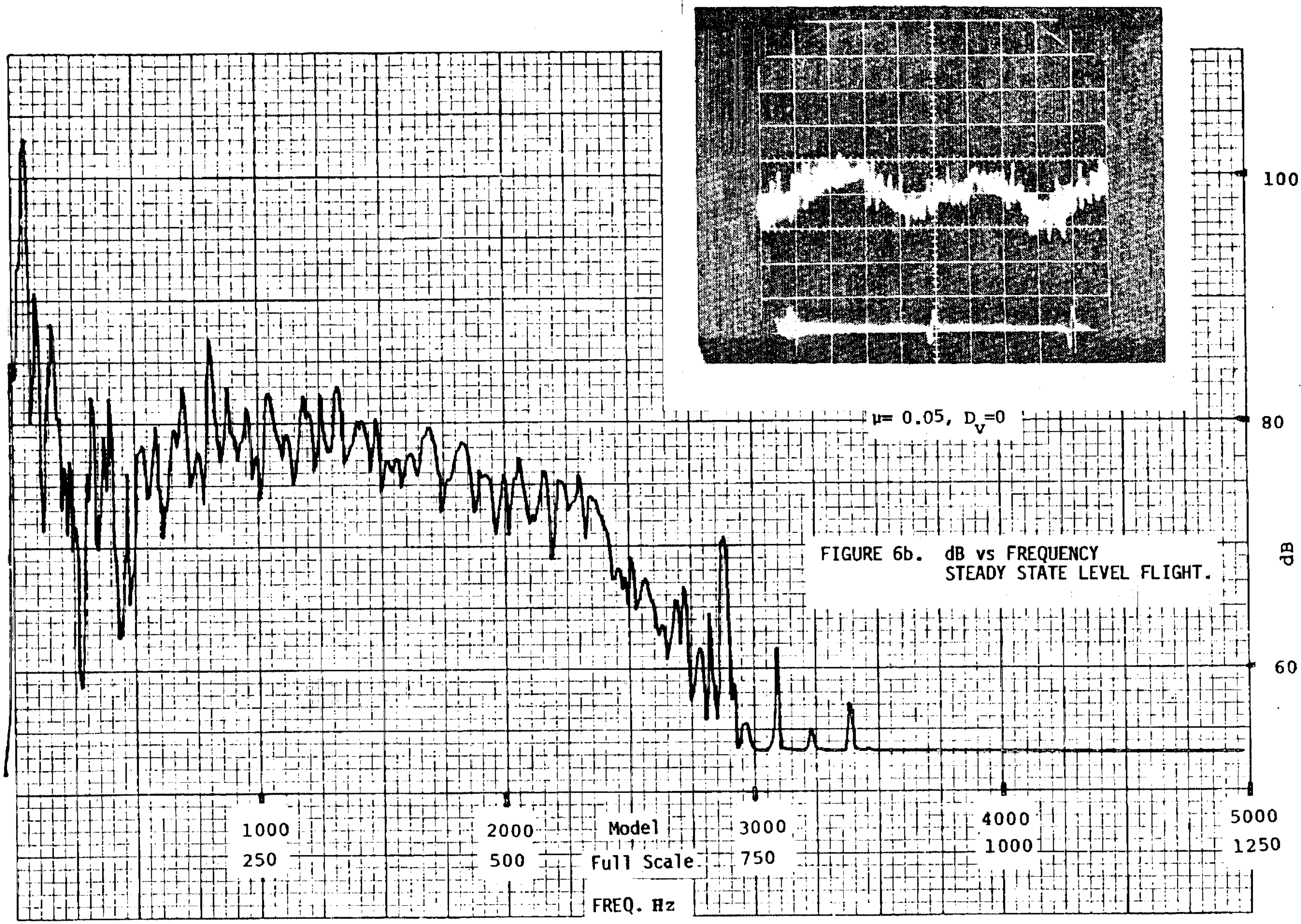


FIGURE 6b. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT.

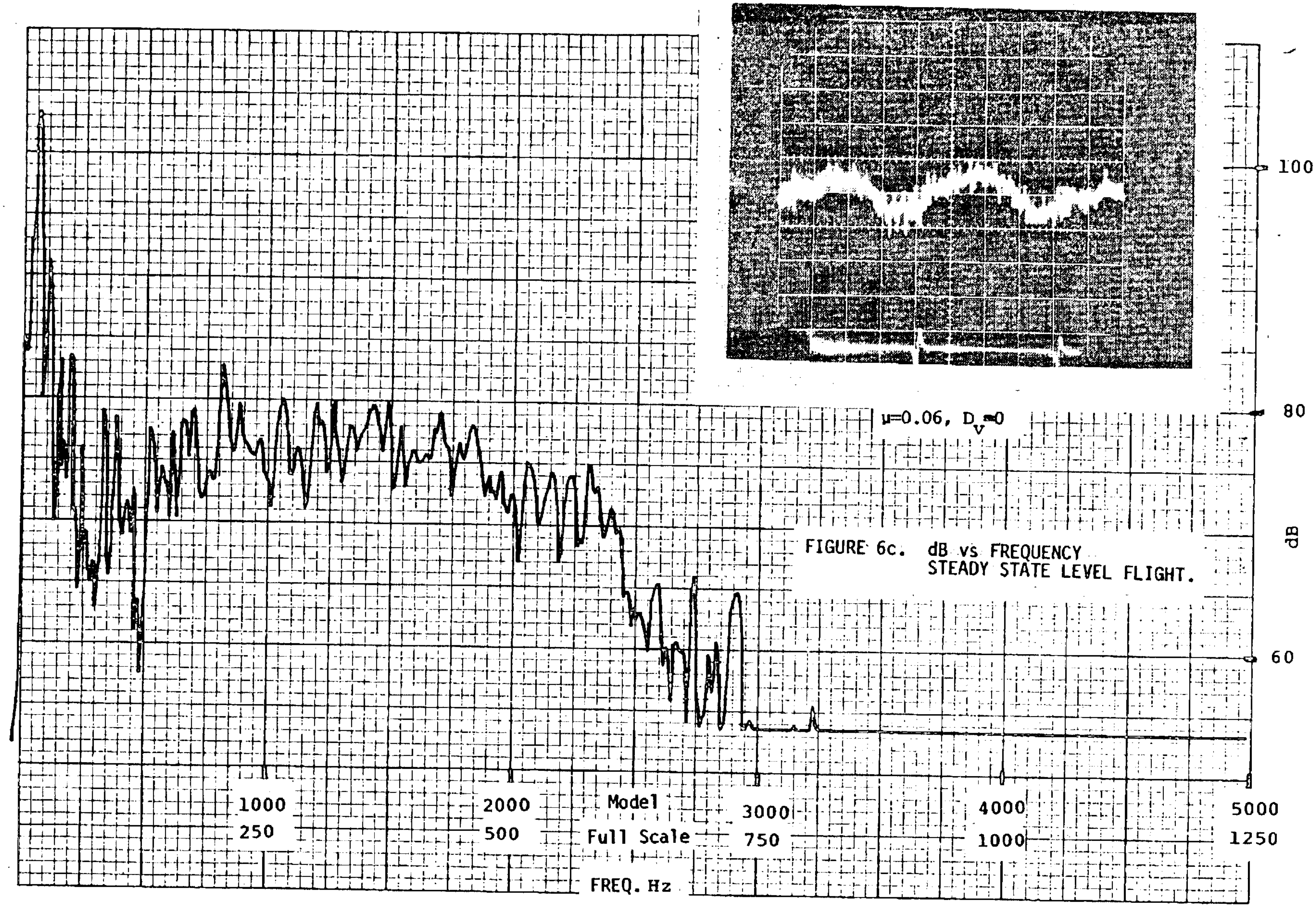


FIGURE 6c. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT.

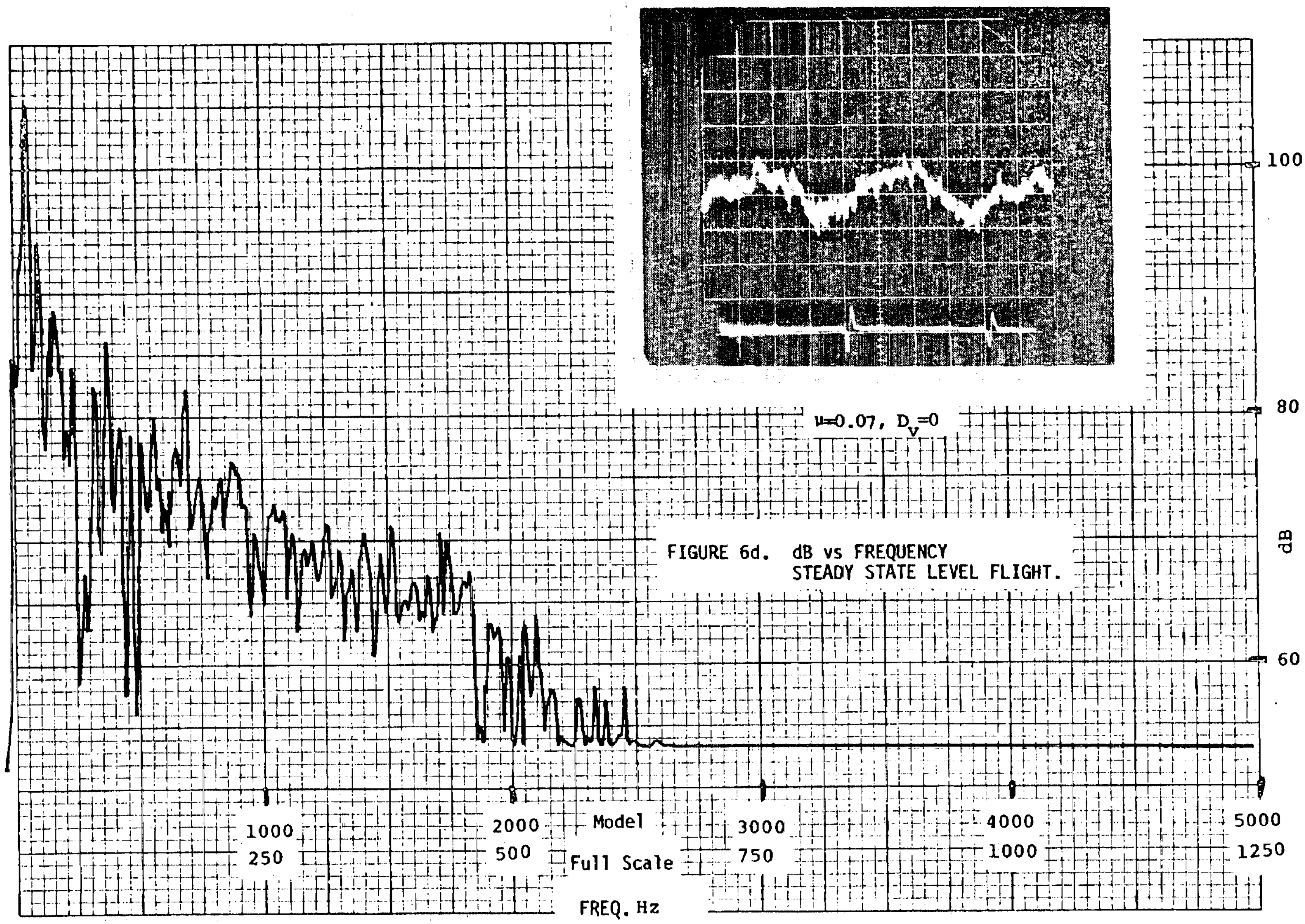


FIGURE 6d. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT.

