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Surface Pressures and Structural Strains Resulting from Fluctuations in the Turbulent Boundary Layer of a Fairey Delta 2 Aircraft

by

D. R. B. Webb, A. R. Keeler and G. R. Allen

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SURFACE PRESSURES AND STRUCTURAL STRAINS RESULTING FROM FLUCTUATIONS IN THE TURBULENT BOUNDARY LAYER OF A FAIREY DELTA 2 AIRCRAFT

by

D. R. B. Webb
A. R. Keeler*
G. R. Allen

SUMMARY

This paper describes measurements of surface pressure and structural strain due to fluctuations in the turbulent boundary layer of a Fairy Delta 2 aircraft. The measurements were made on the lower surface of the wing at Mach numbers up to 1·6.

The results show that both pressure and strain depend on the kinetic pressure q and are independent of Mach number. It is deduced that the levels of strain due to the pressure fluctuations on the surface of a proposed supersonic transport aircraft would be higher than those obtaining on existing aircraft.

The pressure spectrum was found to be rather flat over the whole of the measured range and only slightly affected by Mach number. The strain spectra on a specially constructed test panel indicated that the major mode of response was of a higher degree than when excited by jet noise, with the panel response being primarily non-resonant.

The overall strain was about one-third of that occurring when the panel was excited by the efflux noise of a stationary jet engine, for the same overall sound pressure level.

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1 INTRODUCTION

With the operation of supersonic aircraft at higher and higher Mach numbers, more needs to be known about the turbulence in the boundary layer. In particular, the possibility that there may be sufficient energy in the boundary layer pressure fluctuations to promote fatigue failure in the aircraft structure warrants investigation. Existing information on the nature of the pressures likely to be encountered on supersonic aircraft is somewhat inadequate[1]. Most of this information is theoretical or from measurements in wind and water tunnels, and there is a notable lack of information from full scale flight.

The purpose of this Note is to record the results of measurements in flight on the wing of a Fairey Delta 2 aircraft. Data are presented on the levels of the pressure fluctuations, and on the strain level and response of a structural test panel which was inserted in the lower surface of the aircraft wing.

The main conclusion is that both pressure and strain depend on the kinetic pressure and are independent of Mach number. In addition, the strain level on the test panel was only a third of that obtained by exciting the panel with jet efflux noise, and the mode of response of the panel of a higher degree than that due to jet noise.

2 AIRCRAFT FLIGHT PLANS

The Fairey Delta 2 was flown up to a Mach number of 1·6 and an E.A.S. at least as great as that likely to be obtained on an M = 2 supersonic transport aircraft. The aircraft was flown on various flight plans, some of which were at constant altitude and increasing E.A.S., and some at a nominally constant E.A.S. of 450 kts, and increasing altitude.

3 INSTRUMENTATION

3.1 Pressure and strain transducers

A block diagram of the pressure and strain measuring circuit is shown in Fig.1, and the positions on the wings at which measurements were taken is shown in Fig.2. On the port wing a Massa piezo-electric-type microphone was used to measure the pressure fluctuations, and a C.E.C. electro-kinetic-type microphone was used on the starboard wing. Both microphones were flexibly supported in order to reduce spurious output signals arising from any vibration of the panel assembly to which they were attached. The stiffness of the support and amount of damping was decided by subjecting the microphones to vibrations having a spectrum similar to that previously measured on the aircraft in flight. The support stiffness and amount of damping were then varied until the output of each microphone was a minimum; further checks were subsequently made in the actual flight environment (see para. 2.4). Each microphone was mounted flush with the outside of a specially constructed panel containing the necessary power supplies, which was inserted into the wing as a unit. Photographs of one of the panel installations are shown in Fig.3. Part of this panel, which was inserted in the port wing, was reduced by chemical etching in thickness to 0·018 in. over an area of 8 in. × 6 in. to form a test panel, to which were attached two barium titanate strain gauges, one at the centre of the test panel and one at the edge, and with their axes parallel to the wing chord. These were intended primarily to give information on the frequencies of the modes of deformation of the test panel and on the spectrum of strain.

3.2 Recording of measurements

The signals from the transducers were transmitted to the ground by means of an amplitude-modulated V.H.F. telemetry system and the signals received were recorded on magnetic tape. The system was calibrated to cover a frequency range of 10 c/s - 10 Kc/s, although signals below 100 c/s were discarded because of aircraft vibration. The upper frequency was limited by the frequency response of the telemetry system and magnetic tape recorder. The transmitting system contained a high quality amplitude-controlled oscillator so that compensations could be made for changes in signal strength due to radiation directional effects from the aircraft in flight.

3.3 Record analysis

The pressure signal recordings were analysed in two ways. Overall pressure levels were measured on the meter of a Bruel and Kjaer Audio Frequency Spectrum Analyser, whilst pressure spectra were obtained by means of the S.P.A.D.A. System of Analysis used by Guided Weapons Department, R.A.E. Overall strain levels and spectra were obtained by means of the Bruel and Kjaer Analyser.

3.4 Effects of vibration on measuring equipment

At an early stage in the experiment, both microphone diaphragms were blanked off so that any signals received were due to vibration of the microphone, leads, pre-amplifiers and telemetry transmitter. The aircraft was flown to a particular flight plan which was repeated with the microphone diaphragms unblanked. For both microphones, it was found that the overall signal received when blanked off was approximately 40 db less than the unblanked signal, so it was concluded that signals from the microphones when they were being used to measure pressure were not significantly affected by signals arising from microphone vibration.

The sides of the structural test panel were heavily boxed in with deep members which were made very stiff compared with the adjoining wing structure, in order to minimise the transmission of static or dynamic stress from the wing through the attachment points of the panel. The inside surfaces of the box containing the test panel were lined with $\frac{1}{2}$ in. glass wool in order to reduce any flanking noise transmitted through the sides of the box containing the test panel.

4 GENERAL QUALITY OF RESULTS

Prior to the experiment both microphones were calibrated in an anechoic chamber by comparing them with a Bruel and Kjaer microphone which had previously been calibrated by means of an electro-static actuator. During the flight experiments, however, an appreciable discrepancy was noticed between the overall sound pressure levels from the two microphones, and as the experiment continued this discrepancy grew to 8 db. Examination of the spectra indicated by the Massa microphone (port wing) showed a phenomenon which is not yet understood. During the take-off and initial climb of the aircraft, both microphones indicated similar levels and spectra, but at about 8000 ft a high frequency content of about 8 Kc/s became apparent on the spectra from the Massa microphone which increased to such an extent that at 30,000 ft it dominated the spectrum; since this effect persisted irrespective of aircraft usage, the results from the Massa microphone have been discarded. The results from the C.E.C. microphone were, in general, satisfactory although it was found that for the last three flights, the microphone sensitivity at the higher frequencies was slightly reduced and compensation for this effect was necessary.

The results from the barium titanate strain gauges were very satisfactory. They were calibrated by comparison with wire resistance strain gauges, which were attached to the same part of the test panel, but on the opposite side of the skin. During the experiment the barium titanate gauges gave fairly good repeatability, and there was no measurable change in the overall sensitivity and frequency response when their calibration was checked in the laboratory at the end of the experiment. Before and after each flight, checks were made on the sensitivities of each transducer by exciting the microphones and test panel with a measured white noise pressure spectrum generated by an electro-magnetic acoustic pressure unit. This, together with the amplitude-controlled oscillator provided a good check on the overall measuring system calibration, so that the records on the tape were probably accurate to within $1\frac{1}{2}$ db. The results shown in this Note however are unlikely to be more accurate than $\pm1\frac{1}{2}$ db, because there are further inaccuracies in reading the tape records and correlating them with measurements of E.A.S.

5 RESULTS

5.1 Boundary layer thickness measurements

The boundary layer thickness was measured adjacent to the point at which the pressure measurement was made.

These measurements indicated that the boundary layer thickness was approximately 0.7 in., and the displacement thickness, δ^*, has been taken to be one seventh of this. The boundary layer thickness did not change significantly during the various flights flown by the aircraft.

5.2 Overall levels

The main results are shown in Figs.4 and 5, which show the relationship between overall sound pressure level and kinetic pressure q, and between overall strain levels and q, points having the same symbol being obtained on the same flight. It can be seen that in both cases, bearing in mind the accuracy of the measurements, a straight line can be drawn showing the trend. In Fig.4, the straight line corresponds to a variation of overall pressure with q, given by $(20 \log q + 76)$ db whilst in Fig.5 the straight line corresponding to a variation of overall strain with q is given by $(14 \log q - 138)$. The form of these expressions gives no indication whether there is any effect due to Mach number, and so, in Fig.6, the results shown in Fig.4 are replotted as a fraction of q against Mach number, and it can be seen that if there is any Mach number effect, it is masked by the scatter, which is shown up by the linear scale to be considerable. A better guide to the effect of Mach number can be obtained by isolating the E.A.S. parameter from Mach number, as in the case of a flight where the aircraft is made to climb at constant E.A.S. and increasing Mach number. The results of such a flight are given in Fig.7, which shows overall sound pressure level and strain levels for a range from M = 1 to 1·5 at a nominally constant E.A.S. of 450 knots. Previous to this experiment, detailed measurements of the flow conditions on the wing surface were made, and included the test panel and microphone area. These measurements indicated that in this area, the local Mach number was near the free stream Mach number.

5.3 Pressure and strain spectra

In Fig.8(a) the spectral density of the pressure fluctuations is shown for three different Mach numbers. The non-dimensional pressure and frequency parameters have been chosen for comparison with similar presentations of spectral density in reports on similar work by other authors, some of which are shown in Fig.8(b), which has been reproduced from Ref.1. The frequency scale however can be misleading when discussing the spectral shape in relation to structural response because of the change in the value of $f\delta^*/U$ with changes in U. Accordingly, the spectrum level is shown plotted against f only in Fig.8(c). In Fig.9, the spectrum of strain at the centre of the panel is shown for the three values of Mach number. The strain is plotted to a logarithmic scale and is quoted in spectrum level. The overall strains are quoted, and for the material used in the panel (D.T.D.606), represent overall stresses of 161, 128 and 113 lb/in^2 for Mach numbers of 0·96, 1·2 and 1·45 respectively. An interesting feature of Fig.9 is the peak in the spectra at about 380 c/s. The fundamental frequency of the panel when excited by sine waves at normal incidence can be seen from Fig.11(a) to be 200 c/s approximately. When the panel is excited by waves at grazing incidence - two further modes could be excited, with some difficulty, at 332 c/s and 390 c/s so that the response at 380 c/s is probably associated with one or both of these modes.

5.4 Comparison between jet noise and boundary layer excitation

In Fig.10 a comparison is made between the response of the panel to noise from the efflux of a stationary jet engine and boundary layer noise having the same overall level. This comparison is made, because, in attempting to discuss fatigue of the panel, what experience there is on structures affected by noise, is largely confined to cases of jet noise excitation. There are two main features of interest. Firstly, the overall rms strain response is five times greater for jet noise than for boundary layer noise, with an even bigger difference between the maximum peaks. Secondly, the major response of the jet noise excited panel is at 200 c/s, which can be identified with the first mode at 205 c/s in Fig.11(a). In order that the difference between the mechanism of forcing of boundary layer noise and jet noise can be discussed, a comparison is made in Fig.12 between the measured overall strain when the panel is excited by jet noise, and that which has been derived by multiplying the response of the panel to sine waves at normal incidence by the ratio between the sine wave pressure and the jet noise spectrum level. An allowance for non-linearity in the variation of strain with pressure was allowed in the estimates. The panel was excited with sound waves, at normal incidence, and the strain noted when the pressure was varied incrementally. This indicated that over the range of 90 db - 122 db., the strain was proportional to the 0·8 power of the pressure. It can be seen that there is good agreement between the mean of the curves, the flatness of the measured curve resulting from the one third octave band-width of the measuring analyser in contrast with the response of the panel to discrete frequency sine waves. The same process has been used in Fig.13, but in this case two derived curves are shown, one being based on normal incidence as before, and the other on grazing incidence. The significance of these results is discussed in para. 6.2., below.

