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THE UNIVERSITY OF KANSAS CENTER FOR RESEARCH, INC.
2291 Irving Hill Rd.—Campus West Lawrence, Kansas 66044

Progress Report

Covering Period March 1, 1977 to June 17, 1977

for

A RESEARCH PROGRAM TO REDUCE INTERIOR NOISE IN GENERAL AVIATION AIRPLANES

KU-FRL-317-2

NASA Grant NSG 1301

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List of Symbols

Symbol	Definition	Dimensions
c	Speed of sound	m/sec
$D = \dfrac{Eh^3}{12(1-\nu^2)}$	Flexural rigidity of a plate	Nm
E	Young's modulus	N/m^2
f_1	Fundamental frequency	Hz
h	Plate thickness	m
K	System stiffness	N/m
M	System mass	kg
m	Panel surface mass	kg/m^2
$\bar{m}$	Panel surface weight	N/m^2
p, p_z, $p(x,y,z,t)$	Sound pressure	N/m^2
P	Peak value of sound pressure	N/m^2
$PSD(\omega)$	Power Spectral Density	
Q	Source distribution	N/m^2
$R_{12}(x_1,x_2,\tau)$	Space time correlation coefficient	
s	Fraction of surface mass fully participating in panel motion at resonance	
SPL	Sound Pressure Level (re. $2 \times 10^{-5} N/m^2$)	dB
T	Time period	sec.
t	Time	sec.
TL	Transmission Loss	dB
V	Volume of a cavity	m^3
$w(x,y,t)$	displacement component of a panel perpendicular to its surface	m
x,y,z	Cartesian coordinates	m

Greek Symbols

Symbol	Definition	Dimensions
ζ	Damping ratio	
$\eta = 2\zeta$	Damping factor	
λ	Wave length	m
τ	Time difference	sec.
ω	Frequency	rad/sec.
$\nabla^2 = \dfrac{\partial^2}{\partial x^2} + \dfrac{\partial^2}{\partial y^2} + \dfrac{\partial^2}{\partial z^2}$	Laplace Operator	

1. Introduction

This report continues the documentation of the research described in Report KU-FRL 317-1 which detailed research accomplished from April 15, 1976 through February 1, 1977, under the funding of NASA Grant NSG 1301. Report KU-FRL 317-1 contained information regarding preparations for a long range follow-up research program including (1) the development of an effective and competent research team at the University of Kansas, (2) the definition of this follow-up program (including pertinent NASA proposals), and (3) the design of a laboratory facility for acoustic testing of light weight aircraft structures.

In the period between February 1, 1977, and May 1, 1977, these activities were completed. A proposal for a follow-up interior noise research program was submitted to NASA in March 1977. This proposal (which was included in Report KU-FRL-317-1) was accepted by NASA in April 1977, and preparations for the follow-up program were then intensified. The construction of the acoustic test facility (a plane-wave tube) was initiated. A description of this facility is given in Chapter 2.

Manufacturers of sound reduction treatments (i.e. panel vibration damping and absorptive materials) were contacted about the existence and availability of materials suitable for light-weight aircraft structures. Information with respect to these activities is documented in Chapter 6.

A large portion of the activities was dedicated to studying the relevance of KU-FRL test results in predicting (theoretically or semi-empirically) interior noise levels in general aviation aircraft. Sections 3 thru 8 report about some pertinent considerations. As a result of this study and discussions with Mr. D. Stephens (NASA project monitor) and K.U. investigators, it was decided to make a few additions to the program as described in the NASA proposal of March 1977. These additions are:

(1) To use three (instead of two) noise sources in the plane wave tube to evaluate the influence of excitation spectrum

on panel response. The three sources will be: a) white noise, b) pure harmonic sound (of variable frequency) and c) actual general aviation fuselage panel excitations (as measured in flight).

(2) To use theoretical and experimental data obtained in the course of the project to develop more efficient noise reduction materials (or procedures to apply these), or to develop guidelines for the design of such materials for procedures.

(3) To use nonstructural materials in the collection of specimens to be tested in the KU-FRL plane wave tube. The original intent was to study the sound transmission through bare and acoustically treated skin and window panels of general aviation aircraft. As the intent of the program is to study sound transmission "in general", while both structural and non-structural "basic" panels are being applied in aircraft, it was decided to include noise-reduction-efficient non-structural basic panels.

A flowchart of events included in NSG 1301 is shown in Figure 1.1. This figure summarizes progress up to June 17, 1977.

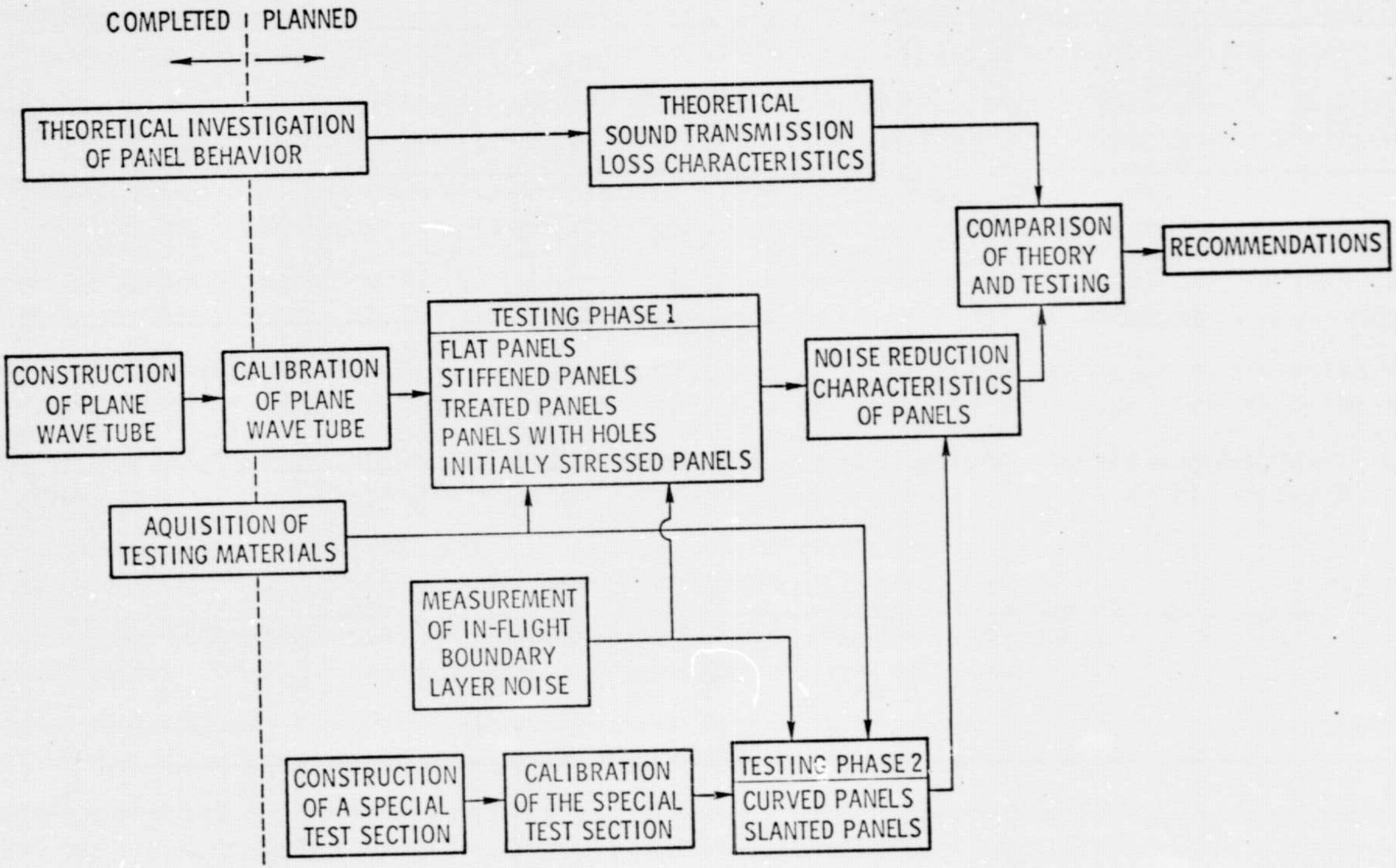


Figure 1.1. Flowchart of Program Activities

2. KU-FRL Test Facility and Procedure

2.1 Description of Plane Wave Tube

Laboratory measurements of Transmission Loss (TL) can normally be made using either a reverberant chamber or an acoustic tube. The reverberant chamber provides random incident noise which is statistically uniform over the test specimen. In the acoustic tube, the sound is propagated normal to the specimen's surface (or at some predetermined angle). Since it is intended to explore the influence of angle of sound incidence on selected panels, the use of the acoustic tube was the practical selection. Financial limitations also made this choice more attractive.