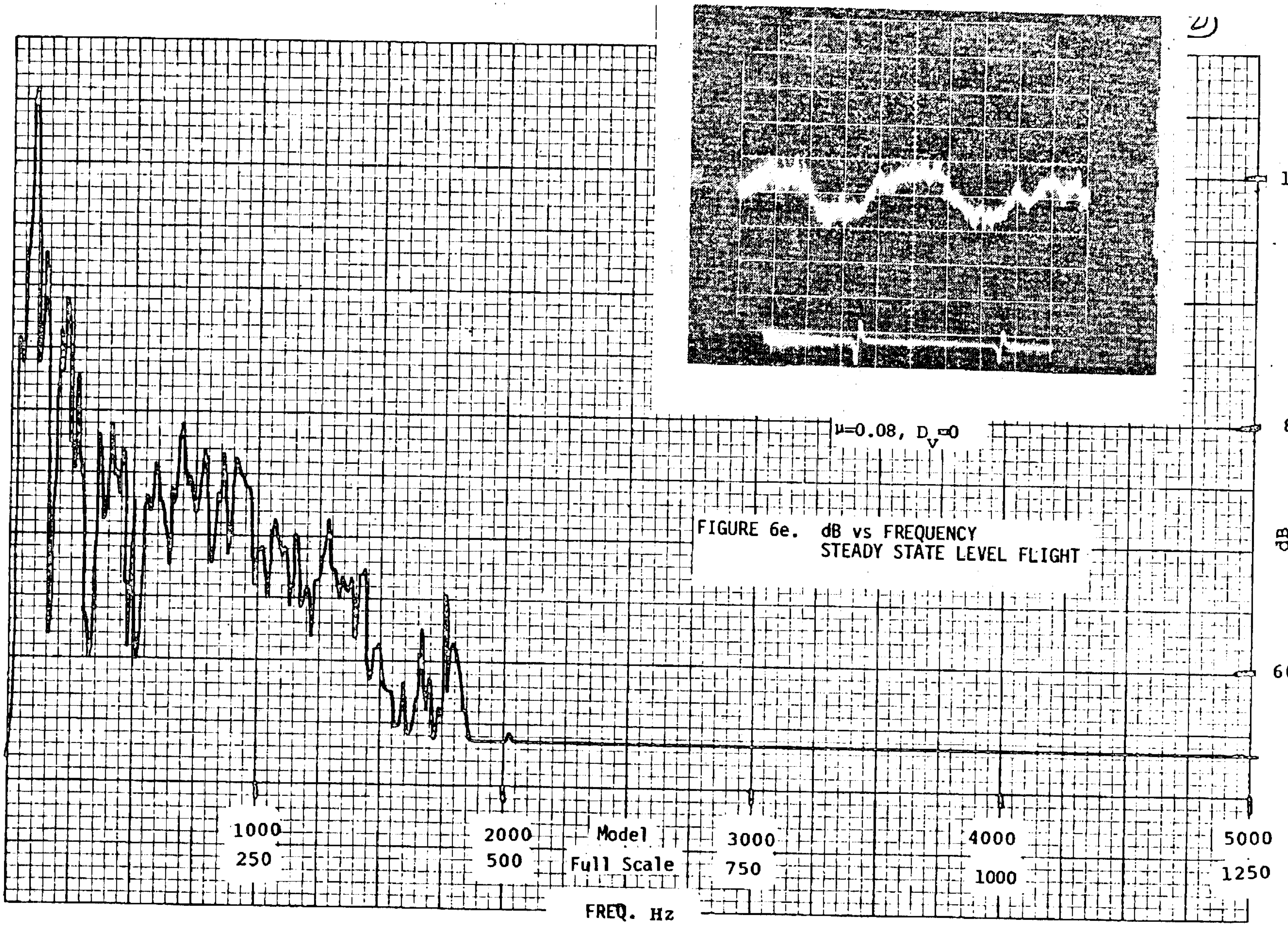
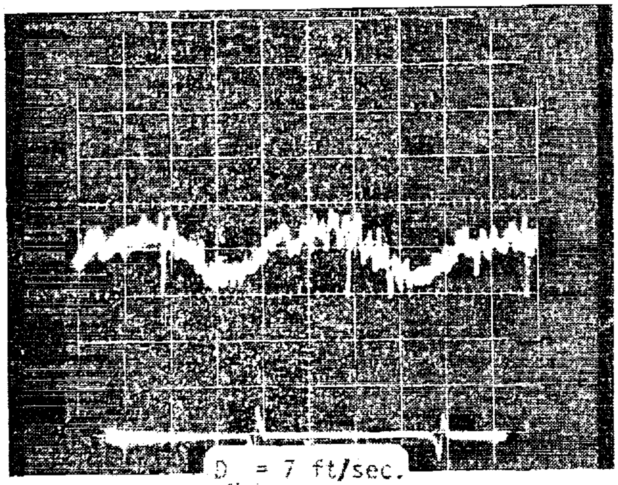
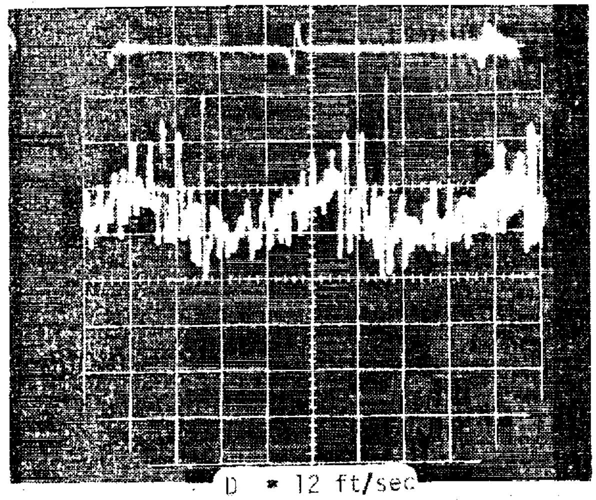


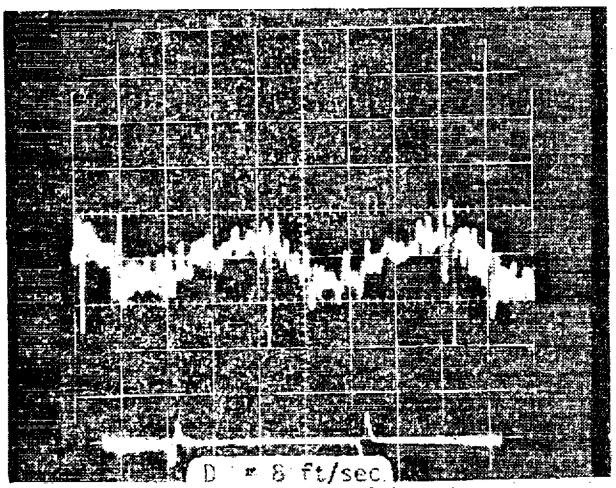
FIGURE 6e. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT



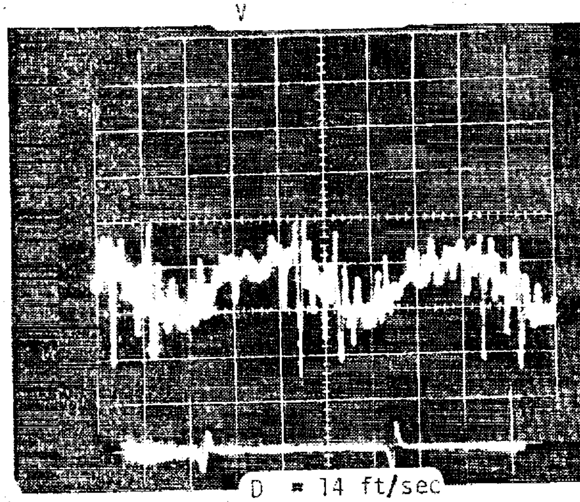
D_V = 7 ft/sec.



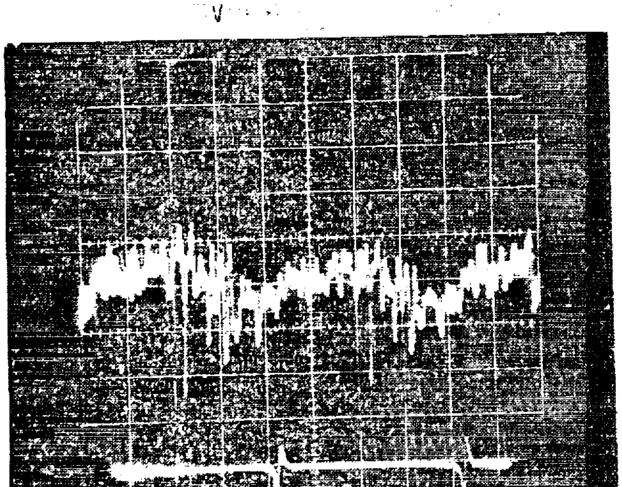
D_V = 12 ft/sec



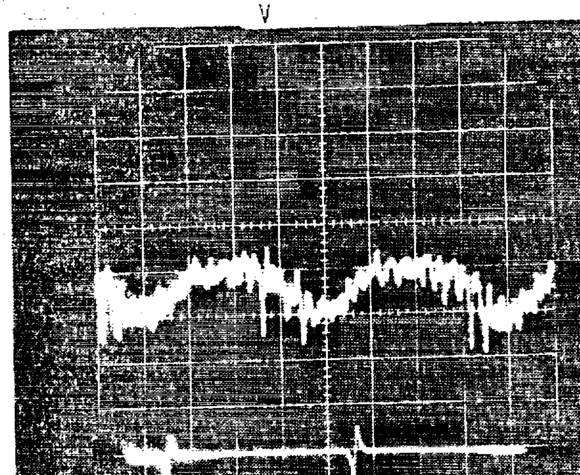
D_V = 8 ft/sec.