Fig.14 shows a comparison of the strain levels of each of the two strain gauges on the test panel for a selected flight. During each flight the ratio of the two strains was as shown in this figure, consequently only one strain gauge result has been quoted in the other appropriate figures in order to make them as clear as possible.

6 DISCUSSION

6.1 Sound pressure levels and spectra

The main results of this experiment which are shown in Figs.4 and 5 seem to indicate that, notwithstanding the scatter on the results, both pressure and strain can be simply related to the kinetic pressure q by a simple law, for a given boundary layer thickness. Whereas there would appear to be no obvious reason why this should be so, this law indicating a dependence upon q can be used as a general guide to the levels obtaining on the surface structure in the presence of a turbulent boundary layer. The straight line shown in Fig.4, shows the pressures to be directly proportional to q which is similar to the law, $\sqrt{\overline{p^2}} = 0{\cdot}006\ q$ found by Willmarth[1,2]. The overall sound pressure levels however, are lower in value than those quoted by Willmarth, but it must be remembered that an 'overall' level is associated with the bandwidth of the measuring apparatus. In Fig.8(b), curves are shown of pressure spectra found by various workers, which indicate that for displacement thicknesses (δ^*) applicable to the Fairey Delta 2, the spectrum level is unlikely to fall away significantly until at least 26 Kc/s, whereas the upper limit used in this experiment was only 10 Kc/s. If therefore, the spectra shown in Fig.8(a) were extended to 26 Kc/s with the same slope and integrated, they would yield a similar overall level to those of Willmarth. That is, the root mean square pressure $\sqrt{\overline{p^2}}$ would be virtually equal to $0{\cdot}006\ q$, rather than $0{\cdot}0026\ q$ as found. This is further confirmed in Fig.8(a) where the level of spectral density is shown to be similar to the appropriate curve in Fig.8(b). No dependence on Mach number of course is indicated by Willmarth's result, and no very distinct relationship is indicated by the result of the Fairey Delta 2 experiment. The scatter of results was considerable, and the plot of sound pressure level against Mach number (Fig.6) is of little assistance in determining any dependence on Mach number. The accuracy of the pressure measurements for any one flight however was much better than between flights, (i.e. better than ±1·5 db) and the results shown for the constant E.A.S. flight of Fig.7 indicates no sensible variation with Mach number. Fig.17 is only one of a number of flights of different flight paths, and although not reproduced in this Note these showed similar relationships to those of Fig.7.

It is possible of course that boundary layer thickness could have an effect upon pressure levels and spectrum, which in turn might influence the response of a structure. In this experiment, the boundary layer thickness was substantially constant for all flight conditions, so that the law relating sound pressure level with q which has been quoted may only apply for a boundary layer thickness of the order of those measured on the Fairey Delta 2.

The pressure spectra in Fig.8(a) show fluctuations about a mean level which is almost constant over the measured frequency range, with a slight increase in high frequency content at the higher Mach numbers. If Fig.8(a) is compared with Fig.8(b) it will be seen to fit in with the lower frequency values and confirm the tendency shown in Fig.8(b) for the spectra to be flat down to very low frequencies.

6.2 Test panel response - strain

Considering the overall strain response to boundary layer noise indicated in Fig.5, it can be seen that the exponent of the variation of strain with q is lower than that for the variation of pressure in Fig.4. It can be seen from Fig.10 that the overall response of the panel is considerably less for the same overall sound pressure level than when excited by jet noise, indicating that the panel is being excited less effectively. In addition, Fig.10 shows considerable differences between the strain spectra, indicating differences in the modes of vibration of the test panel. Further support for this is shown in Figs.12 and 13, which indicate that the mechanism of forcing in the jet noise case may be similar to the case when the panel is excited by plane waves, of normal incidence, and, if Fig.12 is typical, it appears that the overall response of a panel to jet noise can be estimated with fair accuracy if the response to plane sine waves at normal incidence is known. In the boundary layer noise case however, in Fig.13, the assumption of normal incidence yields poor agreement with the measured curve, whilst for grazing incidence although there is still poor agreement between overall level, there is some indication that similar overtones can be excited.

The two small peaks at 330 c/s and 400 c/s approx. shown in Fig.11(b) are the two modes shown in Fig.15(b) and (c) respectively, and in the laboratory could only be excited by grazing incidence waves. Fig.11(b) shows them to be quite small peaks but they are in fact major panel modes, which are excited inefficiently by grazing incidence waves generated by a loudspeaker, because of the poor acceptance by the test panel of wavelengths which are long in relation to the panel dimensions. More efficient excitation of the modes by better matching grazing impulses would result in a more significant response.

Comparing the two spectra of Figs.9 and 11(b) it is evident that the major peak at 380 c/s approx. is associated more with the 400 c/s mode shown in Fig.15(c) than the 330 c/s mode shown in 15(b). Further evidence to support this can be found from two sources. Firstly, when the first four modes at 200 c/s, 332 c/s, 400 c/s and 532 c/s were acoustically excited in the laboratory it was found that in the case of the 400 c/s mode, the ratios of the strain from the two strain gauges were exactly as indicated in Fig.14, the ratios of strains in the other three modes being very dissimilar. Secondly, when the panel assembly was installed in the aircraft wing, the long side of the test panel ran spanwise so that in the case of the 332 c/s mode, the nodal line shown in Fig.15(b) ran chordwise. In the case of the 332 c/s mode, because one side of the nodal line is in anti-phase with the other, a certain degree of this type of pressure correlation might be required in a spanwise direction, that is across the direction of flow. In the 400 c/s case, the same type of pressure correlation would be required to excite the mode to the same extent, but correlation would be required in the direction of flow. Presumably, over the comparatively short length of the test panel it would be unlikely that there would be any appreciable convection of the pressure pulsations across the stream, although of course to be definite about this, measurements of pressure correlation in these directions would be required.

It is almost certain therefore that the structural test panel was excited in a higher degree mode by boundary layer noise than by jet noise, and because this mode can only be excited by grazing incidence, it is likely that the convection of the pressure pulsations has a more significant forcing effect than the pressure pulsations themselves.

7 FUTURE WORK ON BOUNDARY LAYER NOISE

The scope of the instrumentation used for this experiment was limited in order that information could quickly be obtained about the effect of Mach number on boundary layer noise. A much more extensive experiment would be required however to obtain a fuller understanding of the nature of boundary layer noise and its effect on aircraft structures, and it is suggested that any further investigations should cover the following points:-

(a) The effect of boundary layer thickness upon sound pressure level and spectrum should be investigated by taking measurements at various positions on an aircraft such as at the nose, leading and trailing edge of wing, and rear fuselage.

(b) Two rectangular test panels orientated one with its longer sides chordwise and the other spanwise should be installed in the aircraft so that the effects of chordwise and spanwise correlations could be studied. They should carry a sufficient number of strain gauges (or possibly proximity transducers), to enable their principal modes of vibration to be determined.

(c) A greater number of pressure transducers should be used, positioned in streamwise and lateral directions so that the pressure correlation could be studied in relation to the response of the test panels.

(d) An aircraft capable of a Mach number of at least 2 should be used in order to explore further the relationship between sound pressure level and Mach number.

(e) Measurements of strain should be made on a test panel similar to that used in this test, installed in the rear fuselage of a suitable commercial jet aircraft, such as a Comet, in order to ascertain whether the strain levels and modes of vibration excited by the jet exhaust noise in flight, could be used as a guide to the behaviour of the structures of supersonic transport aircraft excited by boundary layer noise.

8 CONCLUSIONS

The levels of pressure measured on the lower surface of the wing of a Fairey Delta 2 aircraft up to an E.A.S. of 500 knots and M = 1·5 are found to be directly proportional to q and independent of Mach number. By extending the results from this experiment over a likely bandwidth the overall sound pressure level can be estimated by the expression $\sqrt{\overline{p^2}} = 0{\cdot}006\ q$. Since proposed supersonic transport aircraft are likely to cruise at a higher E.A.S. than existing commercial aircraft, extrapolation from the results of this experiment indicate that the pressures on the structure due to boundary layer noise will be higher, unless there is a significant alleviation at Mach numbers above 1·5.

The pressure spectrum is rather flat, and covers a wide frequency range. This may be significant in relation to the propagation of a crack caused by other loading actions, because the energy available to excite a small piece of material adjacent to the crack may be just as great as that available for excitation of lower local structural modes.

The instrumentation for this experiment was insufficient to be certain of the type of movement of the test panel, but it is thought that the greatest component of response was due to the third mode with the remainder of the movement being forced response. The general level of response was approximately 10 db less than when excited by jet noise having the same overall level.

ACKNOWLEDGEMENTS

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LIST OF SYMBOLS

f	frequency
$\sqrt{\overline{p^2}}$	root mean square pressure
q	kinetic pressure
$P(f)$	spectral density of $\overline{p^2}$
U	local air velocity
δ^*	boundary layer displacement thickness (In this Note taken as 1/7th of the measured boundary layer thickness)
M	Mach number

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No.	Author(s)	Title, etc
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2	Willmarth, W.W.	Wall pressure fluctuations in a turbulent boundary layer. N.A.C.A. T.N.4139, March, 1958.
3		The supersonic transport - a technical summary. By Staff at the Langley Research Center. N.A.S.A. Technical Note D-423, June 1960.

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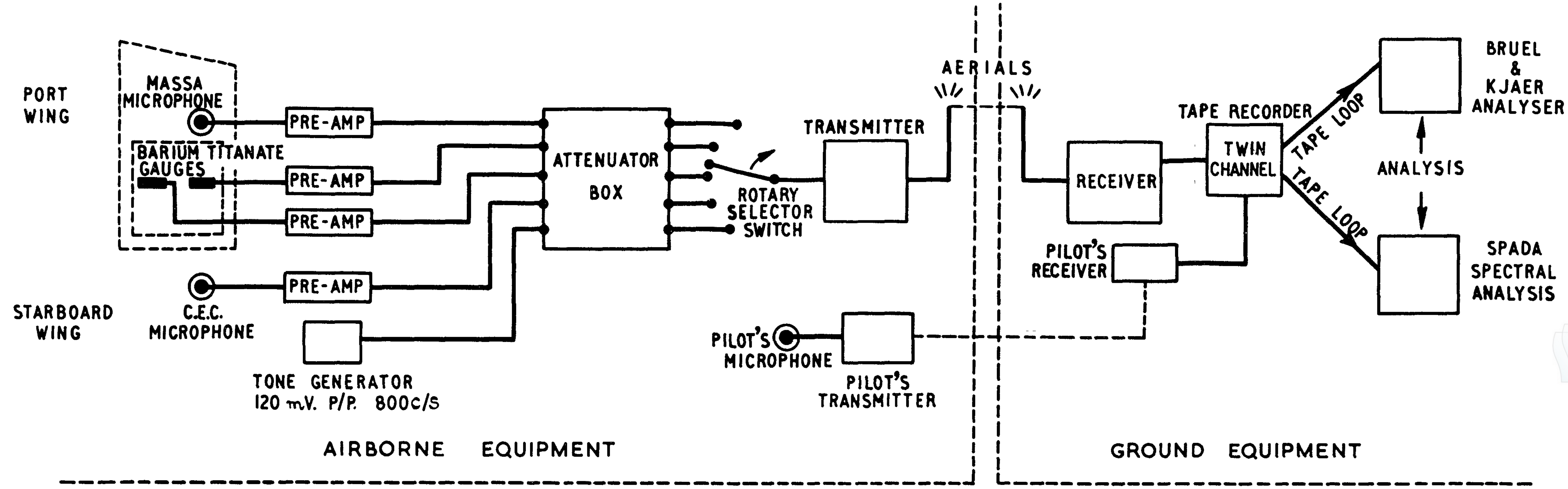