The plane wave tube used in this project is similar to the one designed by L. Beranek for his early work with sound control in airplanes (Ref. 11). A sketch of the basic tube is shown in Figures 2.1 and 2.2. These sketches do not include the pressurization system or the section required for the testing of panels at various angles of sound incidence. Sketches of these are shown in Figures 2.3 and 2.4. The panel to be tested is mounted between two chambers. The source chamber contains 9 high quality speakers to maintain a uniform sound pressure on one side of the panel. To minimize standing waves, the loudspeaker baffle is separated from the panel under test by a small distance, in which sound absorbing material is applied behind the baffle and between the loud speakers in front of the baffle. The receiving chamber is a termination which absorbs almost all of the sound which passes through the panel.

To determine the effects of aircraft pressurization on the transmission characteristics of a panel, the source chamber's static pressure will be reduced in increments of 2 psi up to a maximum of 6 psi differential pressure across the panel.

The loud speakers will normally be driven by the output of a white noise generator, amplified by a common power amplifier. Figure 2.5 shows the electronic equipment being used in this project. For a few select panels the transmission loss characteristics obtained in this manner will be compared to those measured using a pure tone or an actual aircraft noise input.

To allow for the testing of panels at various angles to the direction of sound propagation, a test section will be constructed that will be placed between the two existing sections (i.e. speaker box and termination). This test section will be constructed in a way so as to allow for testing of curved as well as flat panels. Sound pressures in both the source and receiving chambers are measured by high quality ½" microphones placed near the panel on each side. Their signals are averaged, analyzed, and subtracted by a (SD-335) Real Time Analyzer, following both accurate and time and cost effective data reduction. Figures 2.6, 2.7 and 2.8 show the tube in various stages of construction.

2.2 Test Procedure

While a test specimen is subjected to a constant (and accurately defined) excitation, the signals of the microphones located at both sides of the plate will successively be analyzed. First the primary (source) signal will be analyzed and stored by the SD-335 analyzer, and then, the secondary (termination) signal will be analyzed and subtracted from the primary signal. The difference will finally be plotted as a function of frequency on an X-Y plotter.

Though each of these activities is quite short (in the order of 1 minute), it is expected that the average test period for 1 sample (i.e. including calibration, installation of panels, etc.) will be approximately half an hour. Such an estimate seems even more realistic for cases in which, for example, pressurization is applied (and closely monitored).

The graphical test results will be corrected to account for the presence of reflected sound on the primary side of the panel. An approximation of the transmission loss (T.L.) of a specimen will be obtained after subtraction of the reflected sound from the recorded difference between the two microphones. This correction factor will be measured and calculated during the calibration of the plane-wave tube. Since its magnitude depends on the specimen's sound transmissivity, the correction is frequency dependant. As a result, it is estimated that the subtraction procedure will take another half an hour.

After a thorough calibration of the plane-wave tube (possibly including modifications), the results obtained in this way should be approximations of the transmission loss (T.L.) of the specimens. Obviously, the calibration and modifications are extremely important, since these should yield (1) the optimum microphone locations (to obtain "mass-law" attenuation at frequencies above the plate's resonance region), and (2) corrections for reflected sound at the primary microphone. Just as important is the minimization of anomalies in excitation characteristics (due to standing waves and loudspeaker differences). However, if the tube is calibrated meticulously, it is expected that testing and data reduction of 1 specimen will take approximately 1 hour.

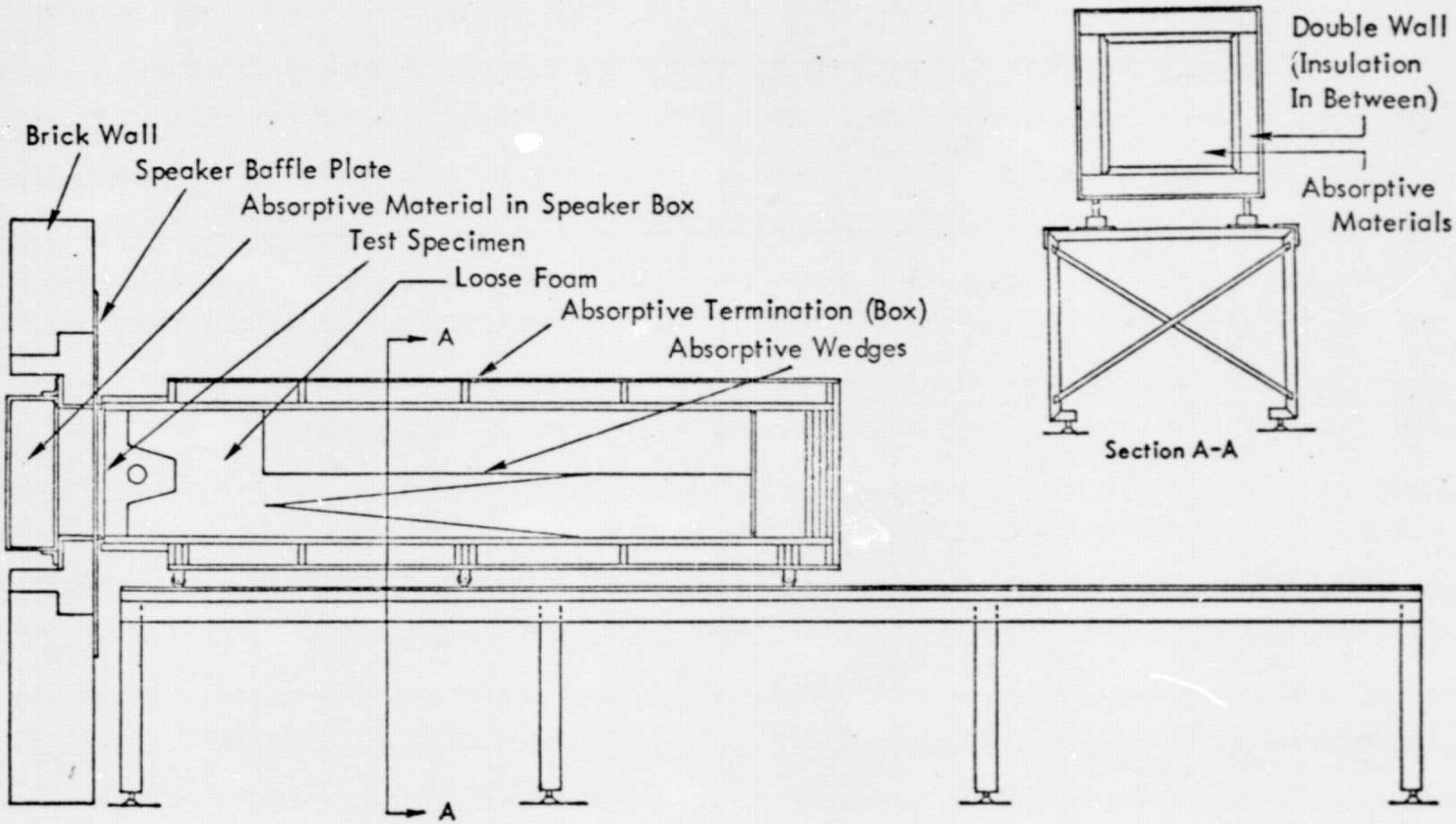
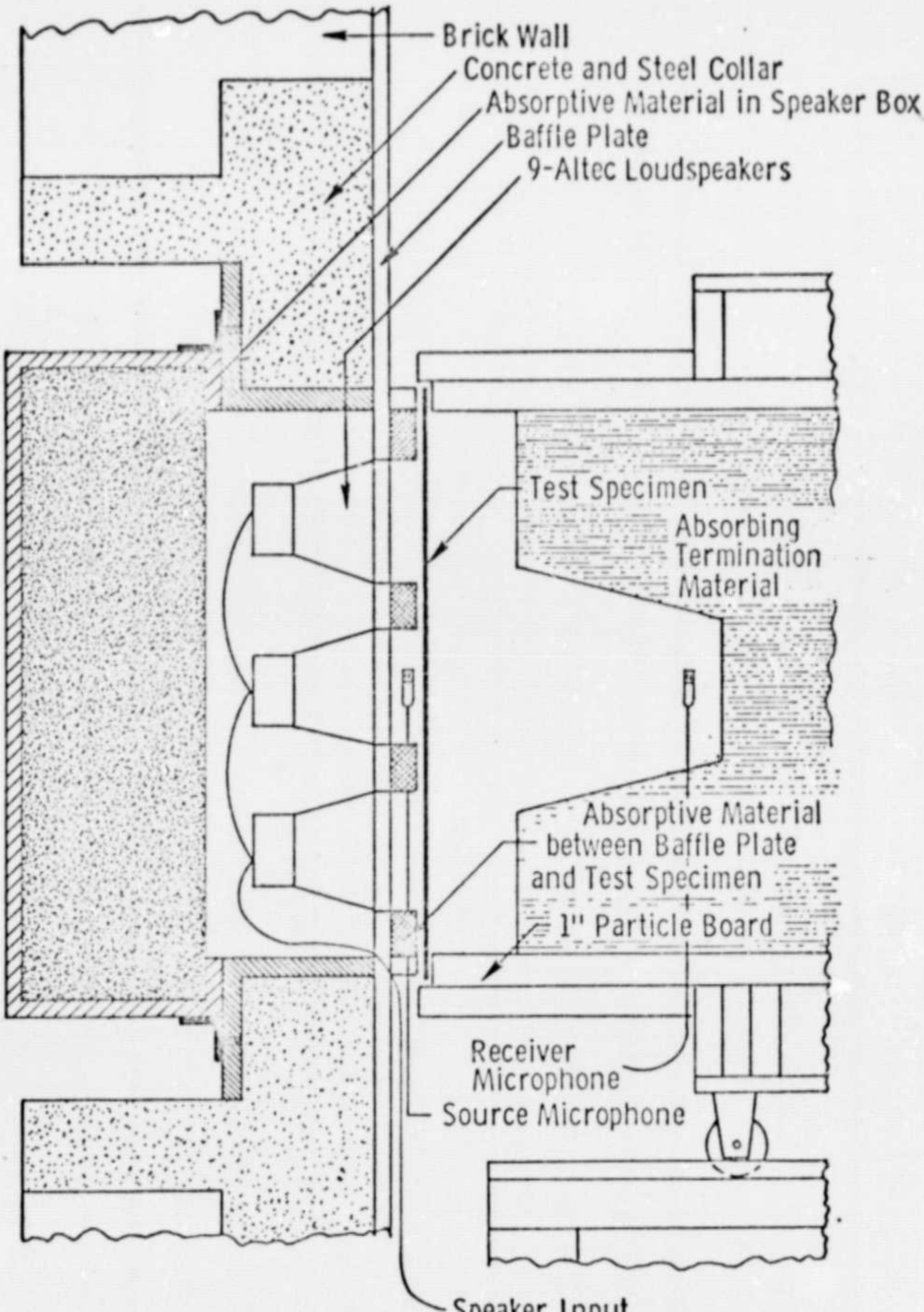