D_V = 14 ft/sec

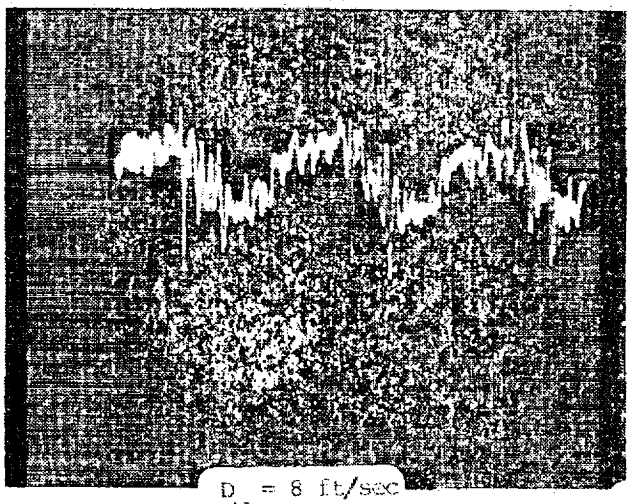


D_V = 10 ft/sec

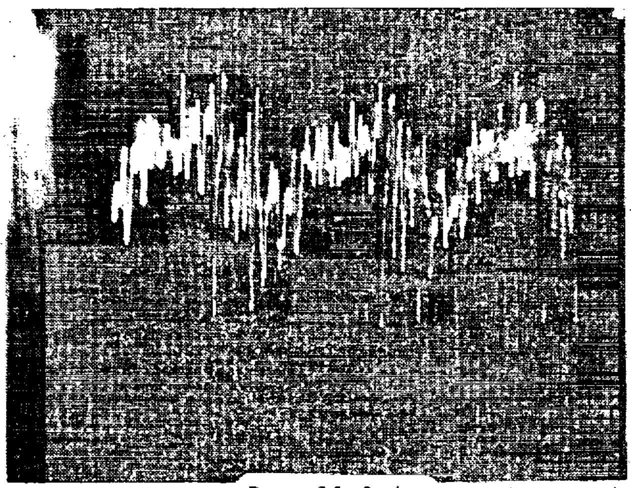


D_V = 16 ft/sec

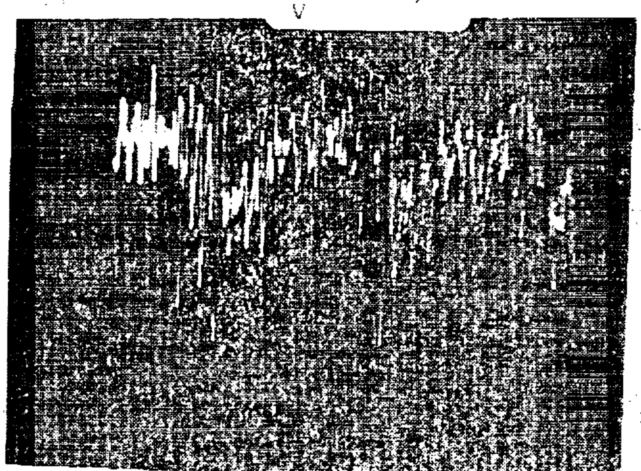
FIGURE 7. PRESSURE TIME HISTORIES vs.
DESCENT RATE
Low Gross Weight μ = 0.14