MASSA MICROPHONE — TYPE 211 PIEZO-ELECTRIC : PRE-AMPLIFIER, SOUTHERN INSTRUMENTS LTD. TYPE M 1201

STRAIN GAUGES — G.E.C. BARIUM TITANATE TYPE SP 1101 : PRE-AMPLIFIER, J. LANGHAM THOMPSON LTD. TYPE CF 1

C.E.C. MICROPHONE — ELECTRO-KINETIC TYPE 4-340 (CONSOLIDATED ELECTRO-DYNAMICS CORPORATION)

TRANSMITTER — TYPE X 7887

TRANSMITTER AERIAL — WHIP, TYPE 228

RECEIVER — TYPE X 11610

RECEIVER AERIAL — TYPE 220 A

TAPE RECORDER — WRIGHT & WEARE FEROGRAPH TWIN CHANNEL

BRUEL & KJAER AUDIO FREQUENCY SPECTRUM ANALYSER TYPE 3310 ($\frac{1}{3}$ OCTAVE BANDWIDTH)

SPADA SPECTRAL ANALYSIS — CONTINUOUS LOOP

10 c/s — 100 c/s	$12\frac{1}{2}\%$ BANDWIDTH	}	CORRECTED TO 1 c/s BANDWIDTH
100 c/s — 1 Kc/s	$12\frac{1}{2}\%$ BANDWIDTH	}	
1 Kc/s — 10 Kc/s	6 % BANDWIDTH	}	

FIG. 1. BLOCK DIAGRAM OF LAYOUT FOR MEASURING PRESSURES AND STRAINS.

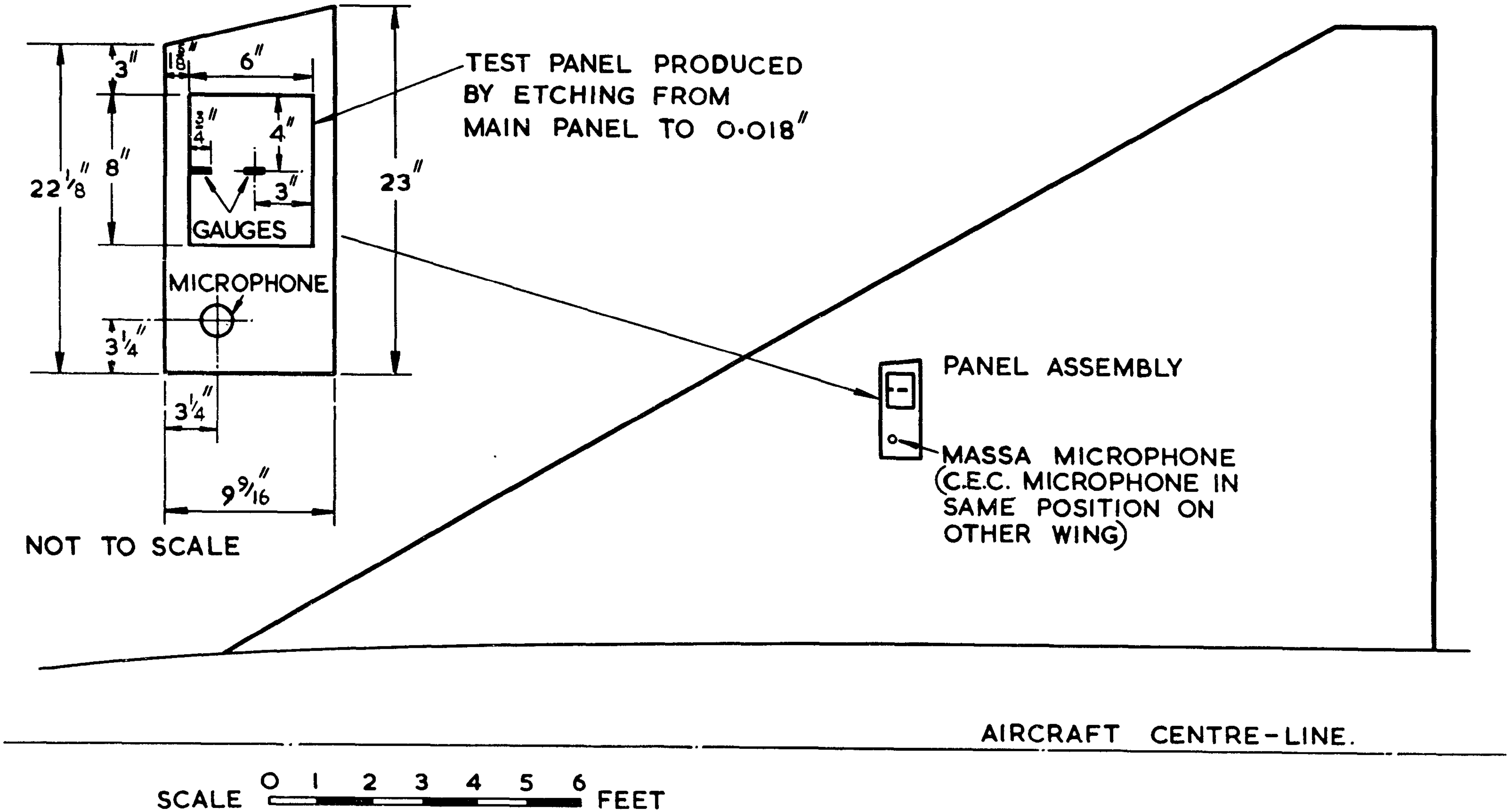


FIG.2. POSITIONS OF MICROPHONES AND STRAIN GAUGES ON LOWER SURFACES OF WINGS.