Figure 2.1. Plane Wave Tube



Note: Microphone Locations to be Optimized

Figure 2.2. Detail Drawing of Test Specimen Installation

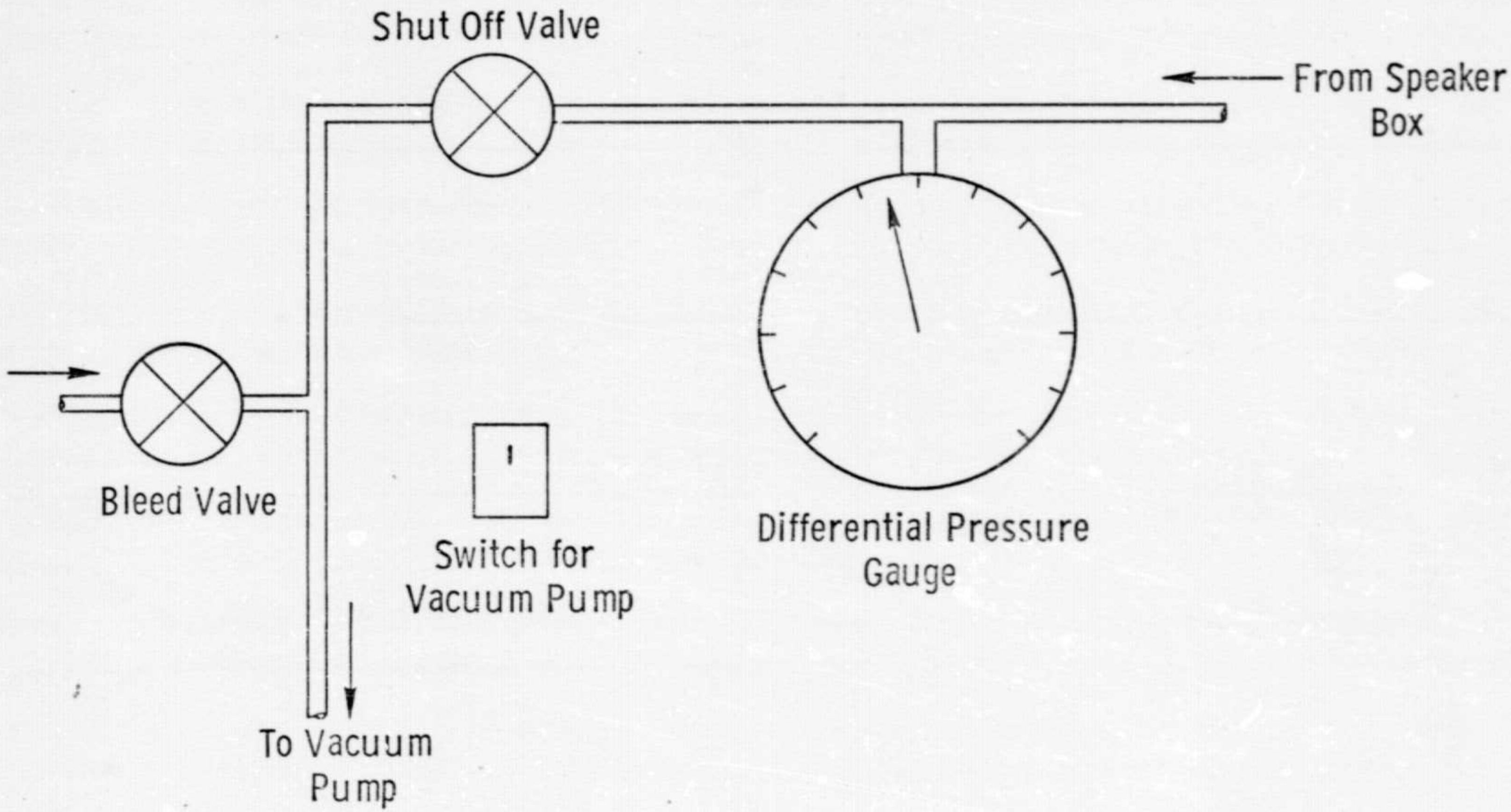


Figure 2.3. Pressure System for Plane Wave Tube

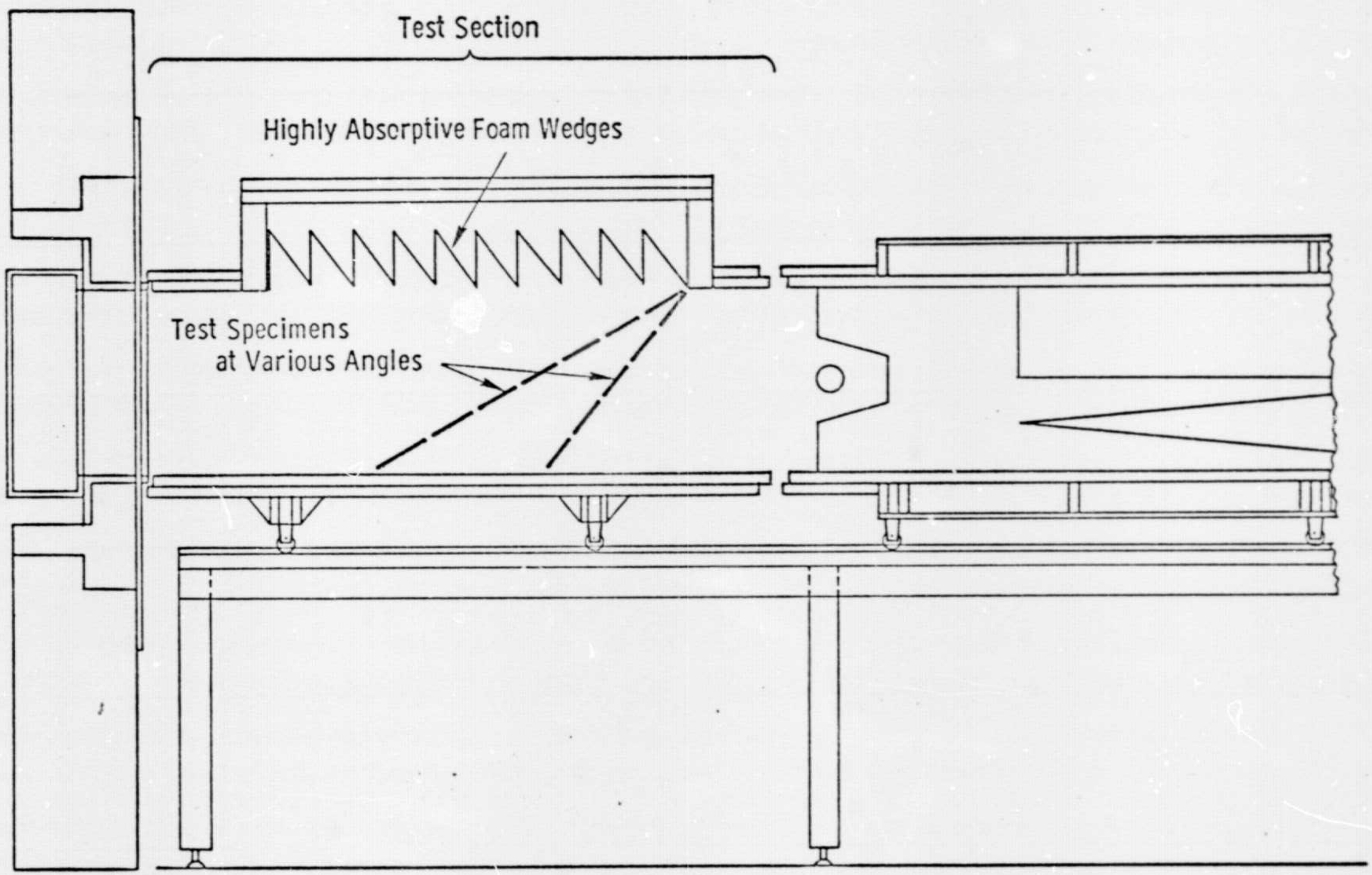
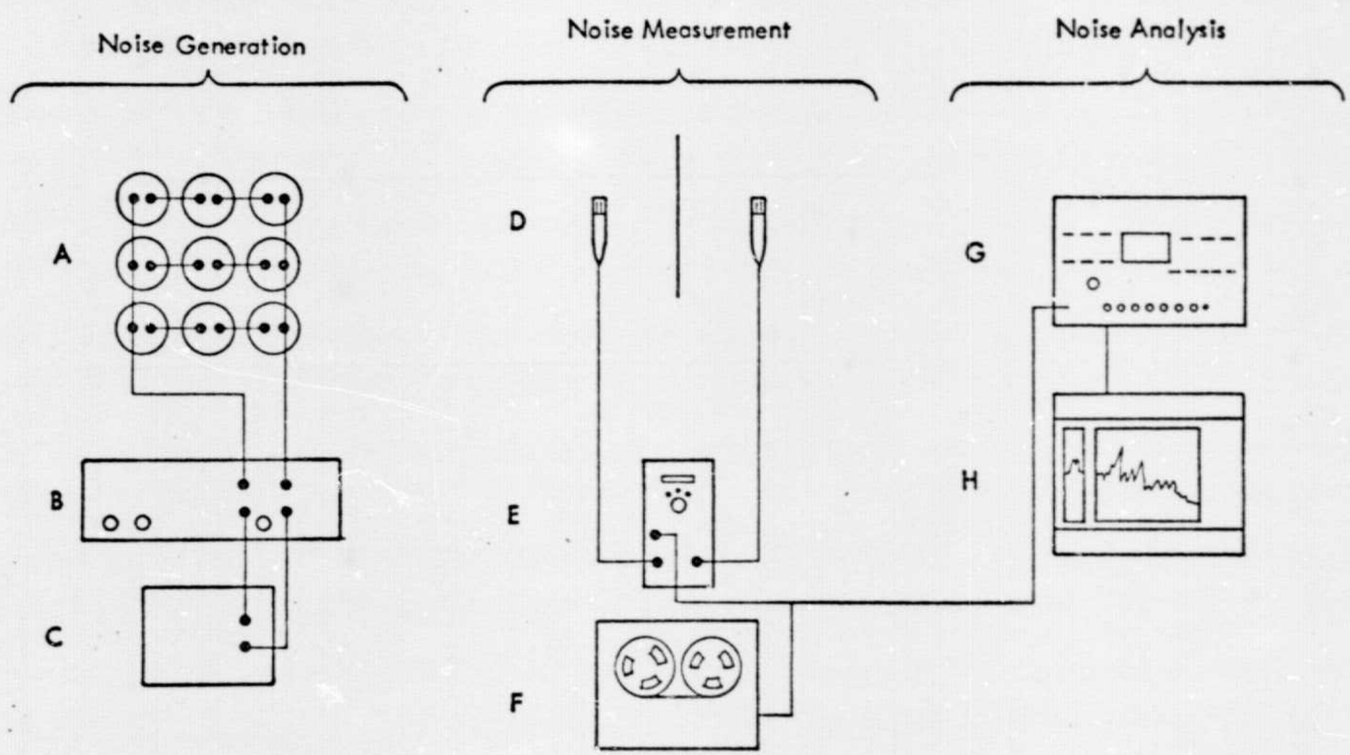


Figure 2.4. Test Section for Plane Wave Tube



A. Altec 405-8G Loudspeakers

B. Crown D-150 Power Amplifier

C. Noise Source
   i.) General Radio 1390-A Random Noise Generator
   ii.) Tape Recording of Actual Aircraft Noise
   iii.) Spectral Dynamics Model SD 104-5 Sweep Oscillator