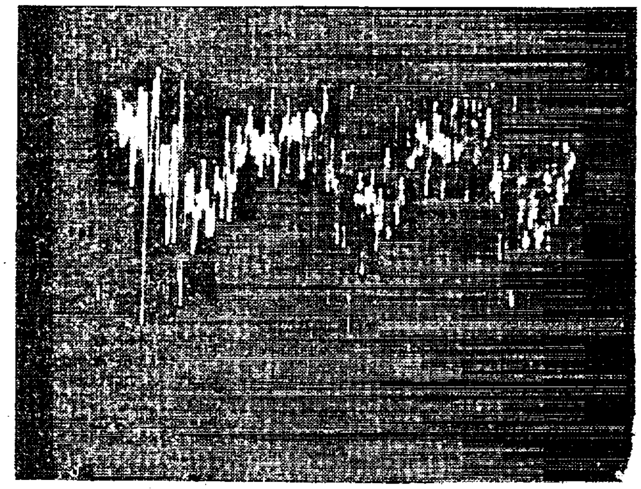
D_V = 8 ft/sec



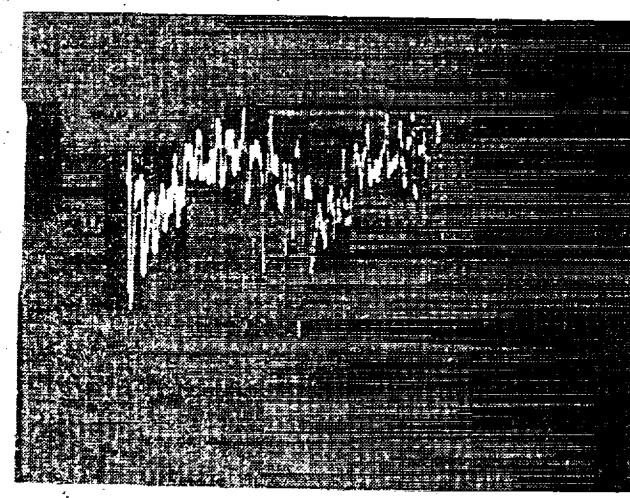
D_V = 10 ft/sec



D_V = 12 ft/sec



D_V = 14ft/sec



D_V = 16 ft/sec

FIGURE 8. PRESSURE TIME HISTORIES vs.
DESCENT RATES
HIGH GROSS WEIGHT μ = 0.14

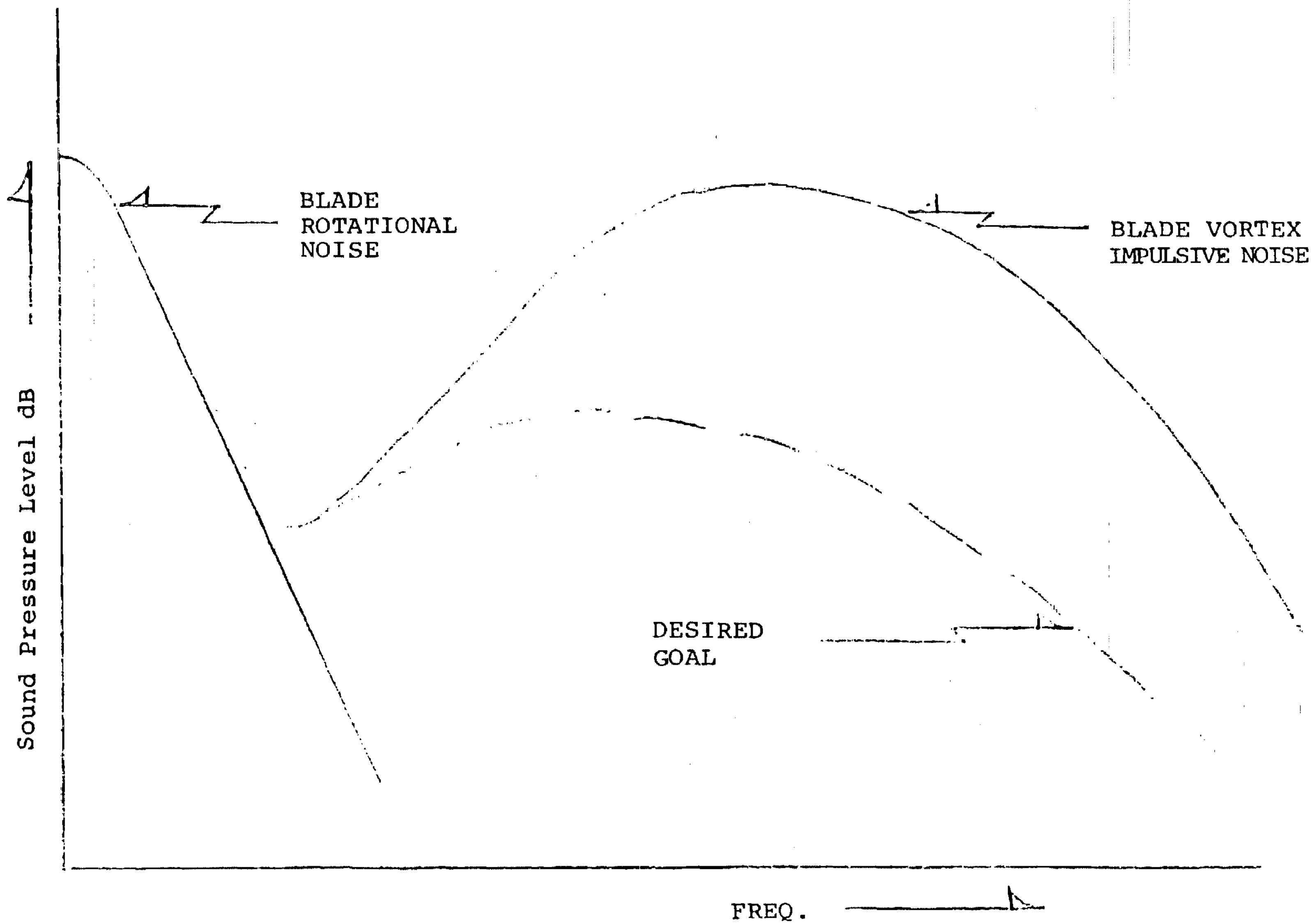


FIGURE 9. IDEALIZED SPECTRUM

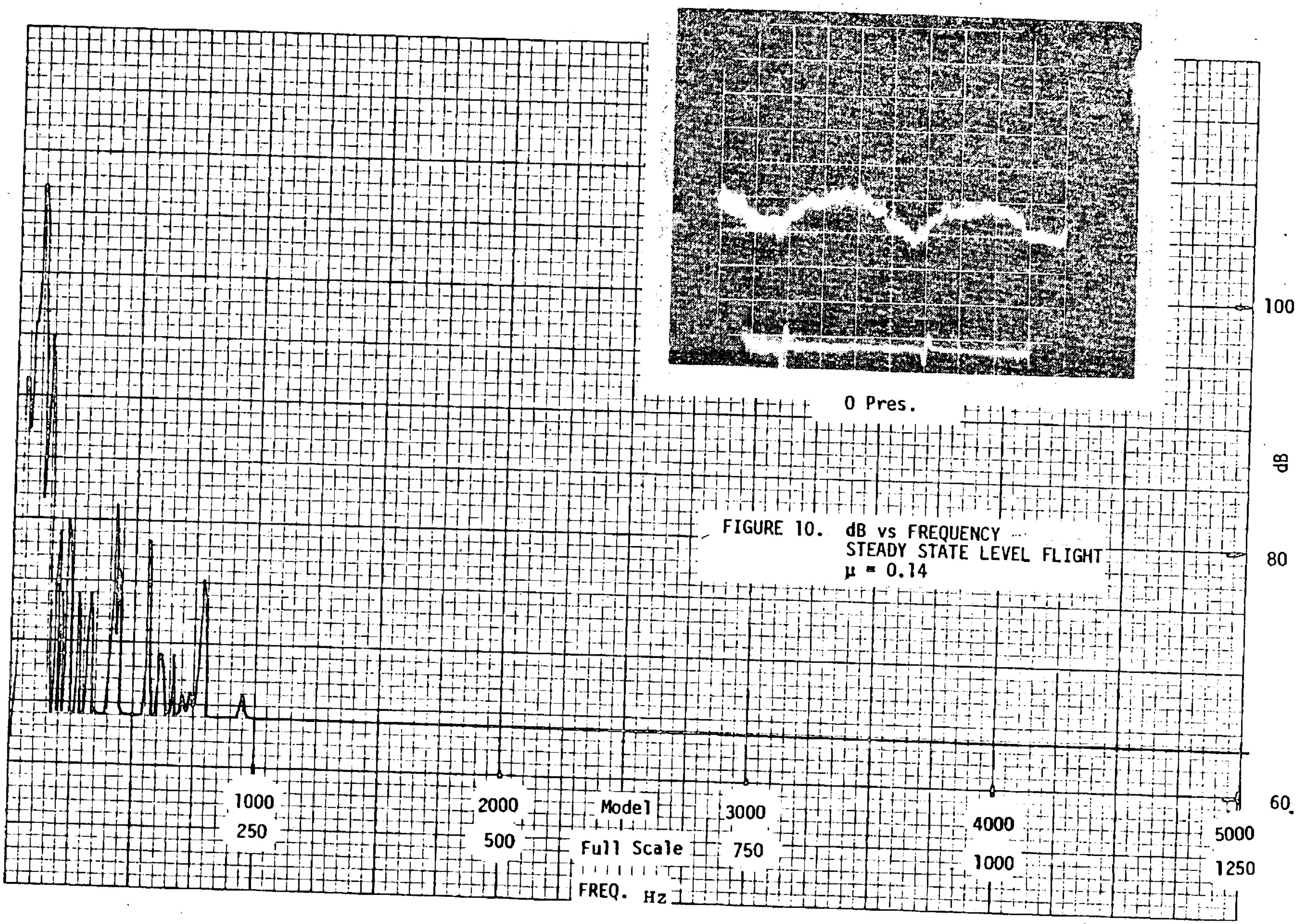


FIGURE 10. dB vs FREQUENCY
STEADY STATE LEVEL FLIGHT
$\mu = 0.14$

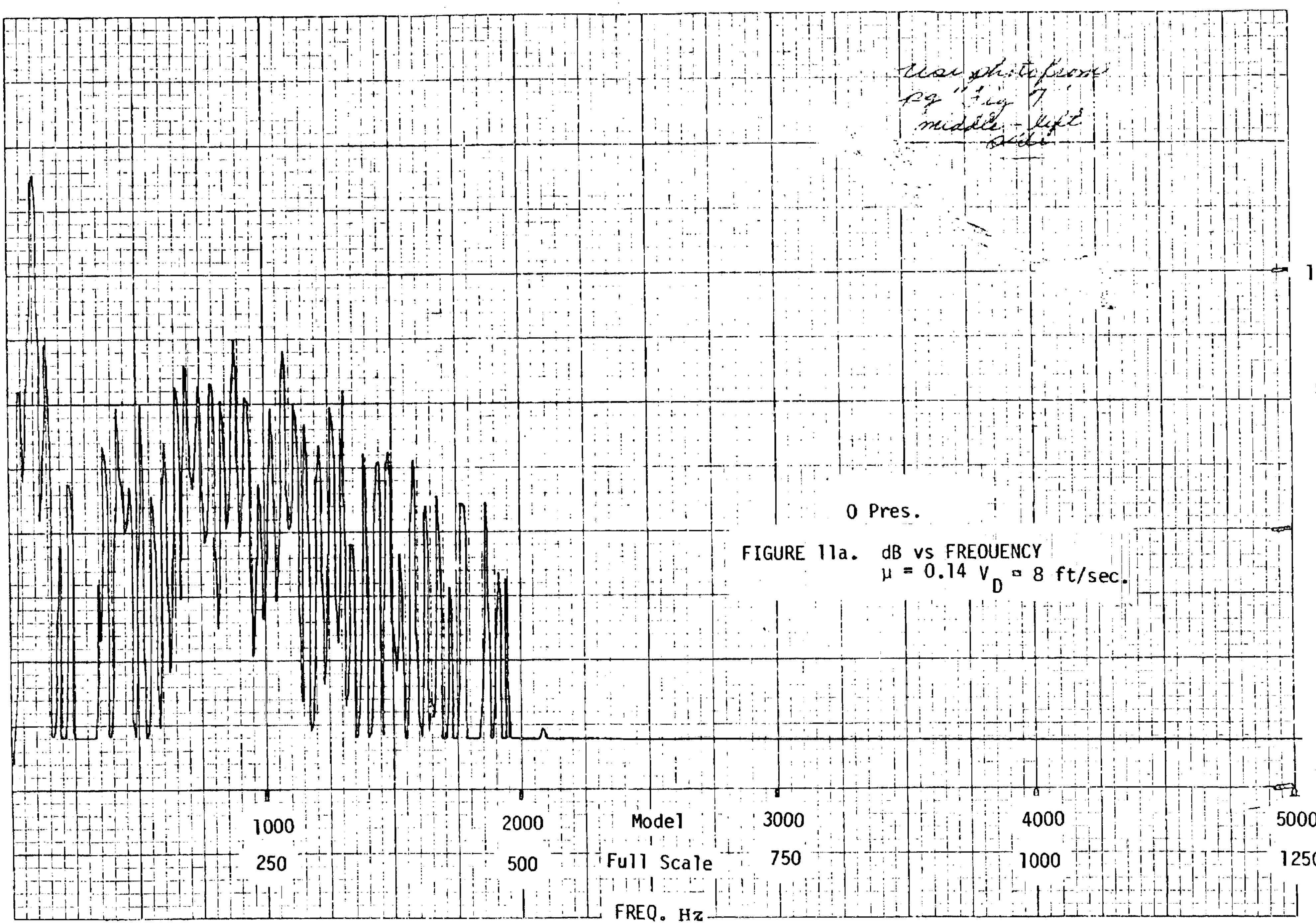


FIGURE 11a. dB vs FREQUENCY
$\mu = 0.14$ $V_D = 8$ ft/sec.

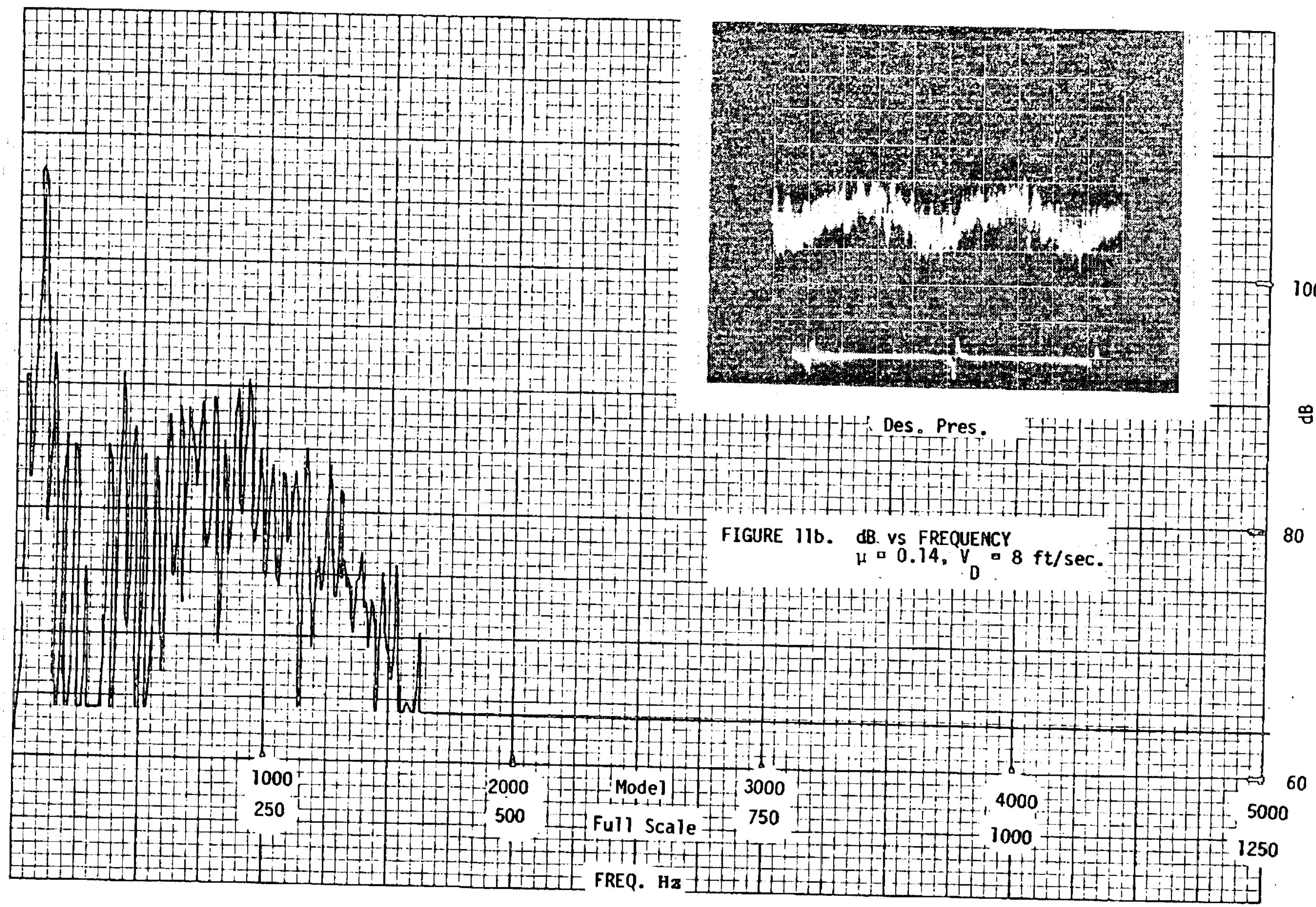


FIGURE 11b. dB vs FREQUENCY
$\mu = 0.14$, $V_D = 8$ ft/sec.