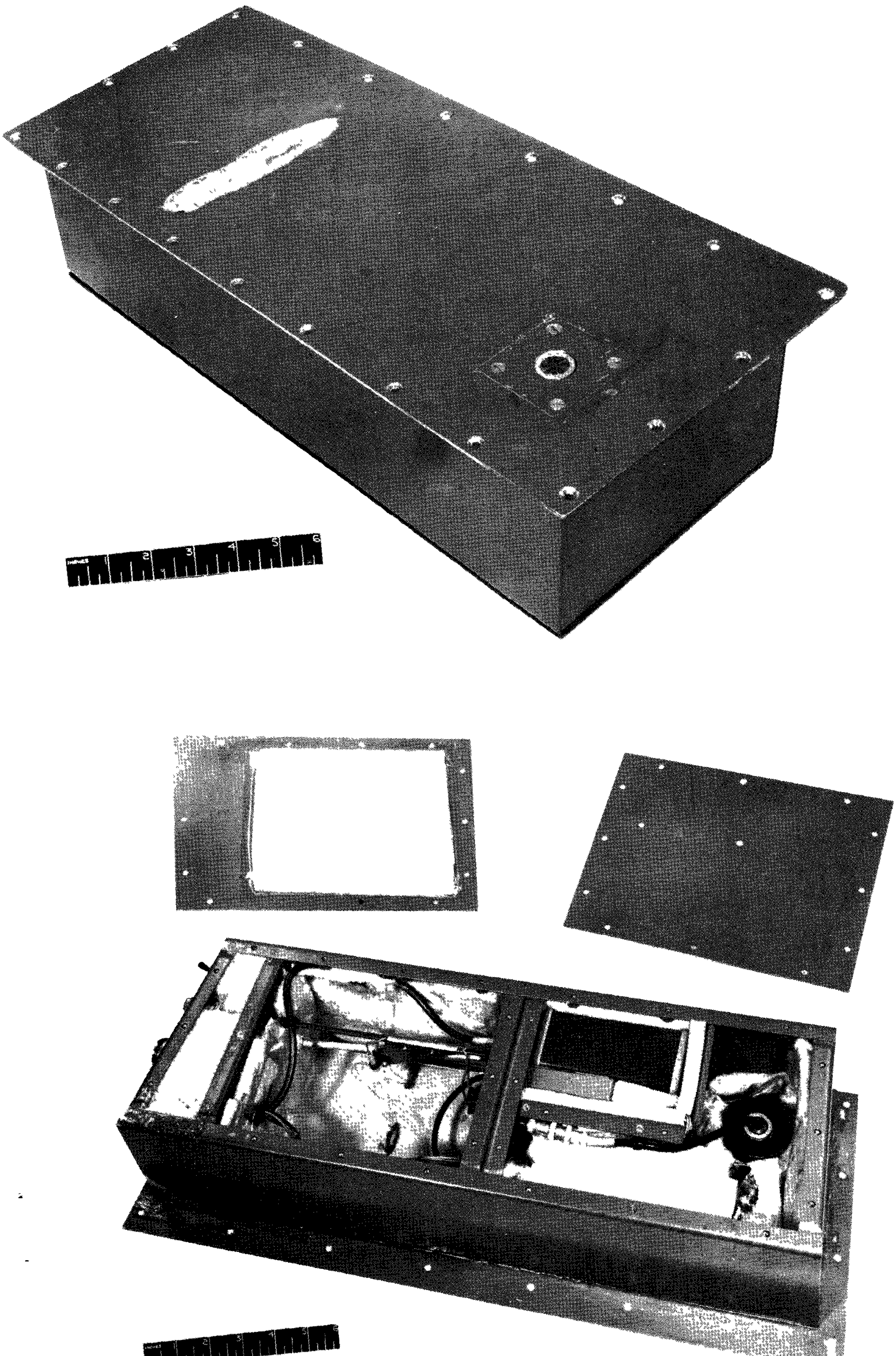


FIG.3. PORT WING MICROPHONE AND STRAIN GAUGE INSTALLATION

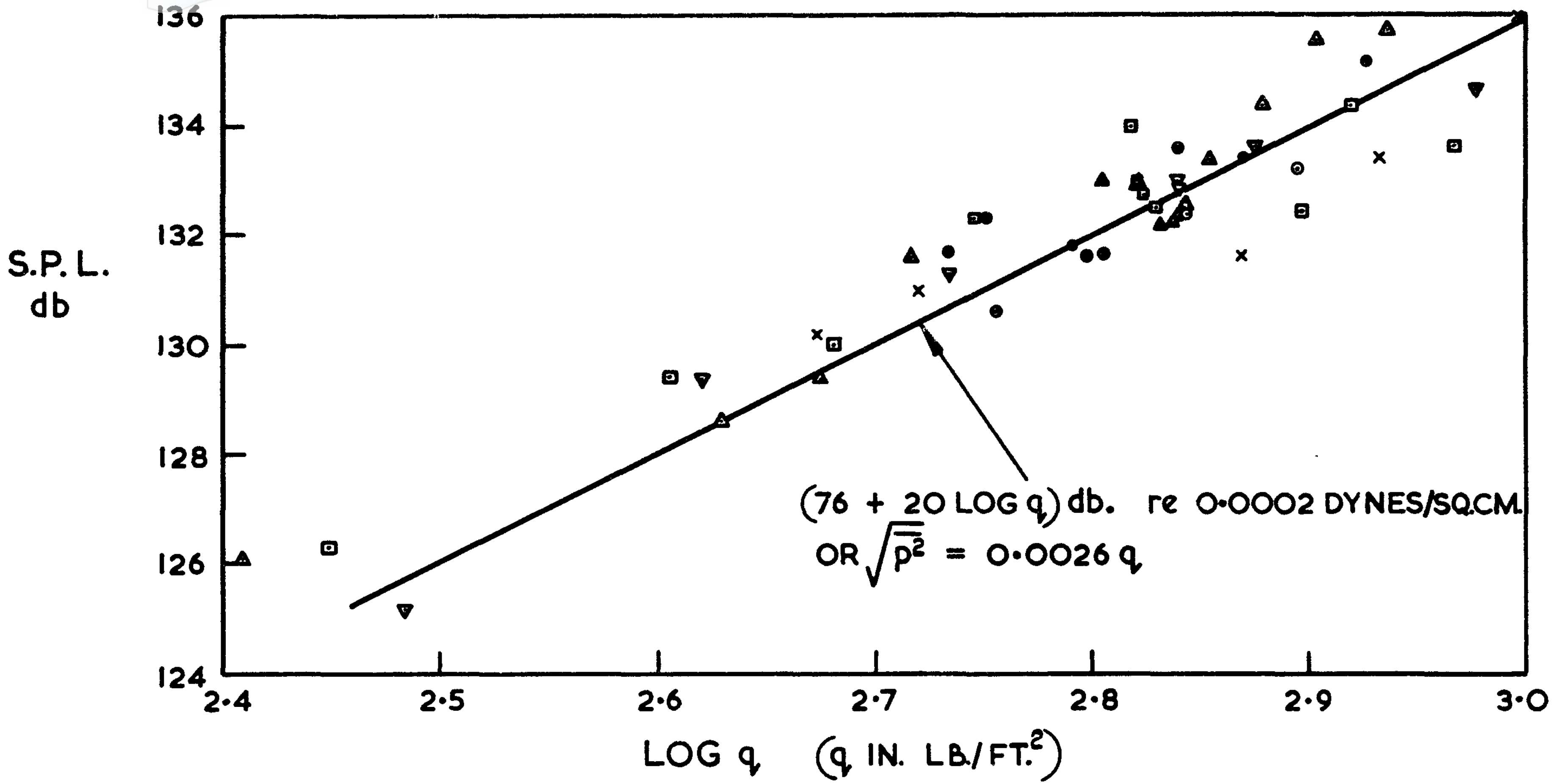


FIG.4. OVERALL PRESSURE LEVELS ON WING UNDER-SURFACE AS A FUNCTION OF KINETIC PRESSURE.

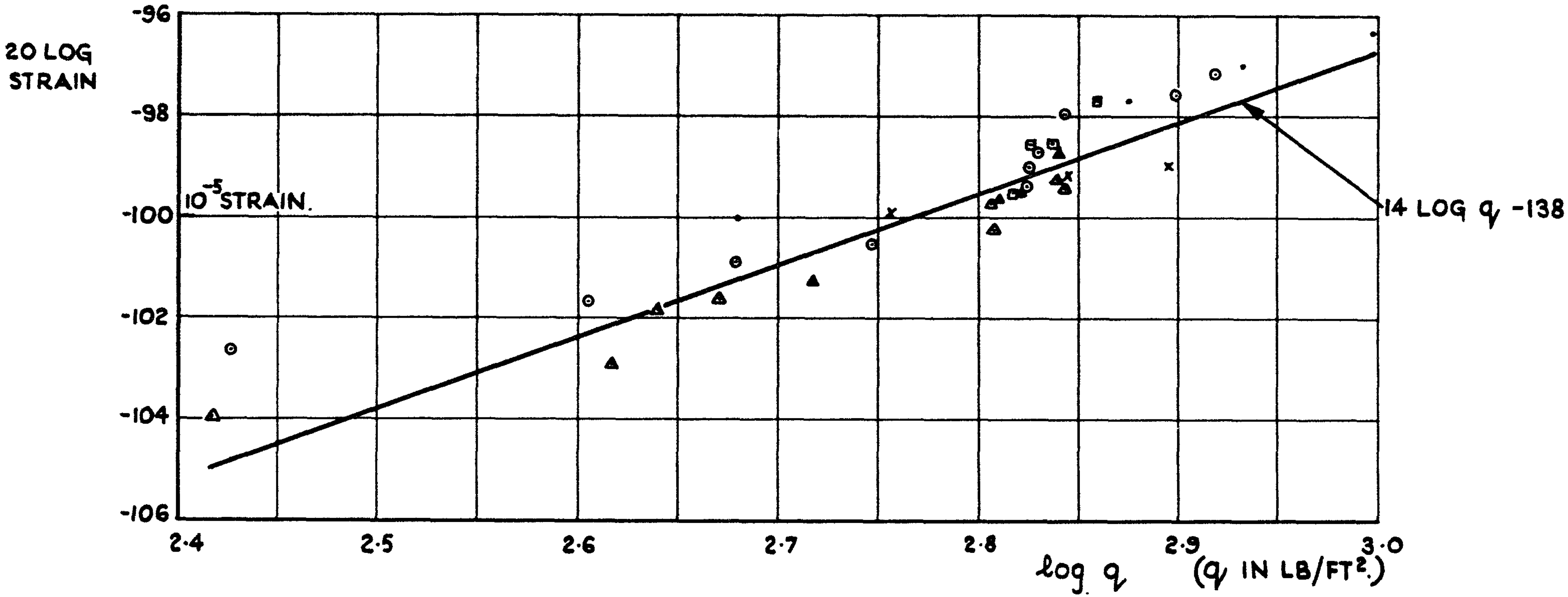


FIG. 5. OVERALL STRAIN LEVELS AT CENTRE OF TEST PANEL AS A FUNCTION OF KINETIC PRESSURE.

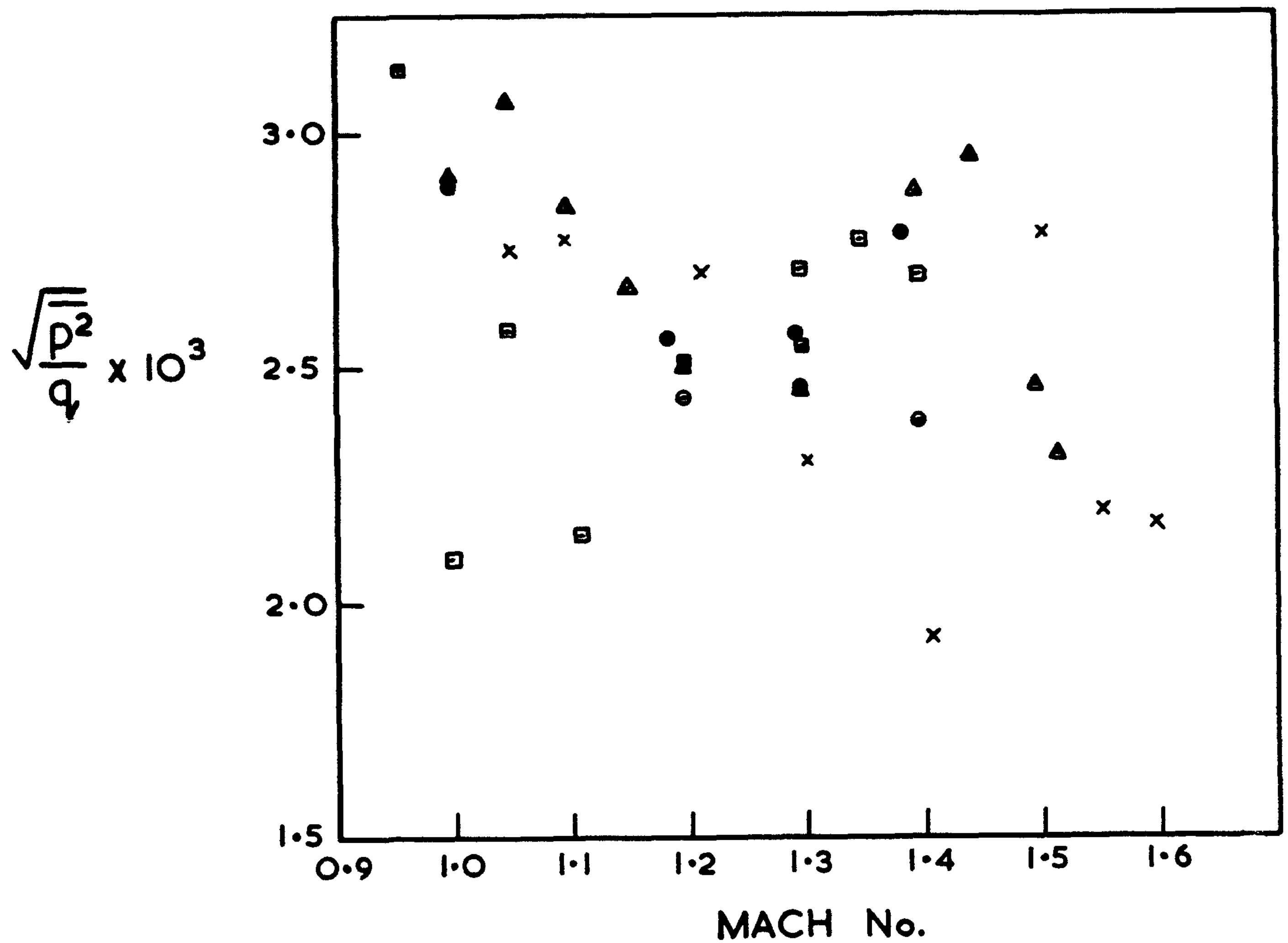


FIG. 6. SOUND PRESSURE LEVEL AS A FUNCTION OF MACH No.

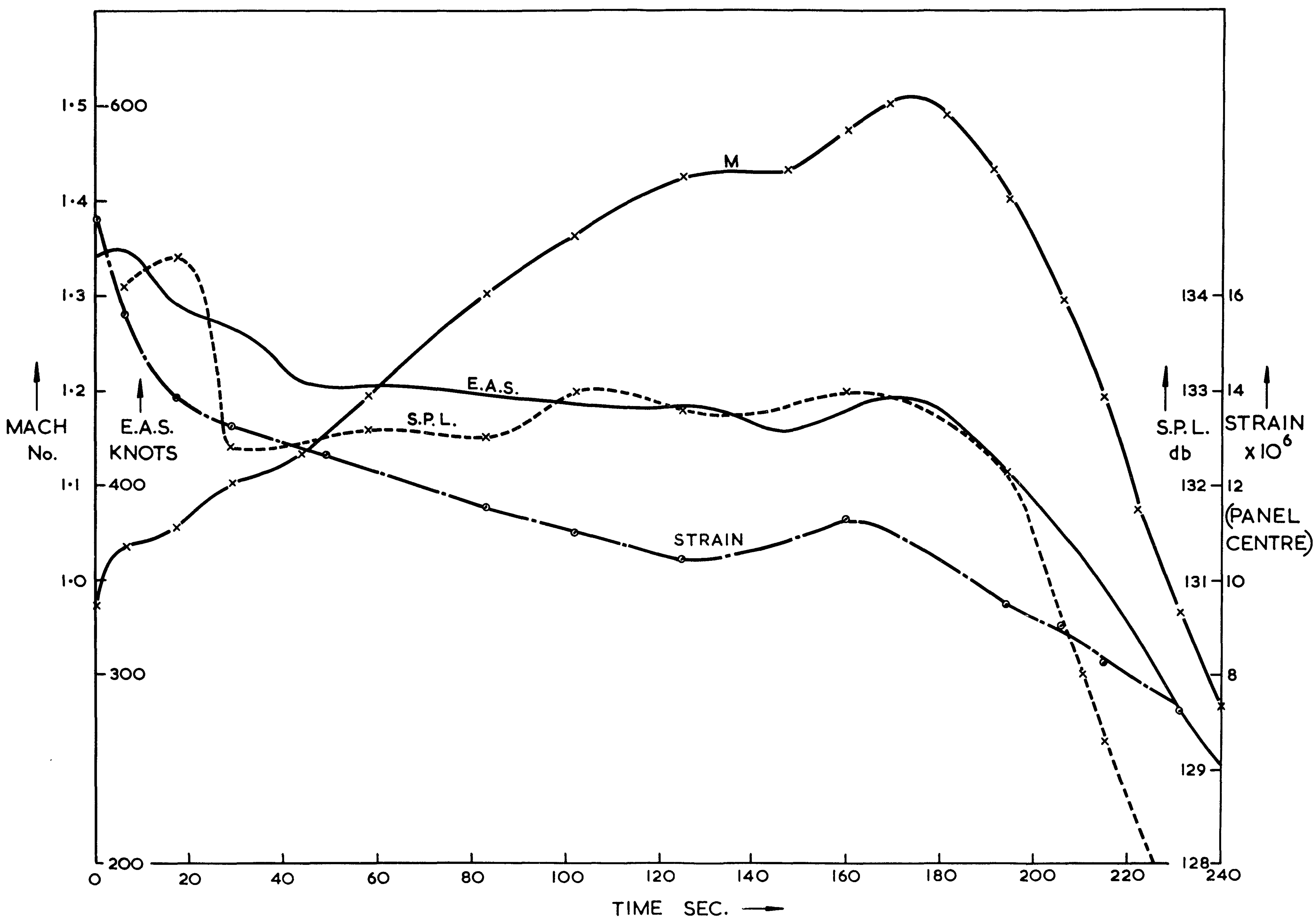


FIG. 7. PRESSURE AND STRAIN FOR A FLIGHT OF NOMINALLY CONSTANT E. A. S.