D. B & K 4136 Microphones with 2618 Preamps

E. B & K 2804 Microphone Power Supply

F. Nagra SJS Tape Recorder

G. Spectral Dynamics Model SD335 Real Time Analyzer

H. HP 7045-A X-Y Recorder

Figure 2.5. General Arrangement of Electronic Equipment

Figure 2.6 Construction of Concrete
Collar for speaker base

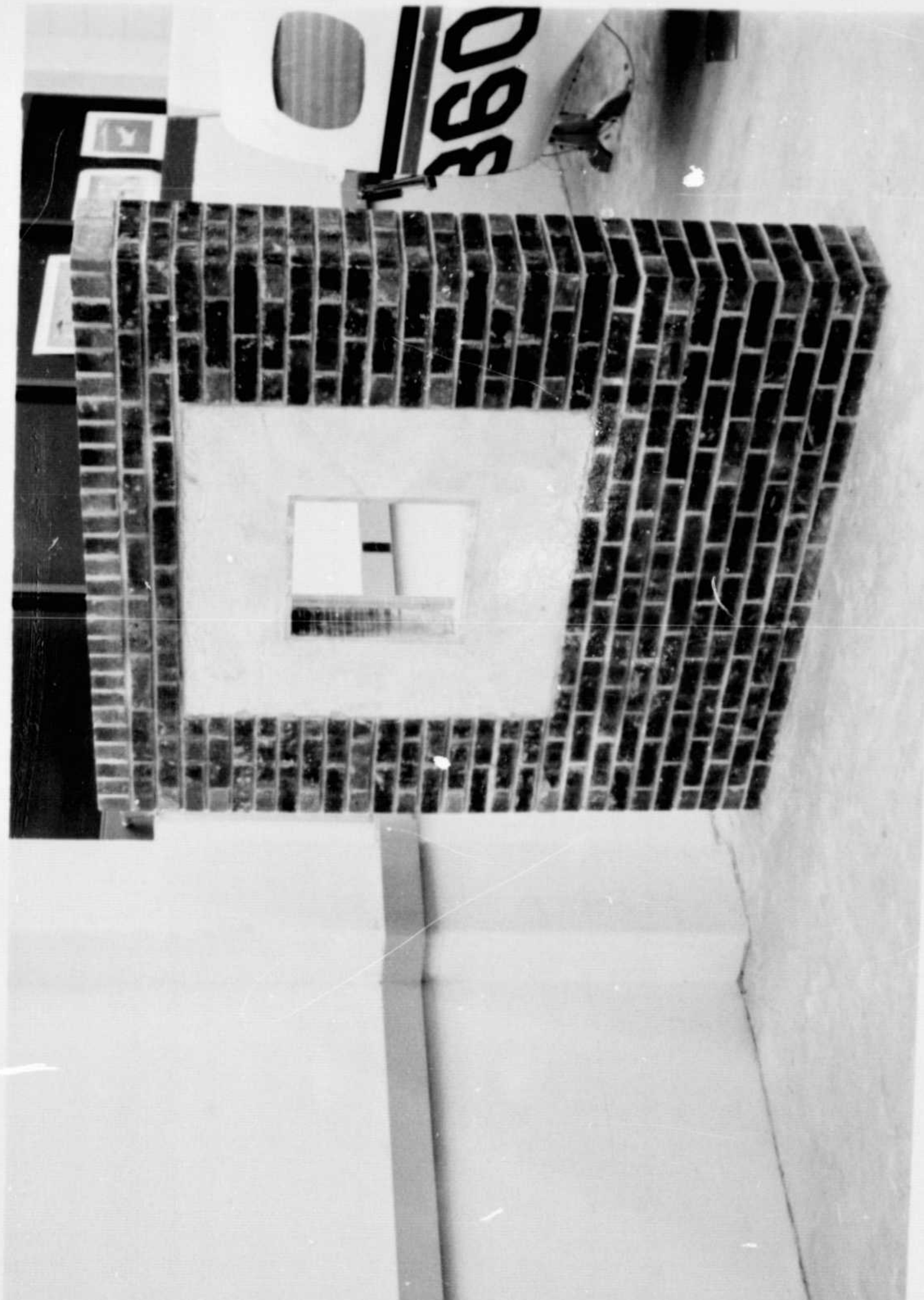


Figure 2.7 Concrete Collar in place

Figure 2.8 Partial Construction of plane wave tube in Flight Research Lab

3. KU-FRL Test Procedure Versus ASTM* Recommended Practice

The purpose of this section is to indicate some of the differences between the KU/FRL test method and the procedure recommended by the American Society for Testing and Materials for measurement of panel sound transmission loss. It should be mentioned that there are several other test practices (for example: using a reverberant source room and an anechoic termination), all yielding a different kind of panel noise reduction. The Transmission Loss (TL) of the panel can be obtained from this kind of data, by correcting for room effects and by selecting the right microphone locations. After a calibration period, the KU-FRL noise research team will provide similar corrections for its plane-wave tube.

3.1 ASTM* Recommended Practice for Measurement of Sound Transmission Loss

A test procedure for measurement of sound transmission loss of materials is specified by and described in ASTM Standard E-90-70, "Standard Recommended Practice for Laboratory Measurement of Airborne Sound Transmisison Loss of Building Partitions". To measure the transmission loss of a specimen it is mounted in the connecting opening between two reverberation rooms. Care is taken to assure that the only sound path between the two rooms is through the specimen. The rooms should be large enough to support a diffuse sound field at the lower frequencies. This requirement is expressed through the relation: $V > 4 \times \lambda^3$, this means that the volumes should be at least 45,000 ft.^{3} to maintain such a field at frequencies as low as 50 Hz. The minimum dimensions of the specimen should be at least 8 by 8 ft. to avoid the possibility that the method of clamping the boundaries of the specimen will affect the Transmisison Loss (TL) measurements.

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* American Society for Testing and Materials

The application of this test procedure has certain implications with regard to its test results. The use of a diffuse sound field can result in a different panel behavior than the use of plane waves (see section 4). The room volumes required for low-frequency measurements are enormous and (due to financial constraints) not possible in a KU-FRL noise research project. However, the use of well-chosen absorptive materials in a plane-wave tube can result in a perfectly anechoic termination (as opposed to a reverberant receiving room) while standing wave effects on the source side of the test panel can be minimized. The large (ASTM) panel size will, in all practical (general aviation) cases, eliminate the effects of panel resonances on the transmission loss characteristics, which the panel size in the KU-FRL tube will certainly facilitate studies in this important frequency region. Finally, the commutation of several microphone outputs in both source and receiving room will result in average transmission loss results that are typical for an ASTM-type procedure. In the KU-FRL test facility, the use of just one microphone situated close to both source and receiving side of the test panel will, at low frequencies, result in position dependent Transmission Loss characteristics.

4. Influence of Type of Excitation on Panel Sound Transmission Characteristics

This section indicates that the response of a panel to an excitation depends on panel properties as well as excitation characteristics. It shows that, in order to generate panel sound transmission data in a laboratory that should be applicable to aircraft in flight, it is desirable to reproduce the actual aircraft environment as accurately as possible. Especially, the reproduction of the actual pressure distribution and phase-differences is hard to realize. The KU/FRL plane-wave tube generates its own characteristic excitations, which are not identical with the actual aircraft environment. As a result its data are not identical to those obtained in flight. It is the objective of this section to warn against the use of the uncorrected laboratory data for predictions of aircraft interior noise levels.

In the case of an acoustic tube, the direction of propagation of sound waves is normal to the panel surface and the pressures are thus, theoretically, in-phase over the panel. The reverberant chamber provides randomly incident noise which (theoretically) is statistically uniform over the panel. To account for such differences the excitation field can be characterized by space-time correlation coefficients ($R_{12}(x_1, x_2, \tau)$)*. The space-time correlation coefficient of the sound pressure, giving a measure of the phase relationship of the pressures over the panel surface, is important in determining which types or modes of vibration will be excited by the pressures.

The greatly simplified governing differential equation of undamped motion of plates can be expressed by: (Ref. 7, 13)

$$D\nabla^2\nabla^2 w(x,y,t) = P_{z_1}(x,y,t) - P_{z_2}(x,y,t) - m\,\frac{\partial^2 w(x,y,t)}{\partial t^2} \qquad (4.2)$$

$$* \; R_{12}(x_1, x_2, \tau) = \lim_{T\to\infty} \frac{1}{2T} \int_{-T}^{+T} F_1(x_1, t)\, F_2(x_2, t+\tau)\,dt \qquad \text{(Ref. 8)} \qquad (4.1)$$

where: F_1 and F_2 are the sound pressures at two points x_1 and x_2 in an acoustic field.

The dynamic parameters of this system are shown in Figure 4.1

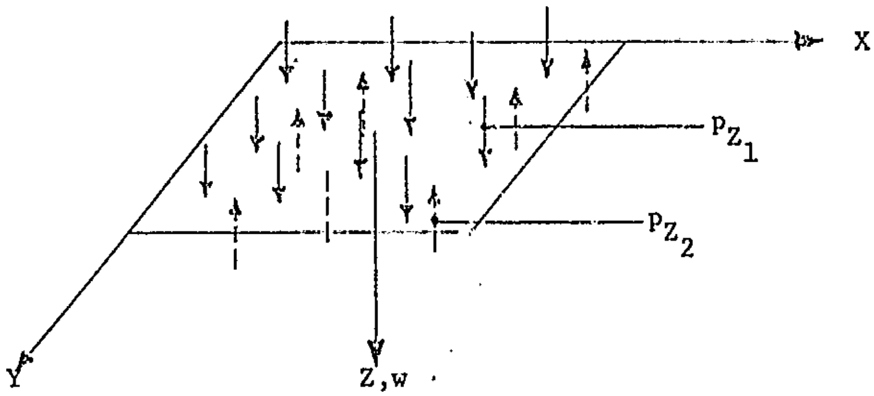

Fig. 4.1: Dynamic Parameters of a Vibrating Plate.

The particular solution of this equation is associated with the panel excitations and has been evaluated for many loading conditions (Ref. 7). For example, using a Fourier analysis it can be proven quite simply that in the case of a uniform harmonic pressure, even order vibration modes cannot be excited.

Since such an acoustic excitation is generated in a plane-wave tube, these modes are not expected to show up in the KU/FRL test results. If the excitations vary randomly with time, a Power Spectral Density analysis can explain the nature of the panel responses. Since the PSD of the panel response equals the PSD of the random excitations divided by the square of the amplitude ratio of the transfer function, most of the energy of vibration due to a random loading will be concentrated in narrow frequency bands around the plate's resonances. Though a basic randomly varying response can be expected, periodic responses with frequencies equal to those of resonances and beat phenomena will determine the overall vibrational character. It can thus be concluded that the type of excitation can have a significant influence on the plate's behavior. As a result an accurate reproduction of the actual noise environment in an acoustic test appears desirable, but, it is normally found that such accuracy cannot be obtained. Obviously, the value of a test would be judged after a comparison of other test methods with the actual noise environment. In this evaluation the following parameters could be of particular interest:

a) overall intensity and distribution over the specimen;

b) pressure spectrum and distribution over the specimen;

c) pressure correlation over the surface.