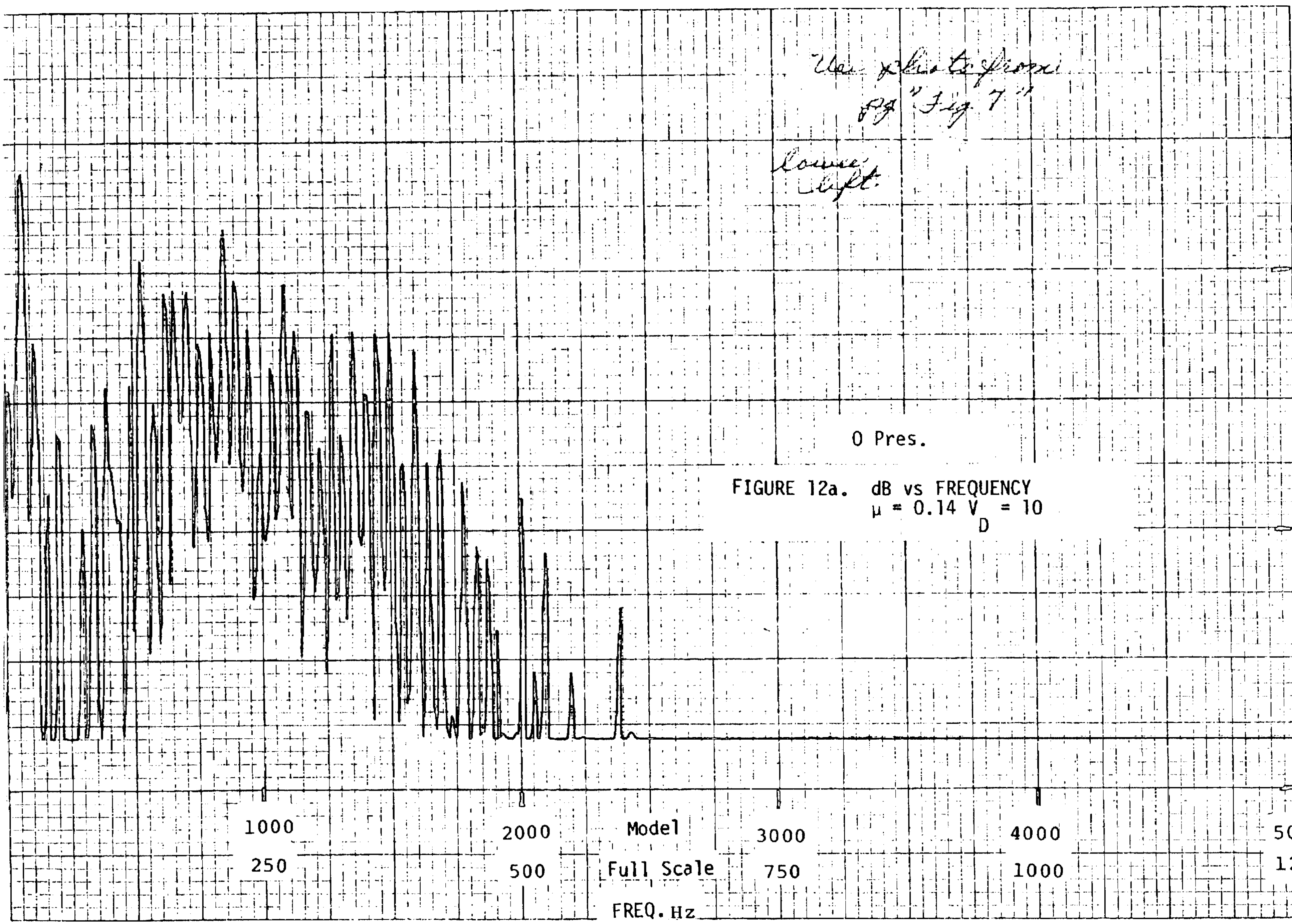


FIGURE 12a. dB vs FREQUENCY
$\mu = 0.14$ $V_D = 10$

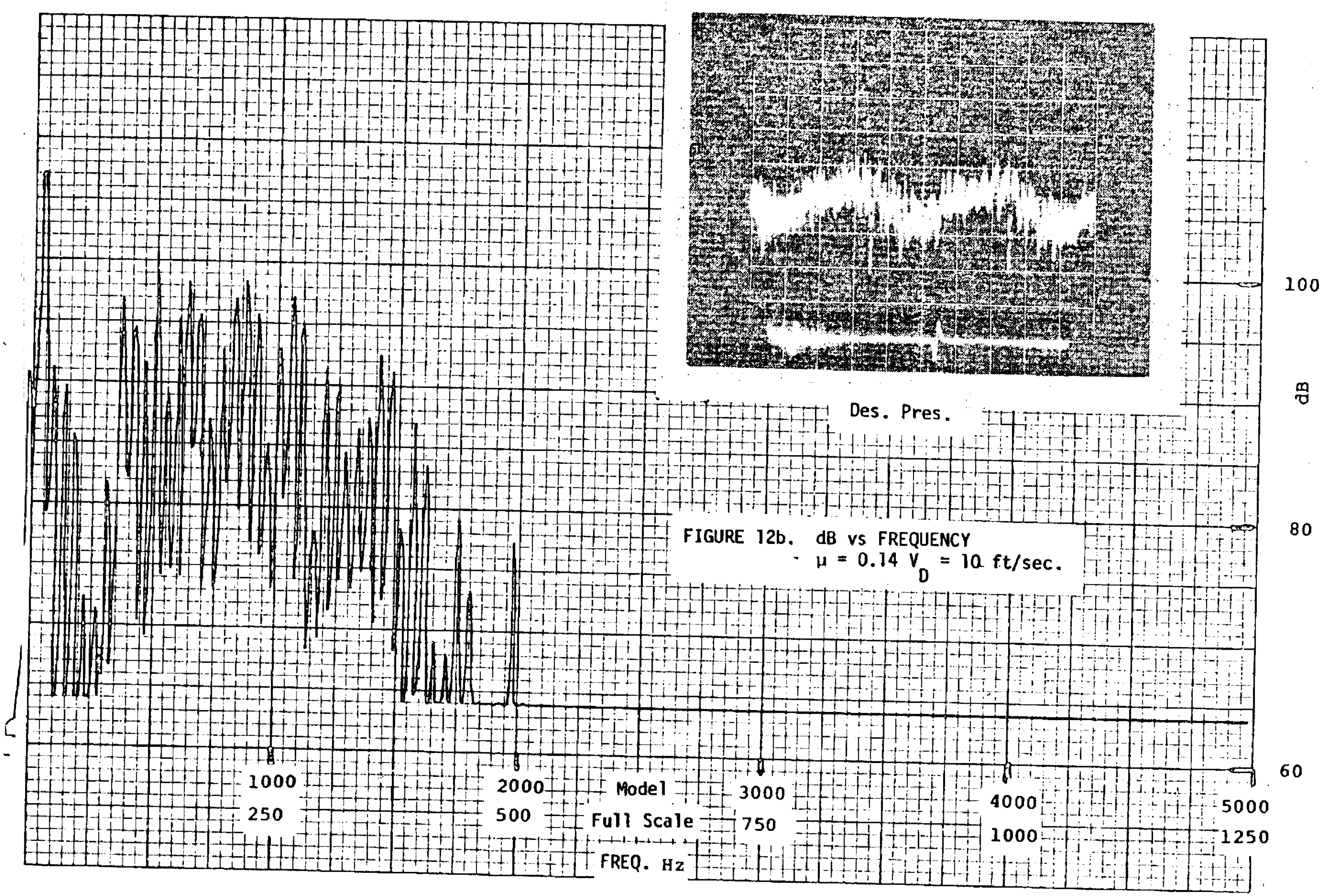


FIGURE 12b. dB vs FREQUENCY
$\mu = 0.14$ $V_D = 10$ ft/sec.

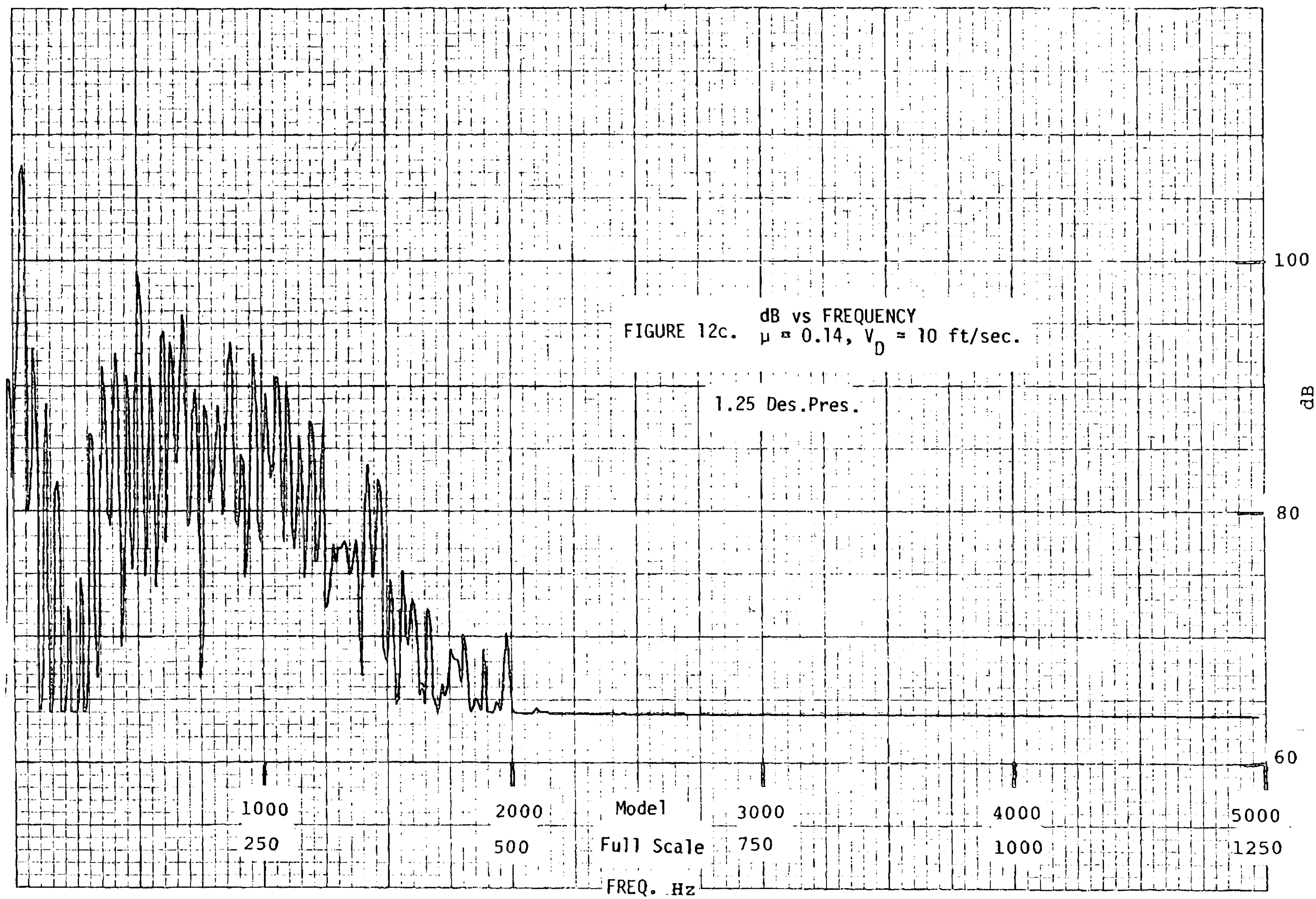


FIGURE 12c. dB vs FREQUENCY
$\mu = 0.14$, $V_D = 10$ ft/sec.
1.25 Des.Pres.
100
80
60
dB
1000
250
2000
500
Model
Full Scale
3000
750
4000
1000
5000
1250
FREQ. Hz