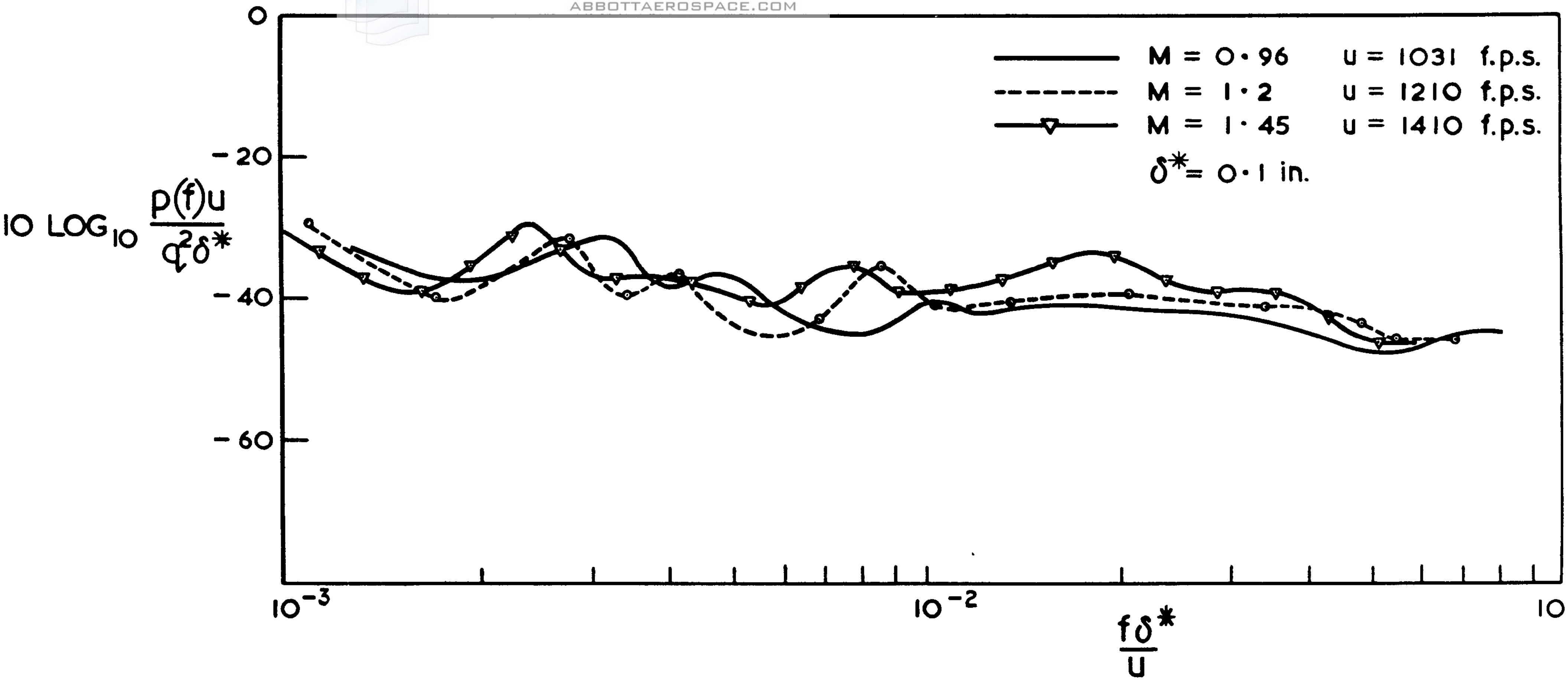


FIG. 8. (a) SPECTRAL DENSITY OF PRESSURE FLUCTUATIONS ON UNDERNEATH SURFACE OF F. D. 2. STARBOARD WING.

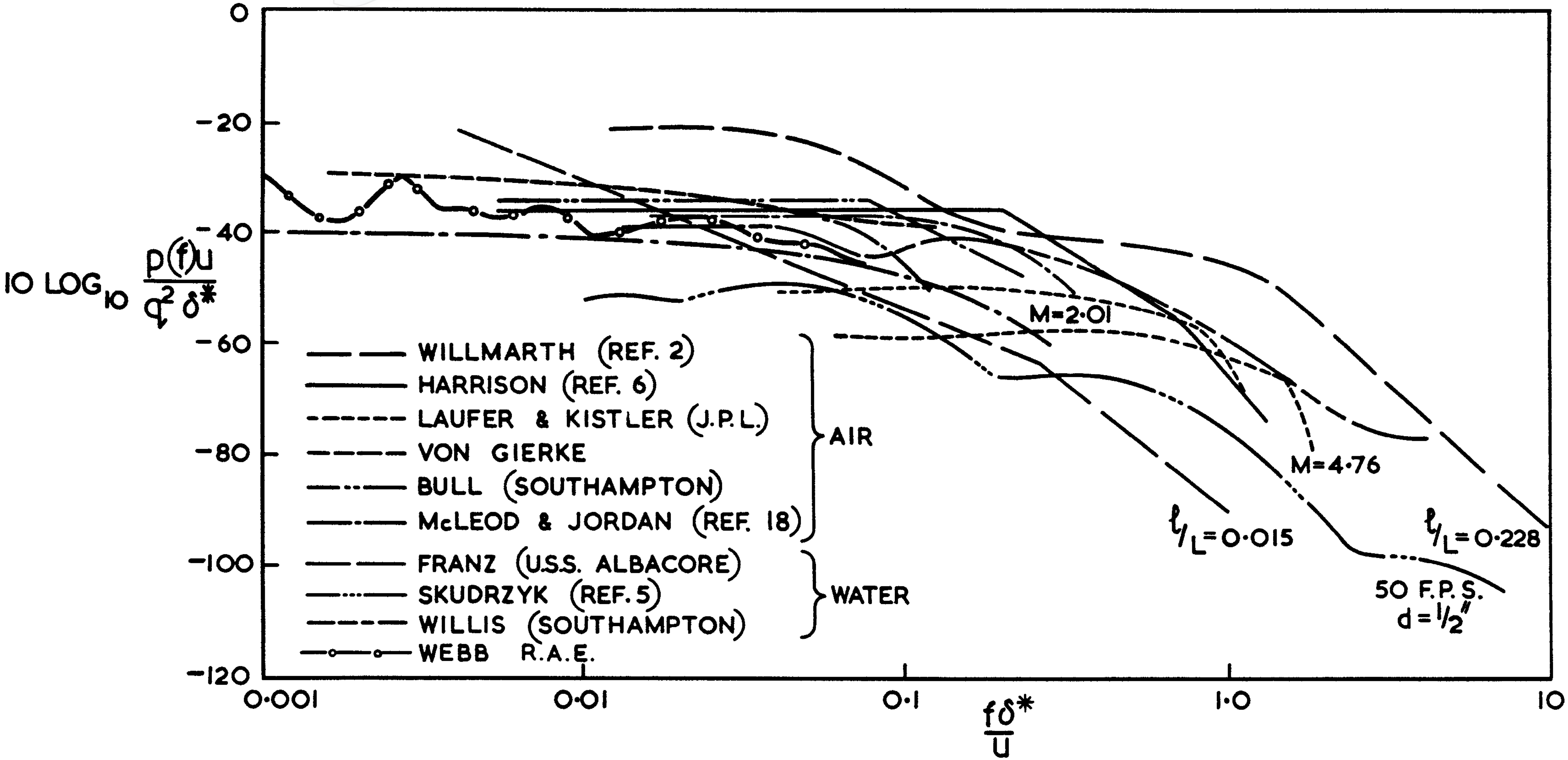


FIG. 8.(b) COMPARISION OF FREQUENCY SPECTRA
(REPRODUCED FROM REF 1.)

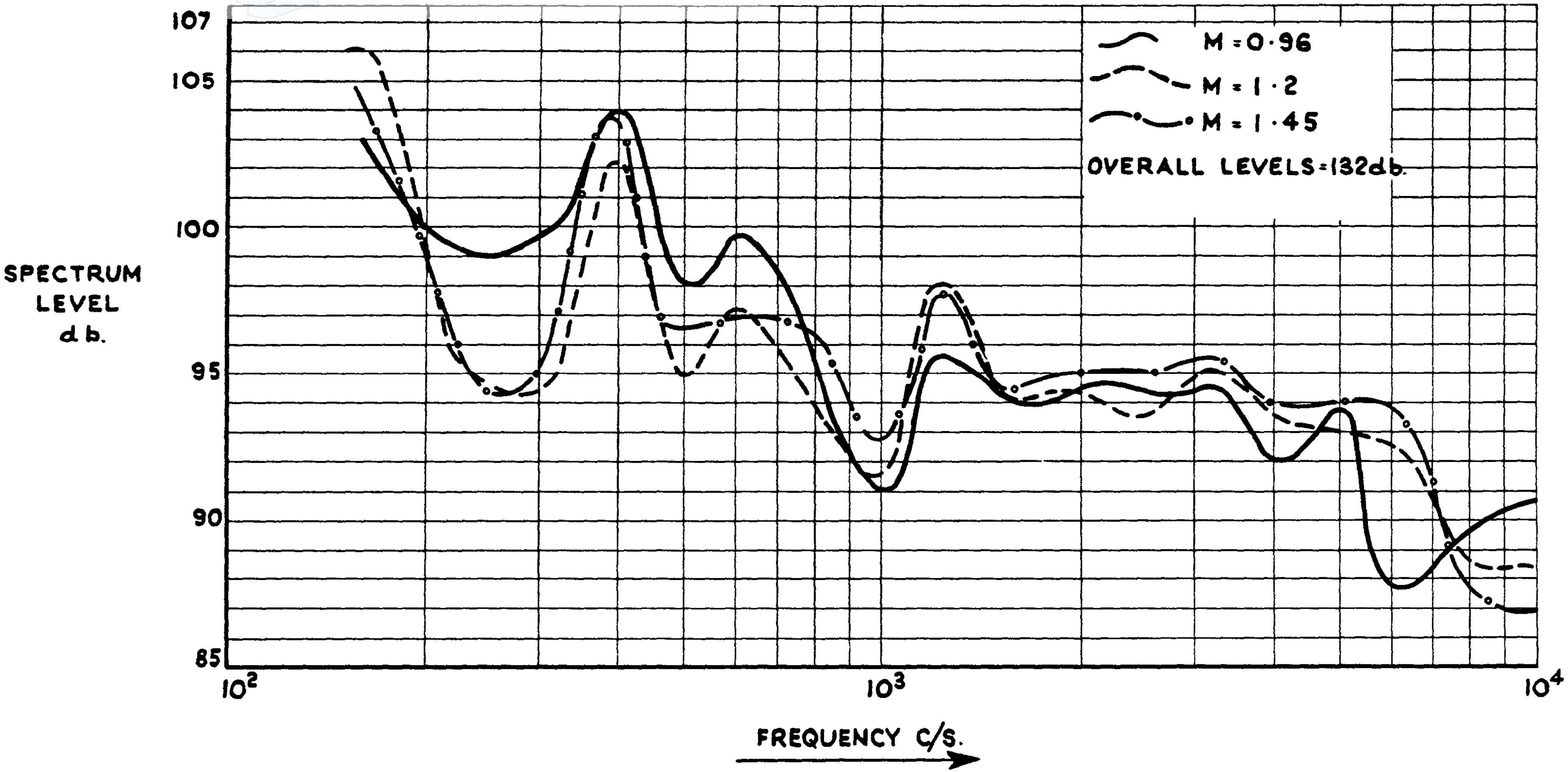


FIG. 8.(c) PRESSURE SPECTRA ON UNDERNEATH SURFACE OF F.D.2. STARBOARD WING.