The differences between aircraft and laboratory panel sound transmission characteristics due to differences in pressure correlations or frequency contents are hard to quantify. Some simplified prediction methods (which assume that the fundamental plate mode is predominant and that the pressure is exactly in phase over the whole panel) appear to produce quite realistic results (Ref. 8,6). However, considering the analytical predictions explained above, it seems justified to warn against the use of (uncorrected) test data for aircraft interior noise predictions.

5. The Actual Excitation Field

In the previous section it was argued that an accurate representation of the actual panel excitations in a laboratory test could be desirable, if test results are to be used for aircraft interior noise prediction. First, this section will briefly describe the complicated character of these excitations. Since pertinent experimental data is very rare, it is the intention of the KU-FRL noise research team to do some pressure measurements in the boundary layer of a single engine general aviation aircraft (as stated in the NASA proposal of March 1977-Ref. 19). The second part of this section will describe these measurements, which will be used in the KU/FRL laboratory facilities as one of the three intended noise sources (the others being: white and discrete frequency noise). It should be emphasized that the actual frequency spectrum will thus be simulated, but not the actual pressure correlations.

5.1 Character of The Actual Excitation Field

The sound inside a general aviation aircraft cabin is caused by airborne and structure-borne sound from the engine and propeller and by aerodynamaic pressure fluctuations associated with the flow of air over the fuselage skin. Clearly, one of the ingredients for noise prediction is the definition of the total excitation field. Exterior noise spectra are expected to vary at different locations on the aircraft. In the near field of a propeller, sound levels and spectrum vary markedly with position, as is indicated by pertinent empirical prediction methods (note: not valid in propeller slipstream!)(Ref. 1,14).At some locations, engine exhaust noise is expected to have a significant influence on the exterior noise specturm.

Information on the spectra and correlations of these noise inputs on the fuselage in flight are unknown to the KU/FRL team. In flight, the noise inside a cabin can have its origin in aerodynamic boundary layer noise associated with the flow of air over the fuselage skin. The boundary layer pressure field is aerodynamic and does not have the characteristics of an acoustic field, but can be detected by a microphone (it was called "pseudo-sound" by Lighthill - Ref. 15). In the case of relatively slow general aviation aircraft, boundary layer pressure fluctuations over parts immersed in the free stream are quite small. Fuselages immersed in the propeller wake, are expected to have significant periodic dynamic pressure fluctuations in the boundary layer. An obvious effect of these fluctuations is the local excitation of the aircraft skin. It has been stated (For example Ref. 1) that the vibrating skin acts like a transducer (converting pseudo-sound to true sound) with a certain transmission loss. The interior sound pressure level, neglicting reverberation effects, is just pseudo-sound level on a decibel scale minus transmsision loss.

5.2 Measurement of Actual Panel Excitation

To accurately represent in the plane wave tube the sound spectra present around the fuselage of an aircraft in flight, sound recordings will be made in the boundary layer of a light aircraft during normal flight operations. With appropriate calibration, this recorded sound will be played back through the test panels in the plane wave tube to match the complex pressure fluctuations found in the propeller slipstream.

The measurements will be made in a 1975 Piper Cherokee 140. This aircraft was chosen because of the ideal microphone mounting locations available without modification to the airframe. The possibility of measurements in other aircraft was found to be less attractive because of the necessity of time consuming and costly modifications, while similar reasons make other locations in the Cherokee 140 unattractive. It is the opinion of the KU-FRL noise research team that the use of aircraft and locations described above will provide representative data at locations that are generally considered as "noise sensitive". Moreover, the data can be obtained at low cost and in a short time. Figure 5.1 shows the mounting locations that will be used. One microphone will be mounted flush in the windshield through the hole normally used for the outside air temperature gauge. Another microphone will be mounted in the pilot's storm window. A special mount can be made on a spare window and it can be inserted into the window opening without removing the standard storm window from its bracket.

Since vibrations of the structure accompany any sound measurements in an aircraft, it will be necessary to either isolate the microphones from the structure or compensate them for the vibrations present. Isolating the microphone from the stucture is difficult because of sealing problems between the microphone and the window panel, since any leakage of air into or out of the cabin will be picked up as wind noise. Some type of airtight diaphragm is necessary for complete sealing, though some vibration would probably still be transmitted to the microphone.

In another method, the vibrations of the structure could be measured with an accelerometer and subtracted from the sound pressure levels through the use of sound level/acceleration conversions. The ¼" Model 4136 Bruel and Kjaer microphones used in the program have a vibration influence of 90 dB/g in the axial direction so with small amplitude vibrations the actual outside sound pressure levels could be calculated from the overall recorded level. A theoretical analysis will be performed on the two methods before one is chosen.

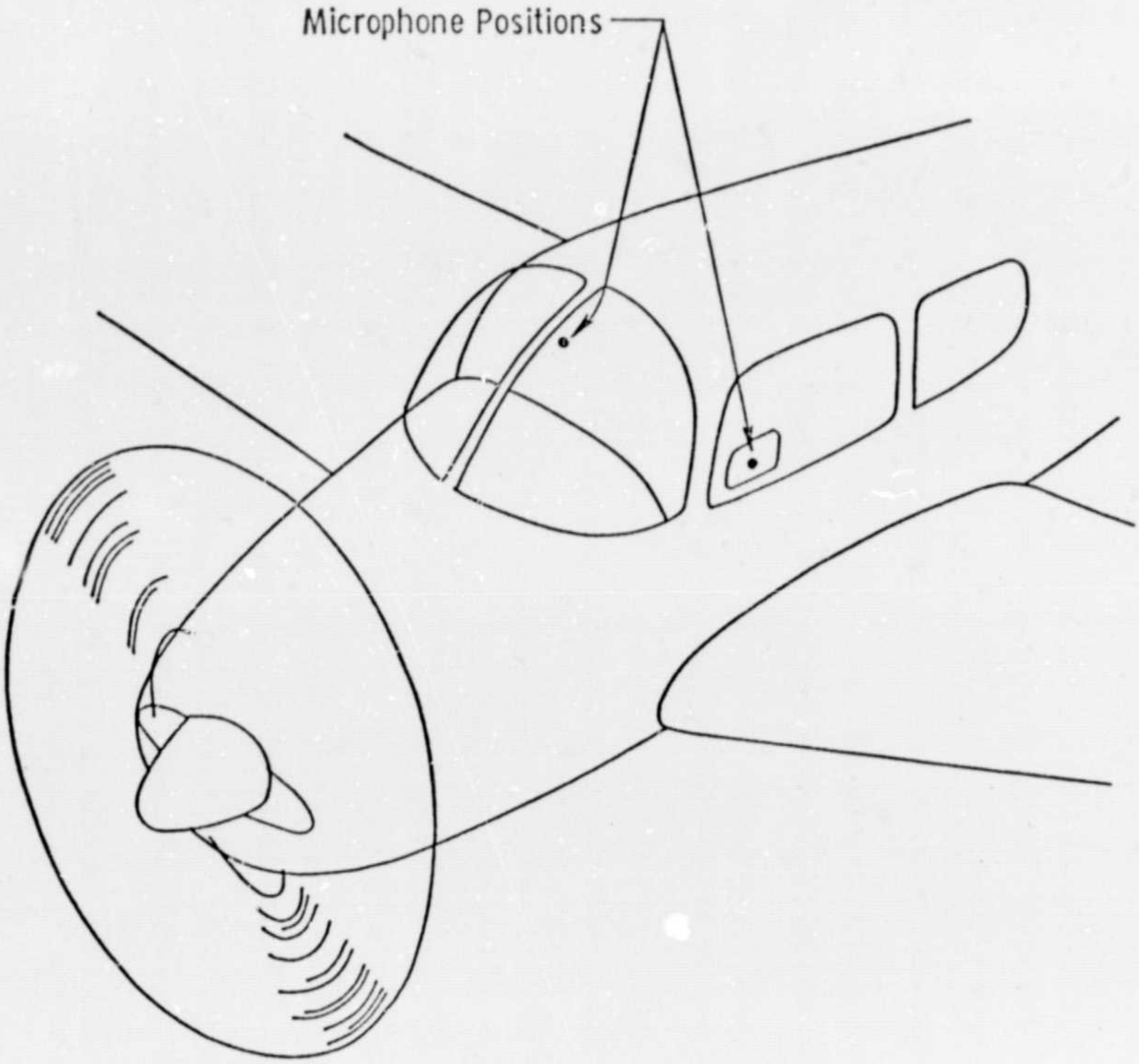

Figure 5.1: Microphone Mounting Positions on a Piper Cherokee 140

6. Aquisition of Testing Materials

Acoustic testing will be performed on various types of aircraft structural specimens normally found in the fuselage area of a light aircraft. These will include stiffened and unstiffened aluminum sheet from the fuselage sidewalls and doors, steel sheet from the firewall and plexiglass from the windows. In addition, panels of fiberglass sheet and composite sandwich materials will also be examined since these types of materials are finding increased usage in aircraft. All materials except the plexiglass will be treated with commercially available vibration damping material and retested to study the sound transmission loss characteristics of the combination.