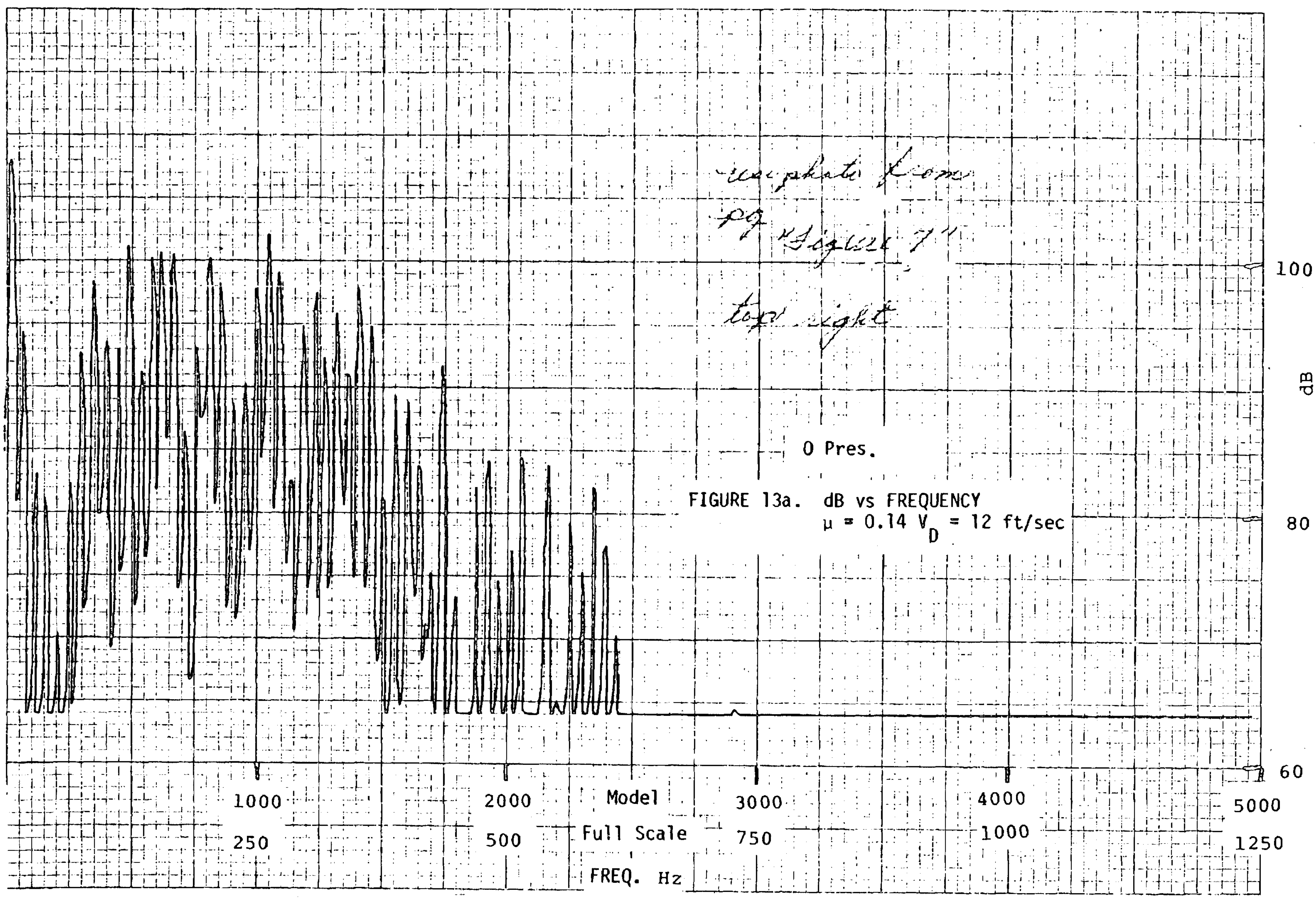


FIGURE 13a. dB vs FREQUENCY
$\mu = 0.14$ $V_D = 12$ ft/sec

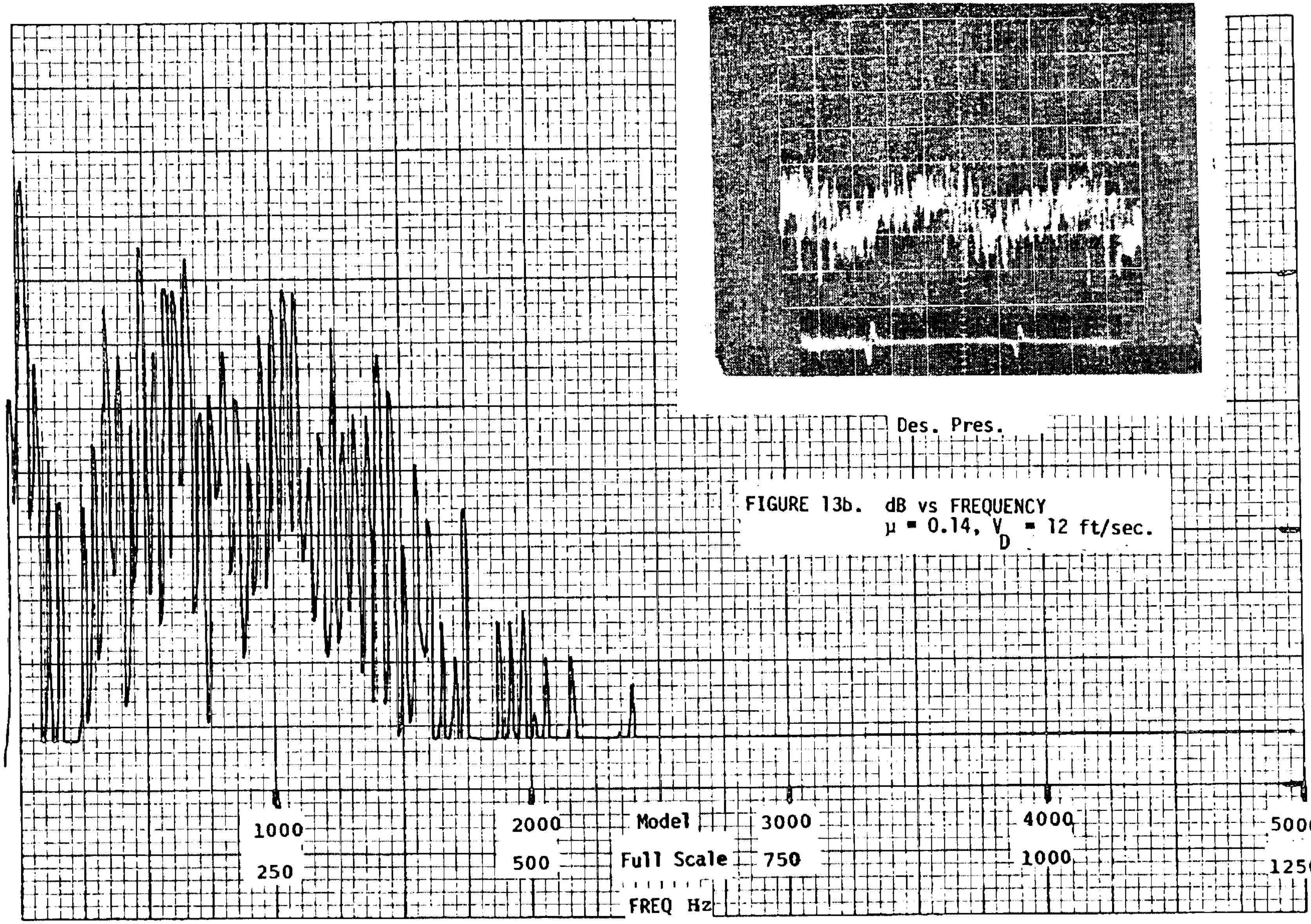


FIGURE 13b. dB vs FREQUENCY
$\mu = 0.14$, $V_D = 12$ ft/sec.