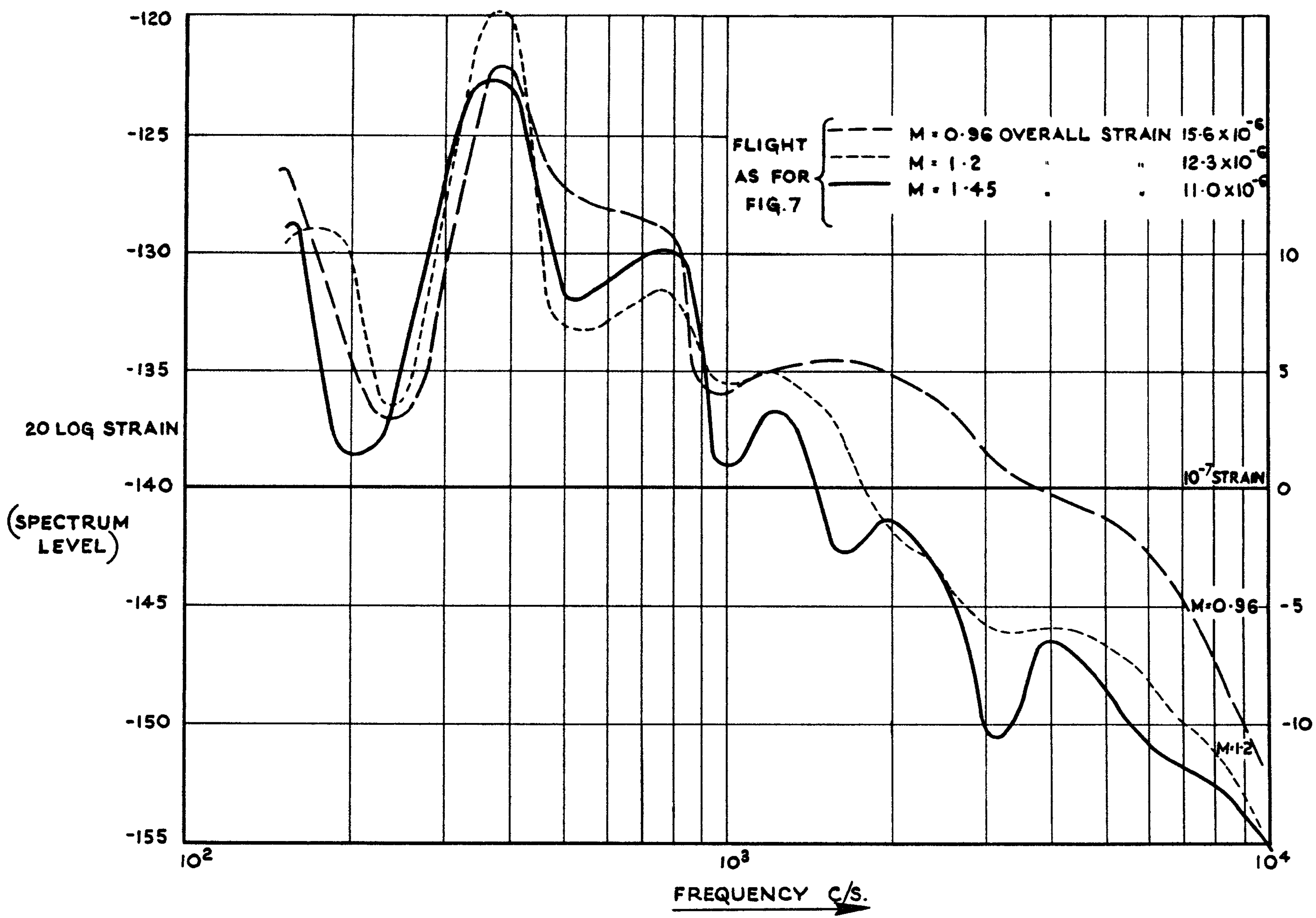


FIG. 9. SPECTRA OF STRAIN AT CENTRE OF TEST PANEL.

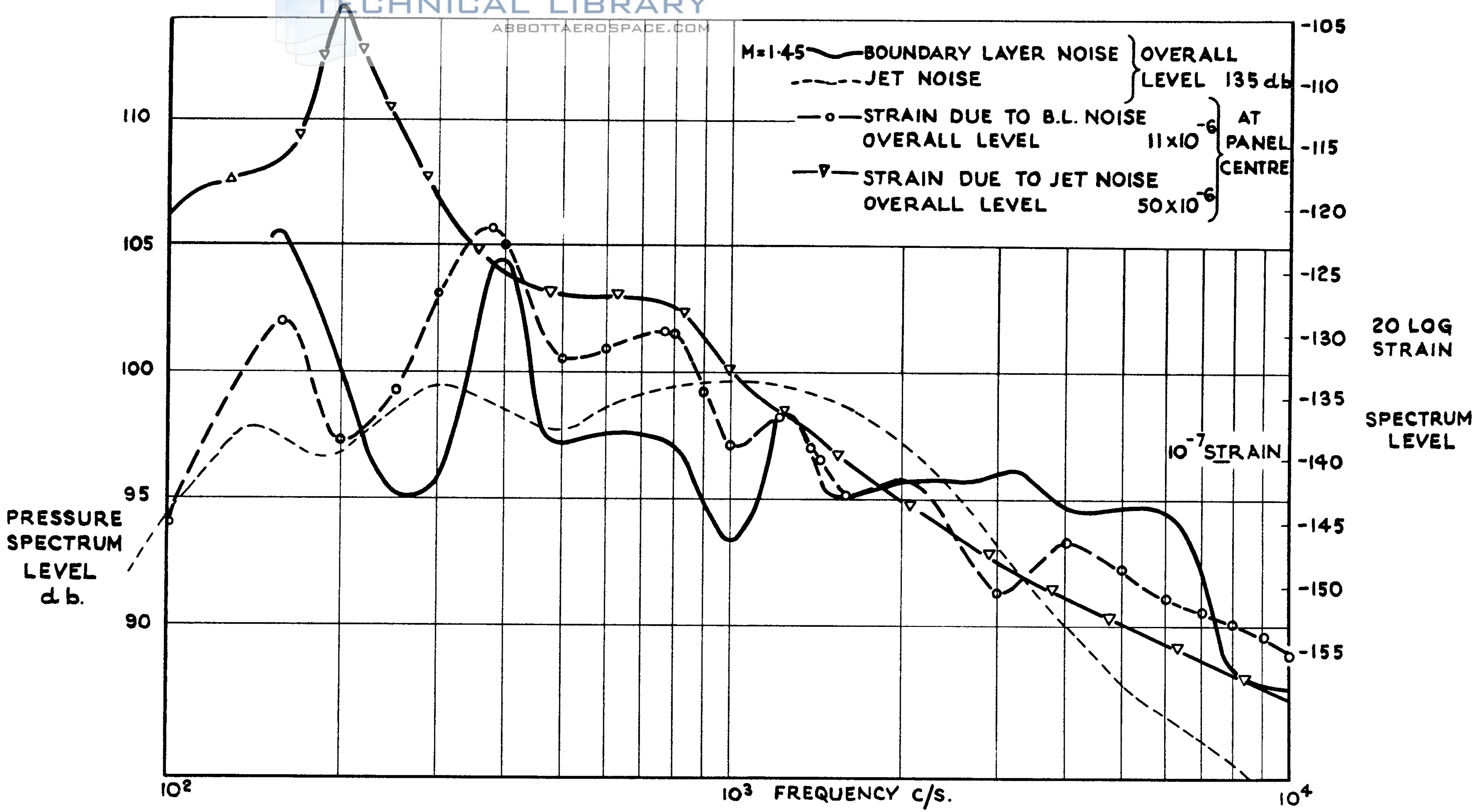


FIG. 10. COMPARISON BETWEEN STRAIN AT CENTRE OF PANEL WHEN EXCITED BY JET NOISE AND BOUNDARY LAYER PRESSURE FLUCTUATIONS.