The aluminum and steel base materials and plexiglass used in the testing program are being supplied by the general aviation manufacturers at no cost to the project. At the present time, two manufacturers have submitted materials including many thicknesses of plain aluminum sheet, supported aluminum sheet with various stiffening patterns, aluminum honeycomb panels and representative samples of aviation plexiglass. Various thicknesses of firewall steel have also been submitted. These materials are listed in Table 6.1.

The vibration damping materials that will be applied to the test panels are being supplied by commercial vendors, again at no cost to the project. A list of forty-five manufacturers of vibration damping and related noise control materials was obtained from Reference 12. These manufacturers were contacted in late April 1977 with requests for material suitable for aircraft use and the samples received so far have ranged from foam to paste to "deadened steel" vibration dampers. The seven manufacturers that have submitted samples to date are listed in Table 6.2.

The only materials that have not been solicited yet are the fiberglass sheets. A catalog and manufacturer search is being performed at this time to locate suitable samples.

Table 6.1. Aircraft-type Base Materials
Received by June 17, 1977

Company	Test Specimen
Cessna Aircraft Co.	.016" Aluminum Sheet
	.020" Aluminum Sheet
	.025" Aluminum Sheet
	.032" Aluminum Sheet
	.040" Aluminum Sheet
	.025" Stiffened Aluminum Sheet
	.025" Stiffened Aluminum Sheet
	.032" Al Sheet w/full coverage LD400*
	.032" Al Sheet w/18" x 18" LD400
	.032" Al Sheet w/14.2" x 14.2" LD400
	.032" Al Sheet w/3" edge of LD400
	.016" Steel (19" x 20")
	.020" Steel
	.032" Steel
	1/8" Plexiglass
	3/16" Plexiglass
	1/4" Plexiglass
Grumman American Aviation Corp. P.O. Box 2205 Savannah, Georgia 31402	Honeycomb panels

* LD400 is a vibration damping material supplied by Lord Corporation and used on most Cessna Aircraft.

Table 6.2. Acoustic Treatment Materials
Received by June 17, 1977.

Company	Test Specimen
Carney & Assoc., Inc. P.O. Box 1237 Mankato, Minn. 56001	Fiberglass - 1" thick
Chemprene, Inc. Div. of the Richardson Co. 570 Fishkill Ave. Beacon, N.Y. 12508	Foam - ¼" thick with backing
Foamade Industries 1220 Morse Street Royal Oak, Michigan 48068	Foam - 1" thick (2 & 4 16/ft.^{3})
Forty-Eight Insulations, Inc. Aurora, Illinois 60504	Fiberglass - 1" thick (6 16/ft.^{3})
Insul-Coustic Corp. Jernee Mill Rd. Sayreville, N.J. 08872	Visco-elastic paste used to bond secondary damping panel to primary sheet.
Singer Partitions, Inc. 444 North Lake Shore Dr. Chicago, Illinois 60611	Visco-elastic paste.
Specialty Composites Corp. Delaware Industrial Park Newark, Delaware 19711	1) Antiphon - 13™ vibration damping pads. 2) Antiphon - 13/foam sandwich. 3) High density foam pads. 4) Multi-density foam sandwich. 5) "Deadened Steel" (steel sheet sandwich with visco-elastic core.

7. Effects of Receiving Space

The objective of the KU-FRL noise project is to investigate experimentally and analytically the transmission of sound through aircraft type panels. The results will thus be valid for isolated panels only, and not for panels installed in aircraft. To use the KU-FRL results for aircraft interior noise prediction, the influence of the cavity behind the plate, the surrounding plates and the absorption inside the cavity should be taken into account.

The KU-FRL noise research team has dedicated some of its time studying these effects and their magnitudes. The first part of this section briefly describes the importance of receiving space effects, and the second part explains the mathematical complications that will be encountered when trying to calculate their magnitudes. Based on these considerations, it was concluded that the analytical prediction of receiving space effects on panel sound transmissivity is not feasible in the course of the current KU-FRL research program.

7.1 Significance of Receiving Space Effects

To estimate the sound pressure levels in a space behind a panel the effects of the receiving space on the panel motion and on the distribution of acoustic energy within the space must be considered. The final result desired is a noise reduction value, which will include both the panel transmission loss (TL) and the effects of the receiving space (Ref. 1). This is illustrated in Figure 7.1:

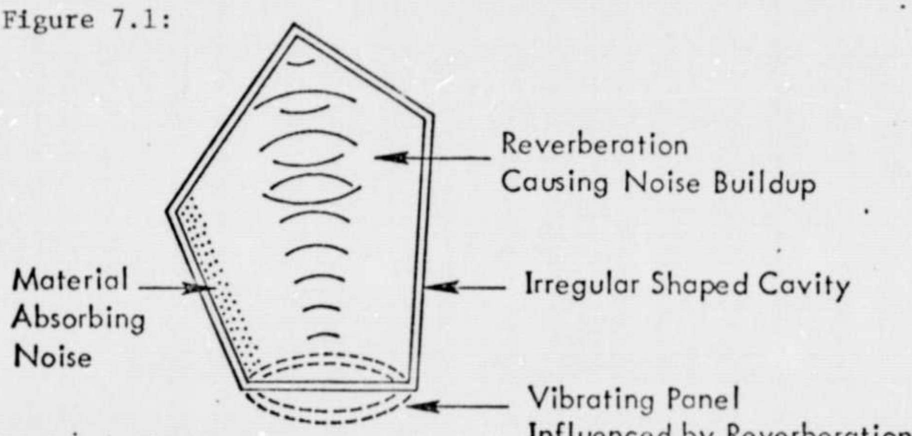

Figure 7.1. Paramaters Influencing Interior Noise

According to Reference 1 the effects of the receiving space can be accounted for by a correction factor for the panel TL which depends in the first place on the relative size of the receiving space. An area defined as a "small receiving space" (relative to the wavelength of sound) behaves essentially as a stiffness, and the acoustic pressure is more or less uniform throughout the space (when the wavelength is greater than six times the typical receiving space dimension - Ref. 1.) In a medium-sized receiving space, discrete resonances with accompanying standing waves will occur. At the maxima in these standing waves the acoustic pressures can build up considerably over those for free-field receiving conditions (which is effectively infinite in extent or perfectly absorptive) while the minima can have sound pressures as low as those for free field conditions. The build-up of standing waves in this frequency region strongly depends on the acoustical absorption as the following table indicates:

Table 7.1: Influence of absorption on difference between maximum and minimum noise levels in a standing wave (Ref. 1)

Absorption	Absorption Coefficient	$SPL_{max} - SPL_{min}$ (db)
High	.4	8 - 30
Medium	.25	15 - 18
low	.13	23 - 30

The average absorption coefficient in a receiving space depends both on the type of surface treatment used, on the fraction of the total surface that is treated, and on the absorbing objects inside. In what Ref. 1 calls a "large" receiving space, the wave length is smaller than one tenth of a typical receiving space dimension. Under these conditions, reasonably diffuse sound fields may be expected. In this frequency region absorption has an appreciable influence on reverberation.

7.2 Analytical Approach

Several analytical techniques have been developed for studying sound transmission into enclosures such as airplane cabins (examples:

Ref. 3, 13, 16). Generally, attempts have been made to solve the wave equation* subject to the appropriate boundary conditions. Although the equation has been known for many years, closed form solutions utilizing the method of separation of variables have been obtained for only a limited number of cases.

These are several reasons for this. When the cavity size is the same as or smaller than the wavelength, a normal mode analysis is appropriate because the wave theory allows for simple expressions at low frequencies, where sound propagation can be accurately described by the lowest mode. Simplicity, however, is lost at higher frequencies where all higher modes must be included. It was suggested that the normal mode analysis is most useful when the wavelength to cavity dimension is between 1/3 and 3 (Ref. 17.). Using this criterion, the normal mode analysis is applicable to the treatment of aircraft cavity acoustics between frequencies of 50 and 500 Hz. For simple enclosures, various techniques have been developed to obtain better accuracy at higher frequencies (for example, the image theory of Ref. 2).