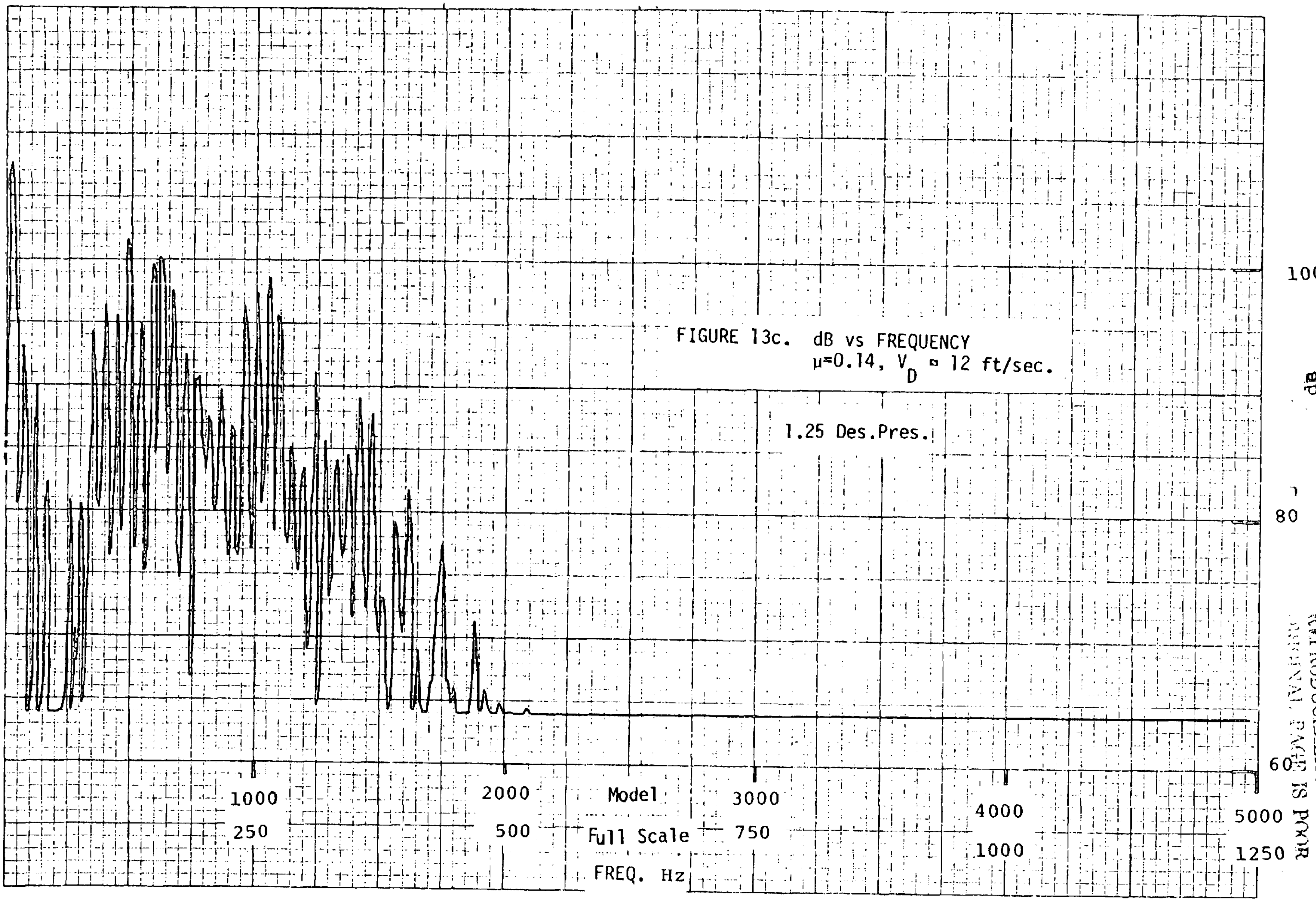


FIGURE 13c. dB vs FREQUENCY
μ=0.14, V_D = 12 ft/sec.