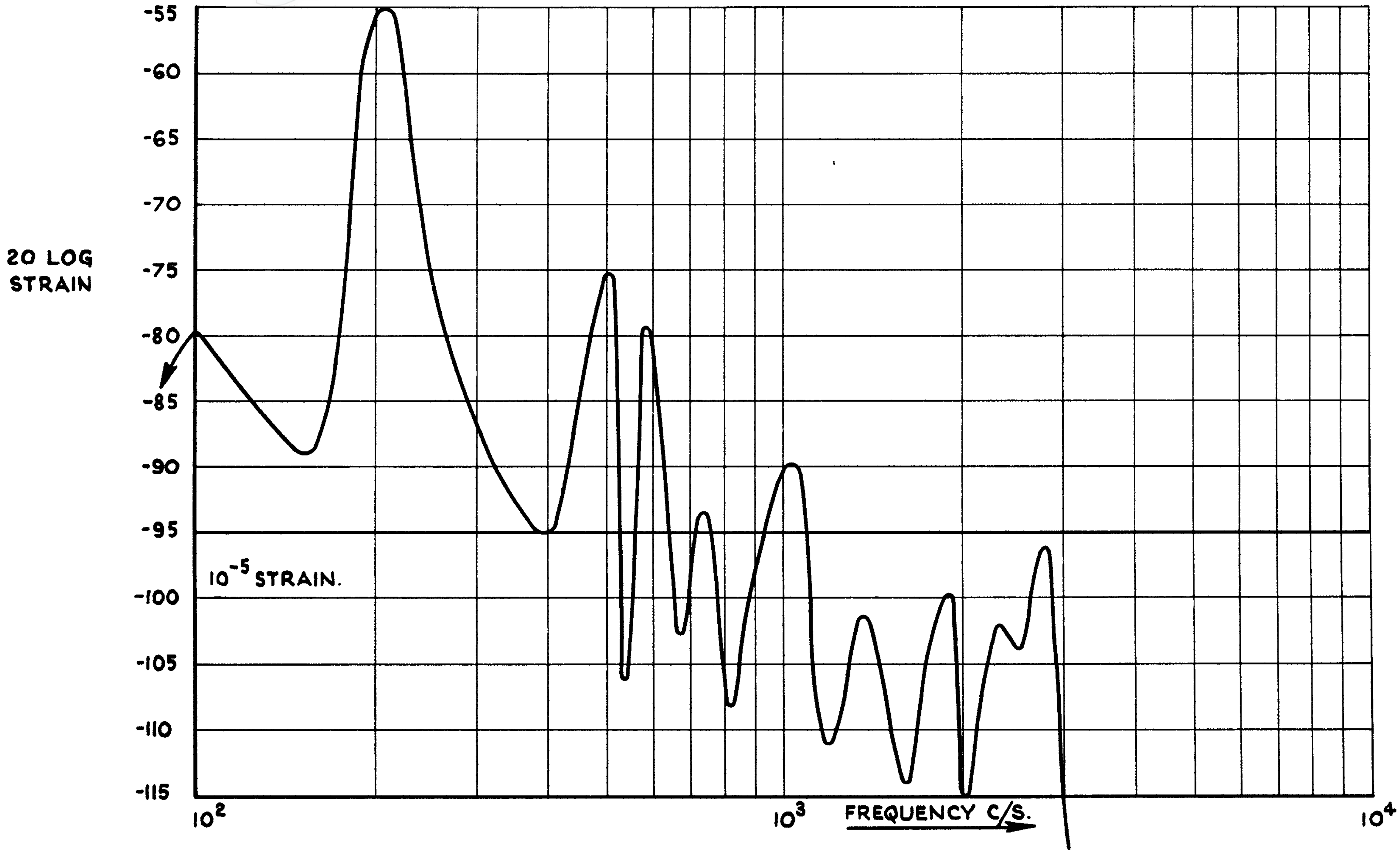


FIG. 11.(a.) STRAIN ON PANEL ACOUSTICALLY EXCITED BY SINE WAVES HAVING A S.P.L. OF 135 db AT THE PANEL — NORMAL INCIDENCE.

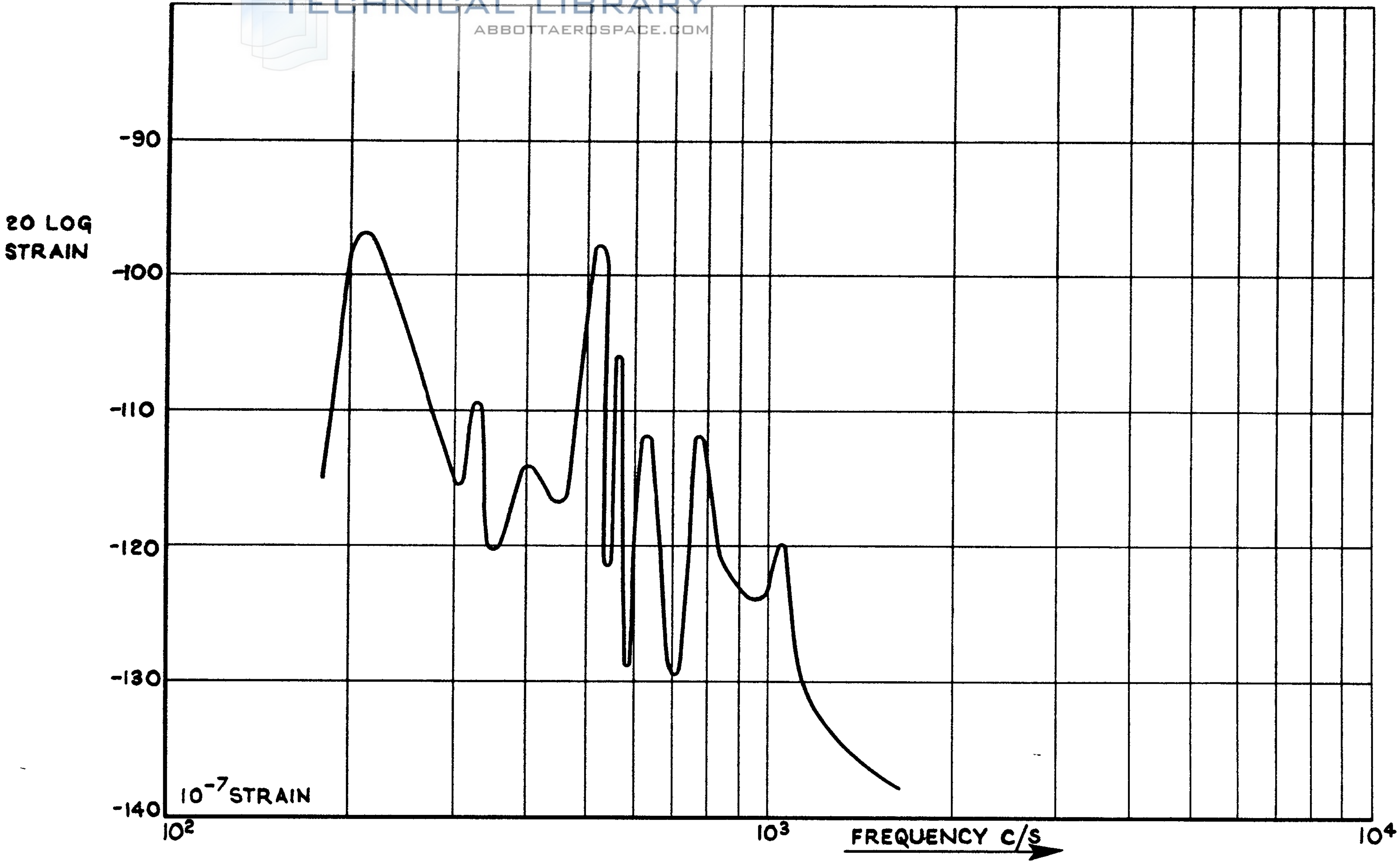


FIG. 11.(b.) STRAIN ON PANEL ACOUSTICALLY EXCITED BY SINE WAVES HAVING A S.P.L. OF 135 db AT THE PANEL — GRAZING INCIDENCE.

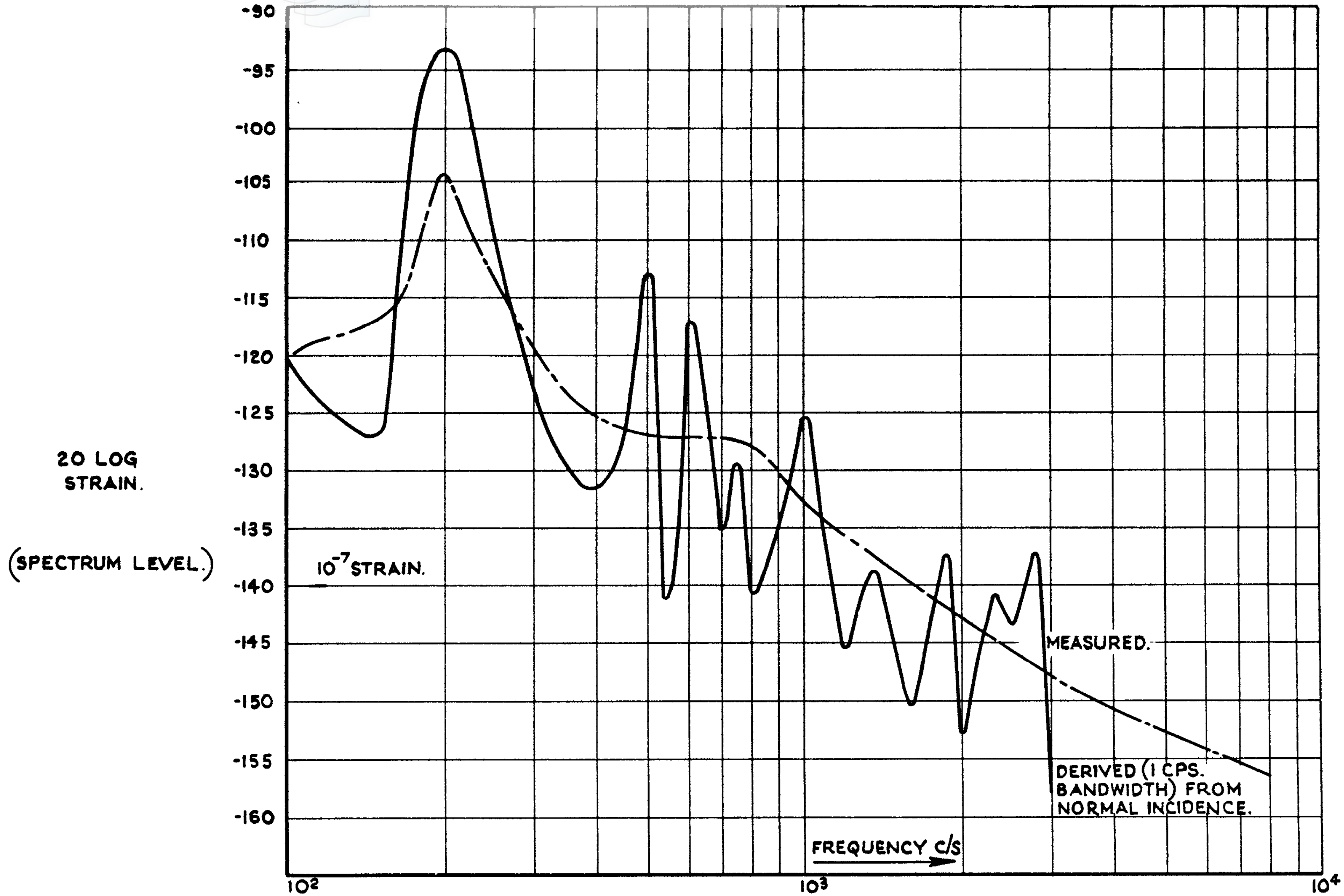


FIG. 12. COMPARISON BETWEEN STRAIN DERIVED FROM THE RESPONSE OF THE TEST PANEL TO SINE WAVES OF NORMAL INCIDENCE, AND THE MEASURED STRAIN RESULTING FROM EXCITATION BY JET EFFLUX NOISE.