Another reason closed form solutions are difficult is because of sound transmission into enclosures through vibration sensitive surfaces. Since the boundary conditions are not stationary, the classical method of separation of variables cannot be applied and the solution to the acoustic wave equation becomes a difficult task (Ref. 3). In some cases this problem can be simplified by assuming that the boundary panels are nonreacting to the cavity pressures, so that two uncoupled equations need to be solved (first the panel motion due to excitation; then, interior wave equation with panel motion as the boundary condition). In many cases this de-coupling is not possible.

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* Wave Equation: $\nabla^2 p = \frac{1}{c^2}\ddot{p} + Q$

Another frequently encountered problem is in the complexity of the cavity geometry. To solve the analysis problem for such cases, one has to rely on approximate or numerical techniques. One of these methods is the finite element method using an approximate formulation of the wave equation derived from a variational procedures (Lagrange's principle**). Variational procedures are also being used to solve the governing differential equations for the fluid (wave equation) and boundaries (wall motion). An example of this is the Galerkin-type procedure.

7.3 Prediction Receiving Space Effects in General Aviation Aircraft

Typical excitations of general aviation aircraft are mainly in the frequency region where a normal mode analysis could be beneficial (50-500 Hz). However, significant excitations also occur at higher frequencies and as a result the theoretical approach becomes extremely cumbersome (see section 7.2). These difficulties are amplified by the normally irregular shape of general aviation aircraft cabins, as well as the non-uniformity of absorptive materials and the presence of flexible skin panels. Thus, a theoretical approach for general aviation interior noise prediction seems only feasible through the use of finite element computer programs (NASTRAN was successfully used for car interior noise studies. Ref. 4.) Experimental results that will support and validate such theoretical results seem necessary. However, the investigation of receiving space effects in general aviation aircraft is outside the scope of the intended KU-FRL research program.

** Lagrange's Principle: Of all pressure fields satisfying the prescribed dynamic boundary conditions, that which satisfies the constitutive and equilibrium equations and the remaining kinematic boundary conditions, is determined by making the Lagrangian function stationary (Ref. 18).

8. Panel Transmission Loss

Knowledge with respect to the response of structural and non-structural aircraft panels to applied time-varying loads is of importance for the development of theoretical and empirical interior noise analysis procedures as well as for the immediate design and modification of general aviation aircraft. The excitations normally encountered in these aircraft have an aerodynamic, mechanical or acoustic nature, but all occur in the frequency region below 1000 Hz. In this region the noise transmission is governed by panel stiffness (below resonance region), structural damping (resonance region), and surface mass (above region of major resonances).

The KU-FRL noise research team has dedicated some of its time to studying the mechanisms that determine the panel response in these regions. This section summarizes some of the information that was obtained through an extensive literature study.

8.1 Sound Transmission Below Resonance Region

At low frequencies (below panel fundamental frequency), the noise transmission is controlled by panel stiffness and the transmission loss decreases at 6 dB per octave to within the neighborhood of the panel fundamental frequency. The problem of stiffness controlled transmission loss of panels has not been completely explored, but estimating schemes and few experimental results are known. Reference 1 gives the following tentative relation at a frequency $f_1/4$ (f_1 = panel fundamental frequency):

$$TL\ (f_1/4,\ \text{stiffness}) = TL\ (f_1,\ 45°\ \text{mass law}) + 10 \log s^2 + 15$$

where:

s = fraction of surface mass fully participation in panel motion at resonance ($\simeq .2$ in case investigated in Ref. 1)

This relation indicates the requirement for high resonance frequencies to achieve a high Transmission Loss at a given frequency in the stiffness controlled region.

Reference 5 presents the results of an experimental study of the noise attenuation characteristics at low frequencies. It was concluded that for a given panel surface density, as its construction is varied, at any frequency an octave or more below resonance, the noise reduction will increase with an increase in the fundamental frequency. The test results showed a trend as predicted by the equation from Reference 1; however, quantative Transmission Loss values were different (on the average 3-5 dB lower).

If stiffness control is to be used to reduce low frequency transmission of characteristic general aviation sound through panels, resonance frequencies have to be raised substantially. This can be achieved by increasing the panel surface density ($\bar{m}$) or bending stiffness (D). As fundamental frequencies of aluminum panels are generally between 60 and 150 Hz, $\frac{D}{\bar{m}}$ should be increased significantly (for example by a factor of 5-10) to use the stiffness control principle effectively. Such an increase could be obtained through the use of, for example, stiffeners (or: in general orthotropic panels), curvature, honeycomb-type constructions, or different basic plate materials (like filamentary composites).

Equations that give the principal flexural rigidities of orthotropic plates can be found in numerous publications (for example: Refs. 6 and 7). Relations to predict the influence of curvature on the (finite) panel resonances are rarer (Ref. 5). The dynamic behavior of three-ply laminates has been subject of many theoretical studies, but simple relations are known for the frequency region below dilational resonances (Refs. 6, 8). The properties of laminated filamentary composites are still being studied, but few results describing the dynamic behavior are known yet (Ref. 9).

8.2 Sound Transmission in the Region of Panel Resonances

When a simple linear system is excited, the damping and stiffness are the system characteristics which control the response at its resonance frequency. When the same system is excited randomly, the mean square value of the displacement is also dependent on the mass of the system.

Harmonic excitation:

$$\text{resonant amplitude} = \frac{P}{2K\zeta} \qquad \text{(Ref. 8)} \qquad (8.2)$$

Random excitation:

$$\text{r.m.s. value of resonant amplitude} = \frac{\pi PSD_f(\omega r)}{2M^{1/2}K^{3/4}\zeta} \qquad \text{(Ref. 8)} \qquad (8.3)$$

Where: ω_r = resonance frequency

K = system stiffness

ζ = system damping ratio

P = amplitude of harmonic excitation

PSD_f = Power Spectral Density of random excitation

M = system mass

Normally three degrees of damping are specified as follows (Ref. 1):

Table 8.1: Damping Categories

Damping Category	Approximate Damping Factor $\eta = 2\zeta$
Low	.007
Medium	.03
High	.1

Panels to which no damping materials have been applied are expected to fall into the category "low damping". For a panel to have "high damping", it must either be of special construction, or it must be heavily treated with damping material. At the moment, the method of controlling the resonant panel response is to add certain anti-vibration materials to the structure. The most effective materials are those that exhibit both a high damping factor and a high stiffness. Since these materials usually come under the category of plastics, their properties are markedly temperature dependent.

Damping materials added to aircraft panels are in the form of unconstrained or constrained layers. An unconstrained layer has one free surface and it dissipates energy as it undergoes oscillating bending strains due to flexural vibrations. A constrained layer is sandwiched between the basic plate and another stiff layer. The damping layer dissipates energy by virtue of the shear strain when the plate vibrates.

The optimum damping treatment for a vibrating panel depends on properties of the damping material, as well as on the basic plate and excitation characteristics. Equations for optimization of the damping treatment (for a special case like lightweight aircraft structures) can be found in, for example, Reference 8.

The intent of the KU-FRL noise project is to study the sound transmission through aircraft type panels. The application of methods of testing the damping properties of a material is not within the scope of this research program.

8.3 Sound Transmission in the Mass Controlled Region

Panels of finite dimensions behave like infinite panels at frequencies above the range containing the lower normal frequencies (and below the coincidence region). As a result its transmission loss obeys the mass-law which can be stated in approximate forms such as:

$$TL_{400Hz} = 21 + 20 \log \bar{m} \ (45°) \qquad \text{(Ref. 1)} \qquad (8.4)$$

This expression indicates an increase in TL of 6 dB for each doubling of the surface mass, but experiments give an average value of only 4.4 dB (Ref. 10). This and similar relations indicate that damping and stiffness properties are of no significance. Similarly it can be proven that the introduction of curvature of modification into a multilayered panel will have no influence on the TL (provided the surface mass remains constant). Such theoretical predictions have been validated with experimental results.

At high frequencies the transmission loss can be improved (above the mass-law results) by adding absorptive materials with or without a resilient skin. The absorption of porous layer is proportional to its thickness (for given material properties). At high frequencies shear losses due to viscous effects occur when the vibrating air enters and passes through the porous material. For acoustical blankets of normal thickness (up to 4 inches) porous materials are only beneficial at frequencies above approximately 500 Hz. By adding an impermeable membrane to the porous layer, the transmission loss in the lower frequency region can be improved significantly. It is in these groups of materials that many improvements have been reported by manufacturers of sound treatments (see: product brochures and sound magazines).

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