1.25 Des.Pres.

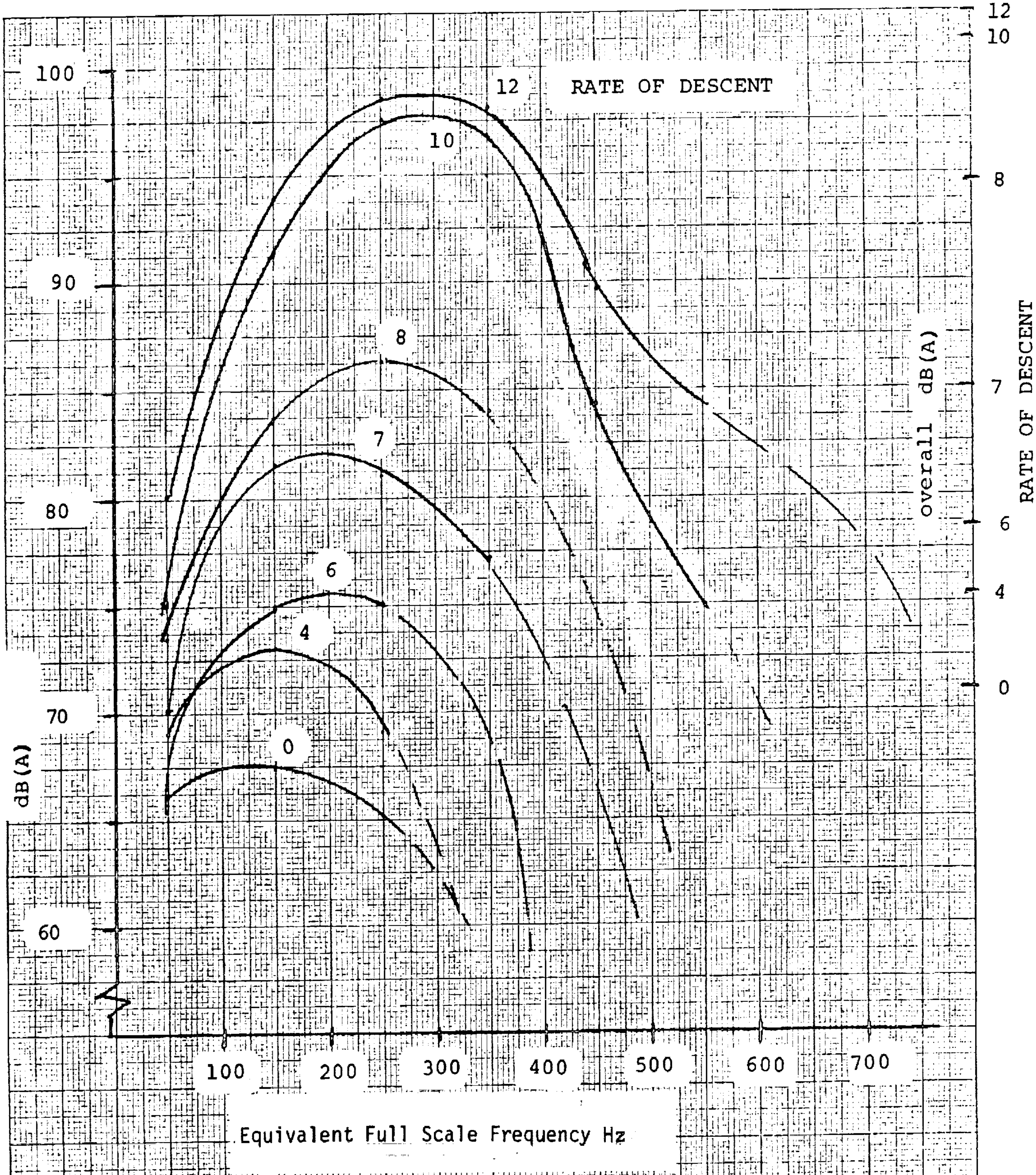


FIGURE 14. dB(A) FOR VARIOUS RATES OF DESCENTS AT $\mu = 0.14$

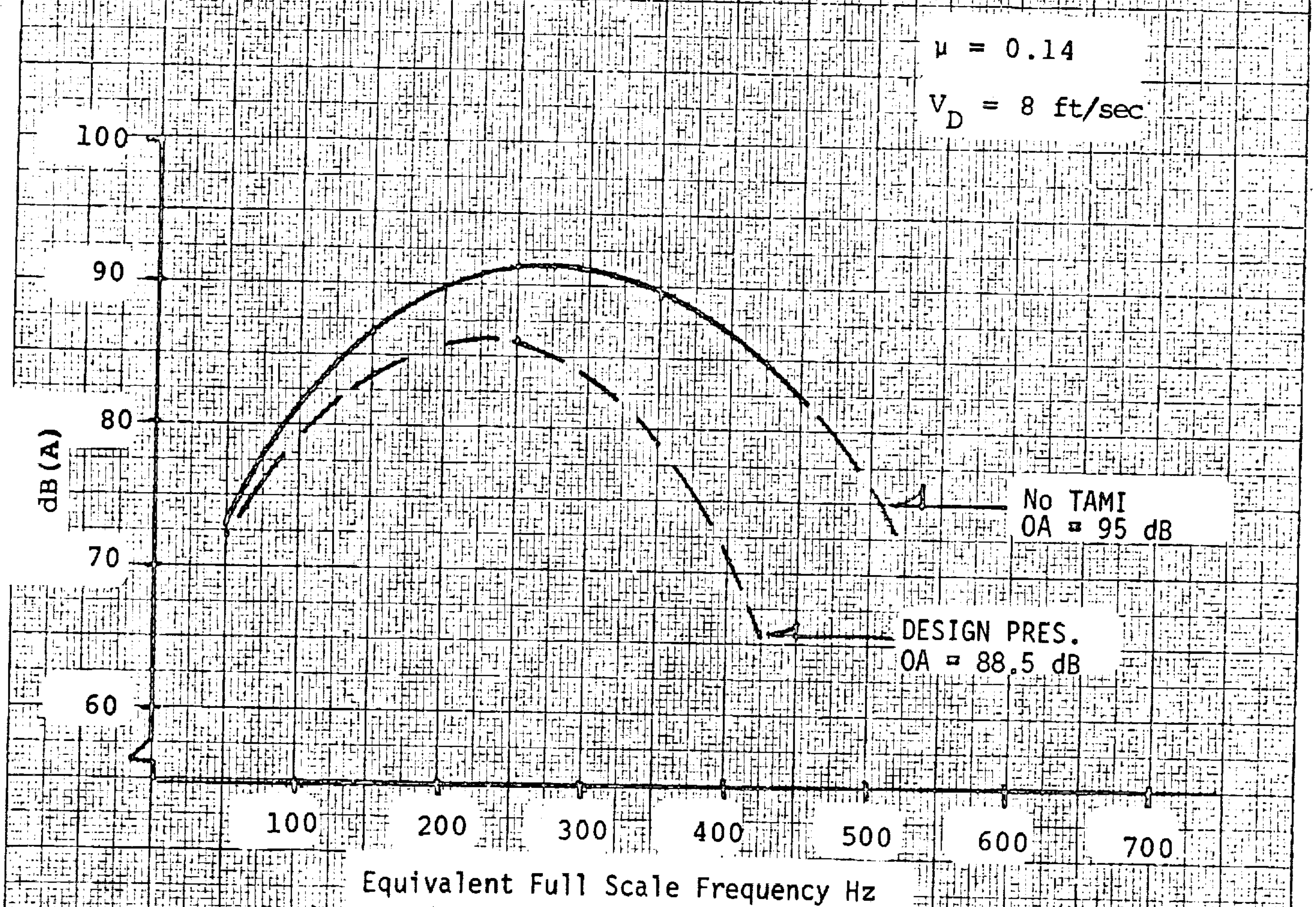


FIGURE 15 dB(A) WITH AND WITHOUT TAMI

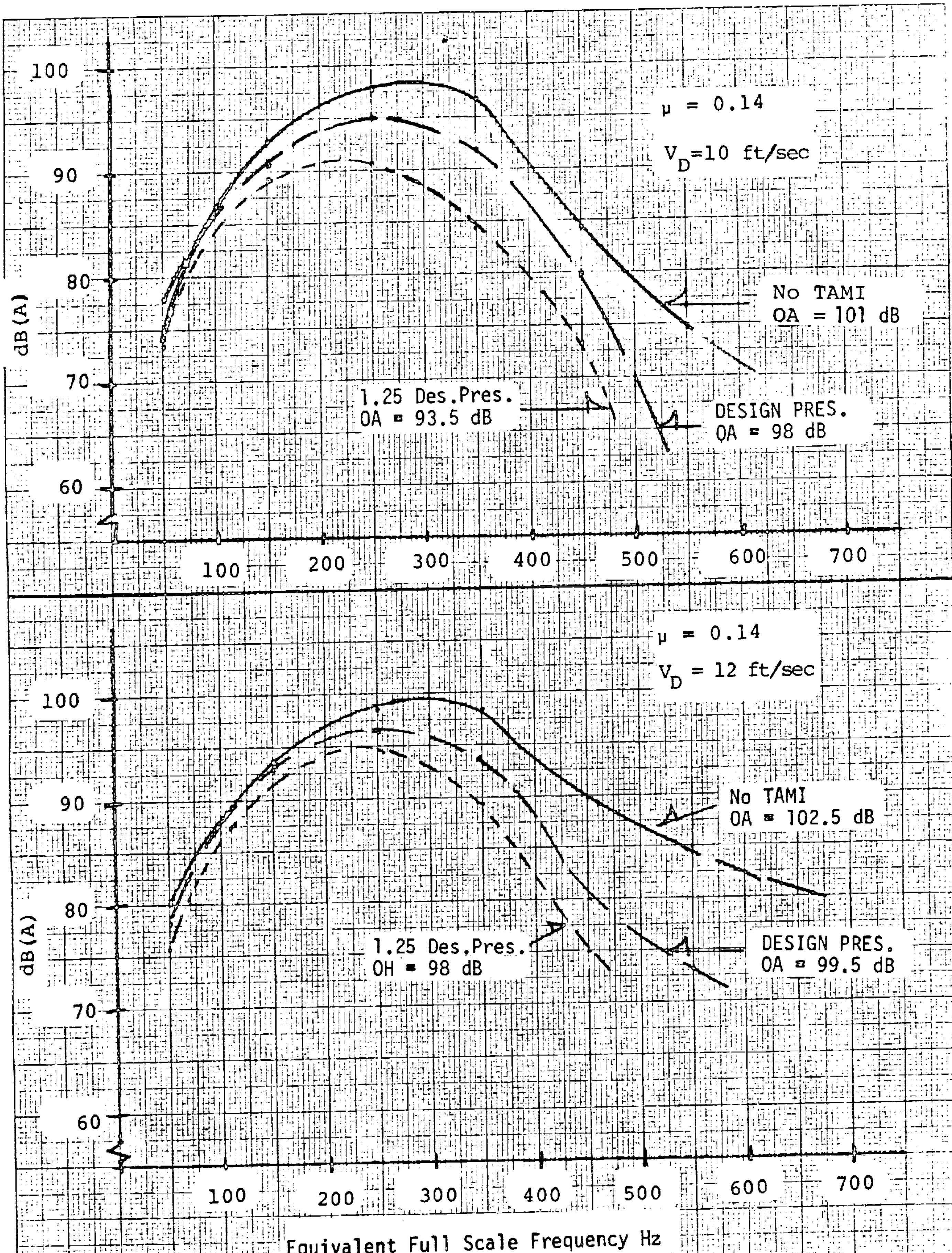


FIGURE 16 dB(A) WITH AND WITHOUT TAMI

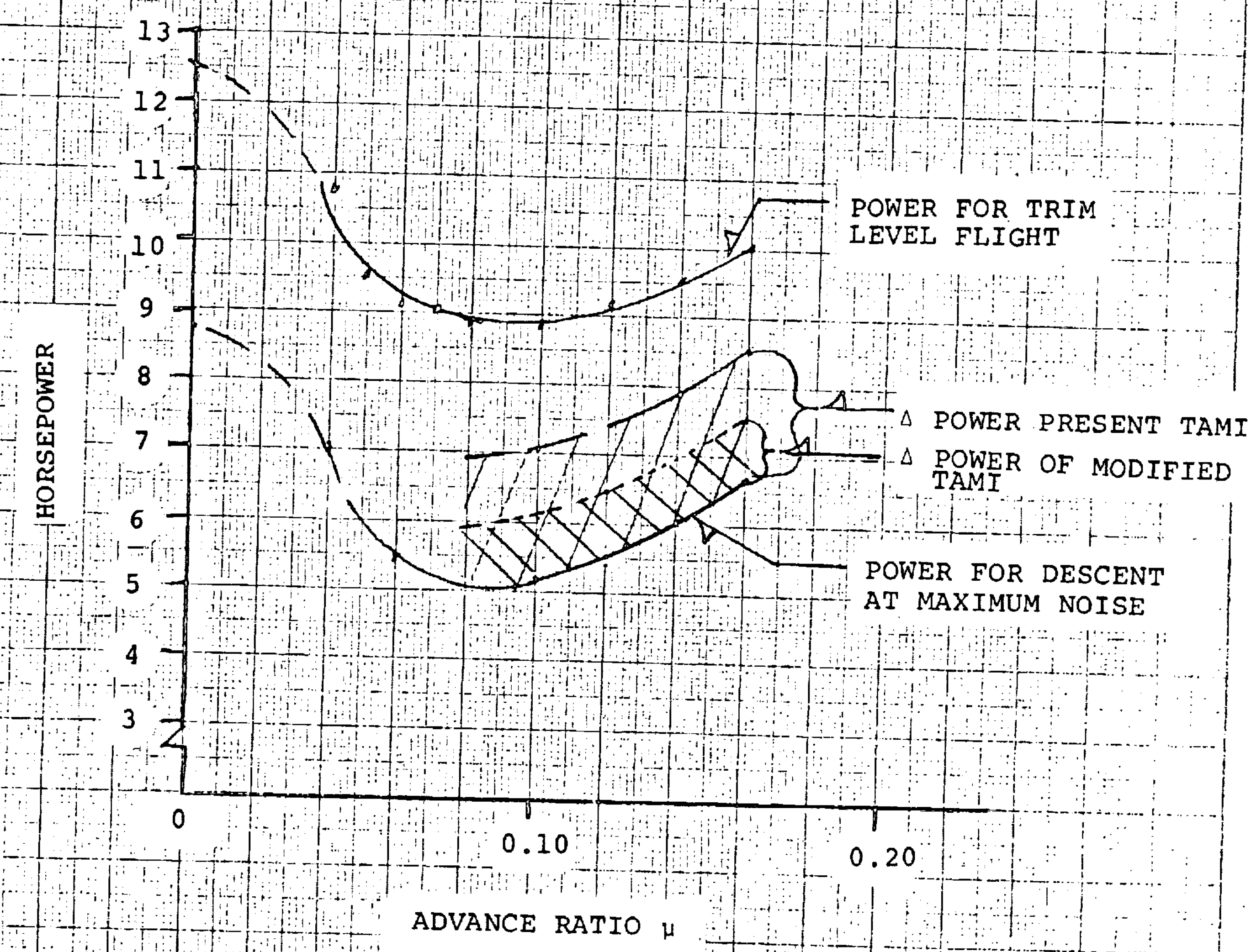


FIGURE 17 HORSEPOWER vs ADVANCE RATIO

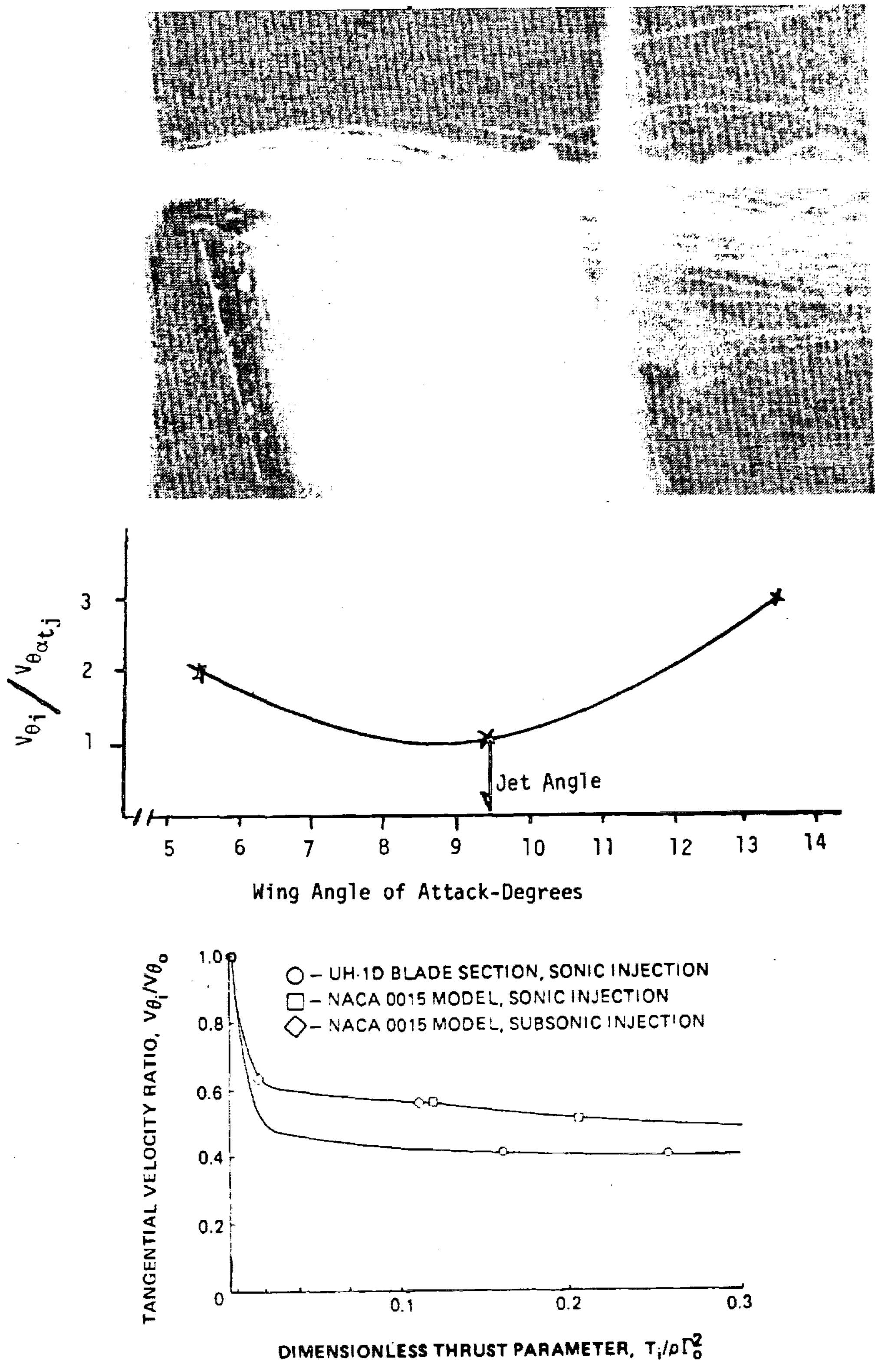


FIGURE 18 - Effect of Relative Vortex-Jet Angle and Model Scale on TAMI