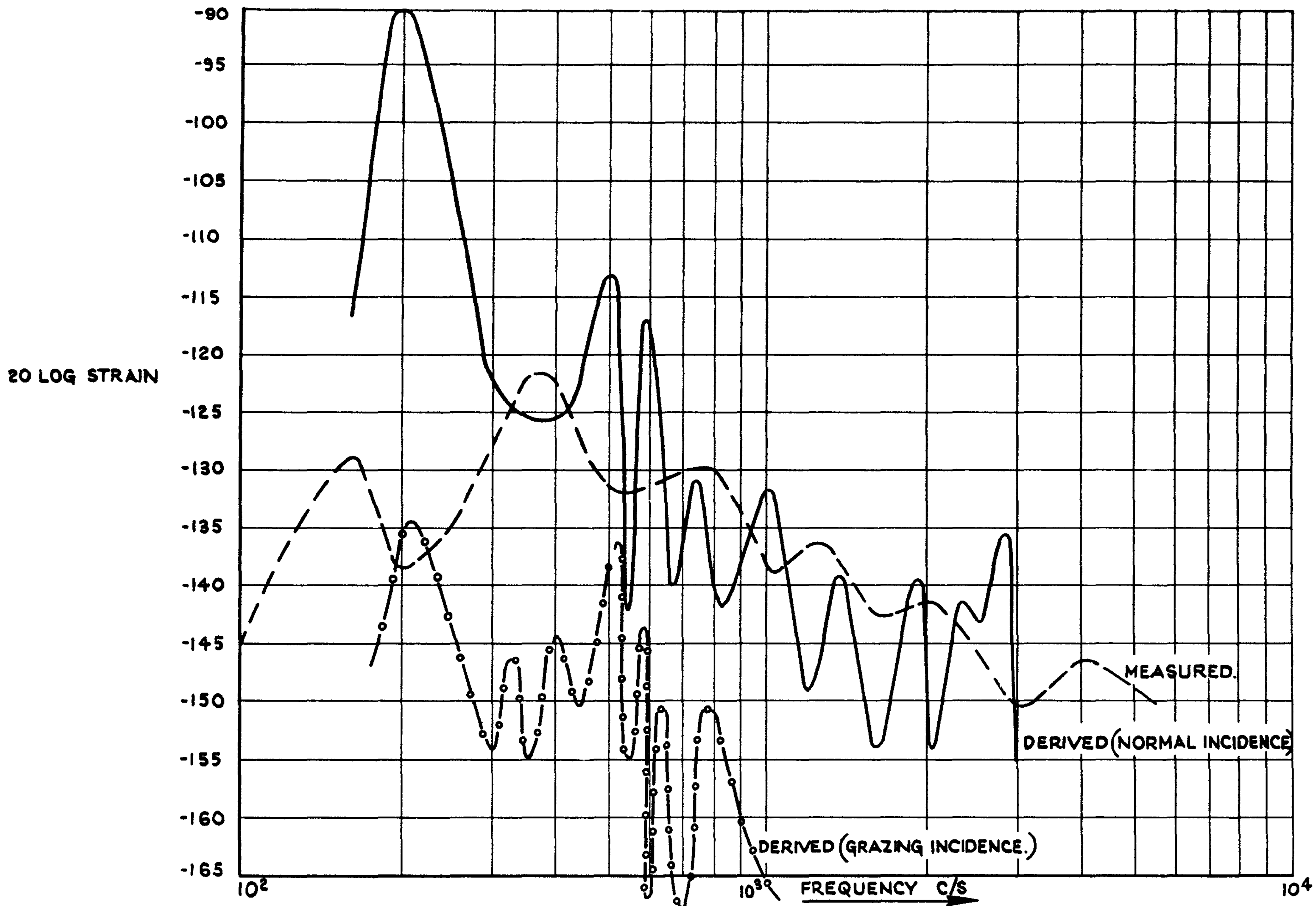


FIG. 13. COMPARISON BETWEEN STRAIN DERIVED FROM THE RESPONSE OF THE TEST PANEL TO SINE WAVES OF GRAZING AND NORMAL INCIDENCE, AND THE MEASURED STRAIN RESULTING FROM BOUNDARY LAYER EXCITATION.

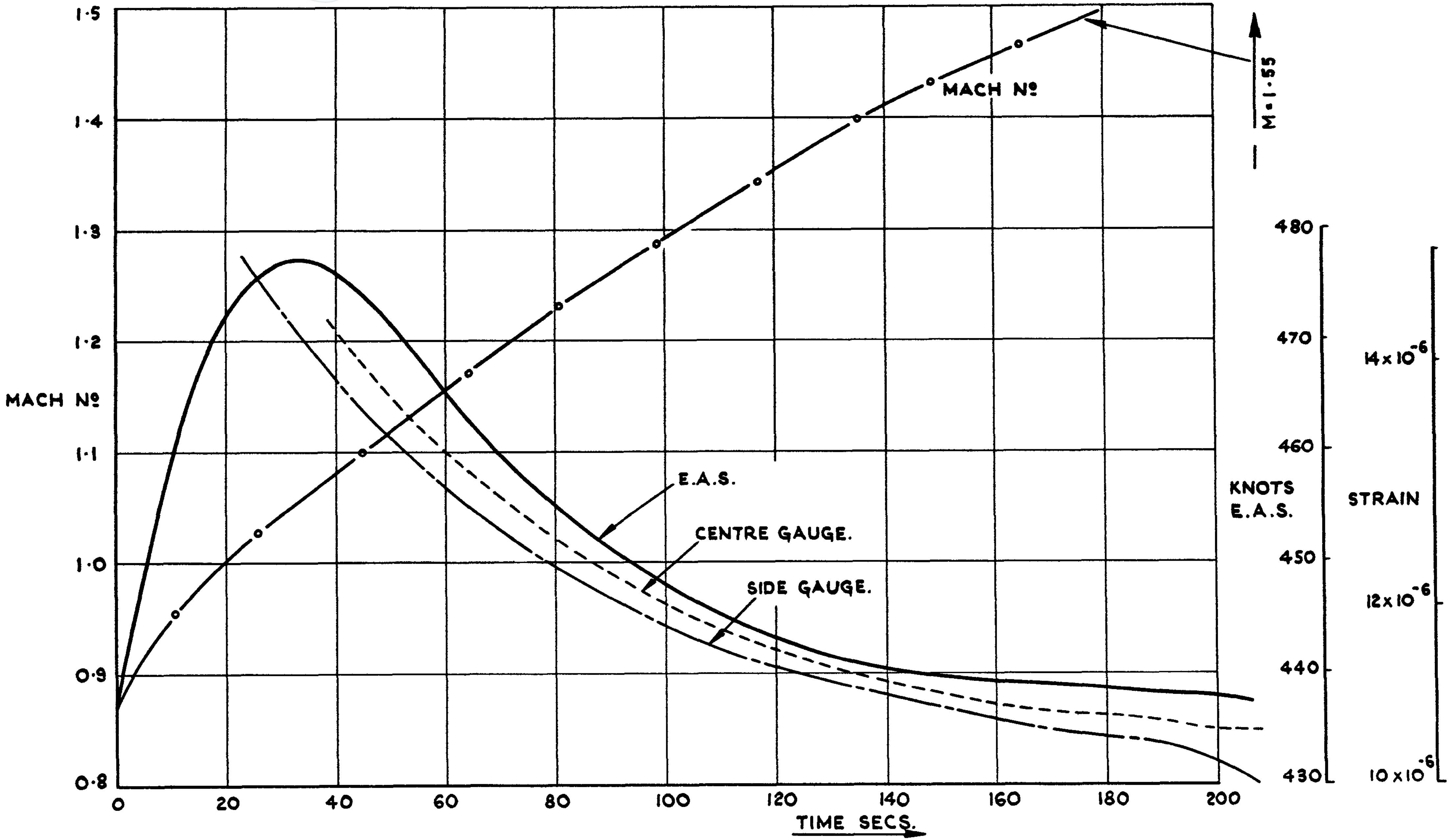
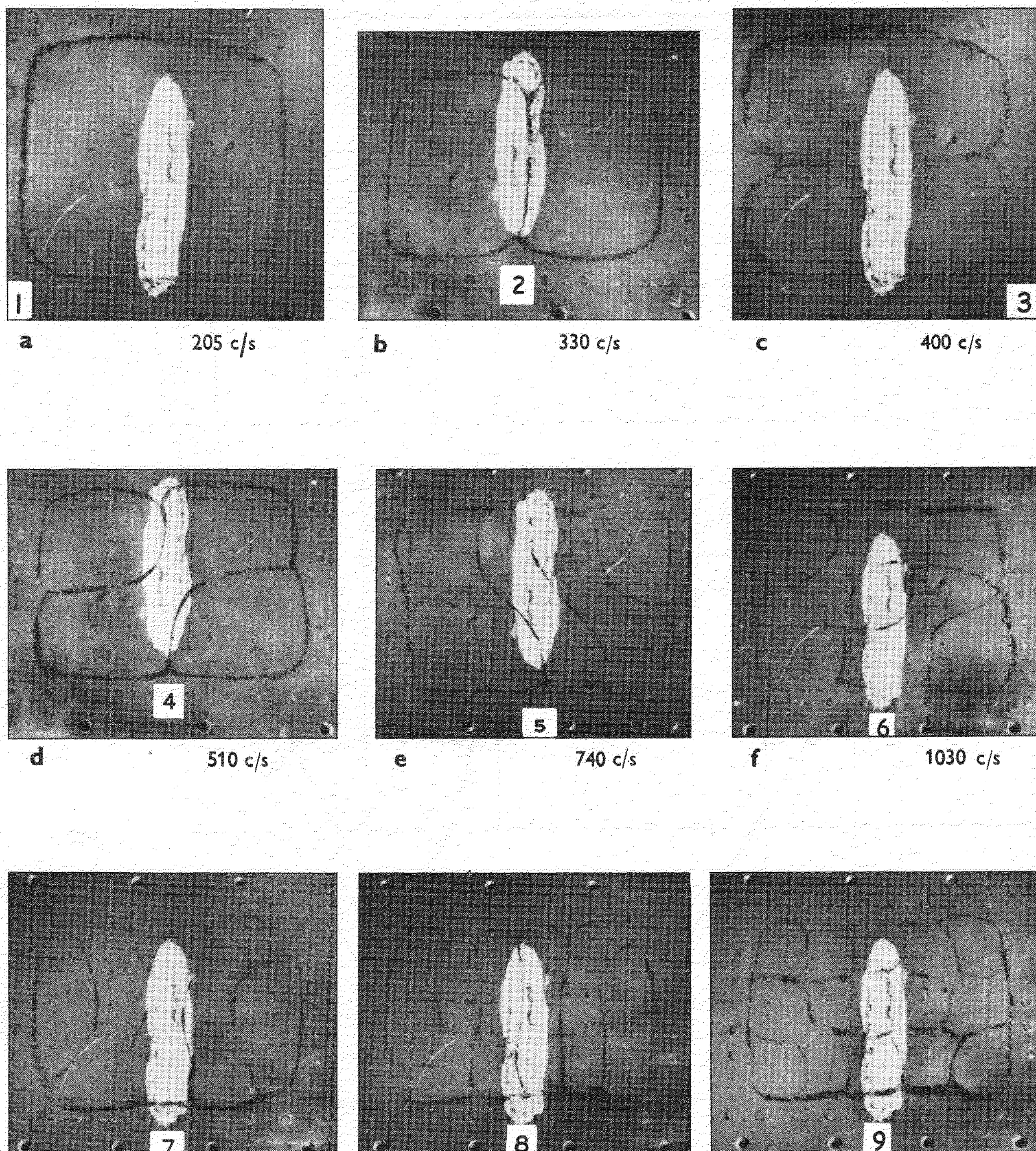


FIG. 14. STRAIN AT TWO POINTS ON TEST PANEL.



a 205 c/s
b 330 c/s
c 400 c/s
d 510 c/s
e 740 c/s
f 1030 c/s
g 1360 c/s
h 1910 c/s
i 2700 c/s

FIG.15. THE FIRST NINE MODES OF THE TEST PANEL WHEN EXCITED BY LOUDSPEAKER GENERATED SINE WAVES

A.R.C. C.P. No.638

532.526.4 :
629.13.012.5 :
533.6.0482 :
539.382

SURFACE PRESSURES AND STRUCTURAL STRAINS RESULTING FROM FLUCTUATIONS IN THE TURBULENT BOUNDARY LAYER OF A FAIREY DELTA 2 AIRCRAFT. Webb, D.R.B., Keeler, A.R., Allen, G.R. May, 1962.

This paper describes measurements of surface pressure and structural strain due to fluctuations in the turbulent boundary layer of a Fairey Delta 2 aircraft. The measurements were made on the lower surface of the wing at Mach numbers up to 1.6.

The results show that both pressure and strain depend on the kinetic pressure q and are independent of Mach number. It is deduced that the levels of strain due to the pressure fluctuations on the surface of a proposed supersonic transport aircraft would be higher than those obtaining on existing aircraft.

(Over)

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(Over)

The pressure spectrum was found to be rather flat over the whole of the measured range and only slightly affected by Mach number. The strain spectra on a specially constructed test panel indicated that the major mode of response was of a higher degree than when excited by jet noise, with the panel response being primarily non-resonant.

The overall strain was about one-third of that occurring when the panel was excited by the efflux noise of a stationary jet engine, for the same overall sound pressure level.

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