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THE COLLEGE OF AERONAUTICS

DEPARTMENT OF AIRCRAFT DESIGN

Design study for noise tests on laminar flow fin

SUMMARY

The study deals mainly with engineering problems associated with the introduction of a noise generating facility on 'Lincoln' aircraft R.F.342 for the purpose of studying the effect of high intensity sound pressure levels at both random and discrete frequencies on the laminar flow fin currently being tested in this aircraft.

The frequencies selected as representative of both turbulent boundary layer and propulsion system disturbances lie within the band 200 to 2000 c.p.s. with an upper limit of 1200 c.p.s. for discrete work. Required sound pressure level at the fin surface is estimated to be 130 db.

Conclusions are that the project is feasible but with the reservation that certain estimated figures and effects will require to be confirmed by a test programme prior to mounting the full-scale experiment.

(Prepared under Ministry of Aviation Agreement No. PD/28/024).

Contents

1. Introduction

It has been established that acoustic disturbances radiated from the turbulent boundary layer on unsucked areas of a laminar flow aircraft, together with noise generated by the propulsion system, may have sufficient effect on the laminar flow surfaces to cause transition.

Dr. Pfenninger[1] of the Norair Group, Northrop Corporation, has investigated the effect of noise on the behaviour of a swept laminar suction wing in the Norair 7' x 10' tunnel. These tests were, however, conducted in a sound field of a rather complex nature due mainly to reflections from the tunnel walls, therefore they are not really representative of the free field conditions which would occur with an aircraft in flight.

Dr. Pfenninger visited this country in July 1962 in order to discuss specifically, the possibility of carrying out acoustic experiments in flight using the Handley Page laminar flow test wing. J.B. Edwards[2] of Handley Page Ltd., has reported on these discussions and, subsequent to Dr. Pfenninger's visit, a decision was taken to measure the ambient external noise level at the test wing on 'Lancaster' P.A. 474 (the test vehicle at that time) in flight.

This work was carried out during the autumn of 1962 and the report[3] published in February 1963. A further report[4], published in April 1963, summarised results and clarified the general proposal to subject the test wing to fairly high sound pressure levels in flight.

The current design study carries the work a stage further and deals, on a feasibility basis, with design and engineering problems associated with the provision of an acoustic test facility on 'Lincoln' RF.342 (the present test vehicle) for the purpose of achieving the required SPL's at the test wing surface.

Note: Ambient levels on 'Lincoln' RF.342 will not be identical to the measured levels on 'Lancaster' P.A.474 since the engine/propeller combination is different. (Lanc. - Merlin 38 - 3 bladed prop.: Lincoln - Merlin 68 - 4 bladed prop.). The order of both SPL and frequency however will be similar and it is considered that the existing report[3] would still be valid.

2. General requirements

Due to the limitations of Pfenninger's experiments an assessment of the required S.P.L's can only be approximate. This is highlighted by Hyde[4] in his note and, when ambient levels[3] are considered also, the resulting generated S.P.L. will require to be of the order of 130 db at the test wing surface.

After discussion within the College, it was established that realistic limits of frequency would lie within the band 200 to 2000 c.p.s. with an upper limit of 1200 c.p.s. for discrete frequency work, the most useful range being centred on 600 c.p.s. approximately.

The noise sources are required to be located such that effects of propagation spanwise, chordwise and normal to the test wing surface can be studied. It is clear, however, that the latter condition would be extremely difficult to achieve since it would be necessary to either position the source near the 'Lincoln' wing tip in order to arrive at a near-normal propagation or fabricate a frame structure of substantial proportions outrigged from the fuselage side in the test wing area. Distances involved with the wing tip proposal are of the order of 60 ft. and attenuation at this distance would require a generated S.P.L. far in excess of the practical limit for both envisaged equipment and aircraft structure, whilst a frame of the type necessary for fuselage mounting would be a major installation and would seriously affect performance of the aircraft.

The only feasible locations therefore are those which produce disturbances in the spanwise and chordwise directions; conditions which can be achieved with fuselage mounting of equipment at realistic distances of approximately 21 ft. maximum and 10 ft. minimum.

3. Test facility

3.1 Transducer/horn design and installation

Consideration of the available transducers has indicated that the model EPT-94A electro-pneumatic unit[5] manufactured by Ling Electronics Division of Ling-Temco-Vought Inc. in the U.S.A. and marketed in this country by Pye-Ling Ltd. of Royston, would be suitable. This is an improved version of the model 6786 unit used by Southampton University for their random siren facility[8] and can generate in excess of 2000 acoustic watts 'white noise' with an electrical imput of 90 VA. The frequency range quoted[5,6] is beyond 10 kcs for 'white' or random noise and up to 1.5 kcs for discrete or sinewave noise. Air supply required is 300 cfm ± 10% at 40 p.s.i. gauge pressure.

In order to achieve the correct concentration and beaming characteristics together with efficient transmission of the sound energy it is necessary to couple the transducer to a horn whose increase of cross-sectional area with length follows an exponential law. There are standard horns available, however, they are usually designed to operate in a ground test environment with progressive wave systems and are therefore too small (mouth dimensions of the order of 7″ × 7″ square and a length of approximately 6 ft.) for our purpose or, when a free-field condition is required for testing of larger components the standard unit goes to the other extreme and is too large (mouth dimensions of the order of 4' × 4' square and a length of approximately 12 ft.). It is fairly clear therefore that we will require to design our own exponential horn for the particular aircraft application.

Preliminary calculations indicate that a horn approximately 4′ 6″ long with mouth dimensions of 2′ × 1′ and a flare ratio of 1.4 $ft.^{-1}$ would give the required low frequency characteristics (theoretical cut-off approximately 150 c.p.s.). Such a horn could be installed externally on the fuselage in each of 3 positions as indicated in figure 1 without incurring a serious physical penalty, however, the aerodynamic effects on the test wing may be considerable but could be assessed by a simple test programme to determine the extent of fairing necessary to minimise interference. A typical horn design is shown in figure 3 and installation methods indicated as a proposal in figure 4.

We have therefore:-

Position 1. Installation fairly straightforward and problems of attachment to the fuselage can apparently be solved without too much complication. Since we are designing and, presumably manufacturing the horn, it should be possible to 'tailor' to the fuselage contour by a slight bend if necessary in order to minimise drag effects and simplify attachment.

Aerodynamic effects on the test wing with a horn in this position are difficult to assess, however, these can no doubt be measured in flight using a mock-up horn and wool tuft techniques and, if severe, a simple tunnel programme could then be initiated to determine the extent of fairing necessary.

The existing boundary layer fence on the test wing would require to be trimmed over a fairly large area in order to a) permit undistorted propagation of the sound waves and b) minimise sound pressure reflections on the fuselage structure. This would also have some effect on test wing pressure conditions of course but allowances could no doubt be made such that spanwise noise tests would still be valid.

Position 2. Again, the actual installation problems do not seem to be too severe but, in this case, assuming an identical horn, a test programme similar to that outlined above could also be carried out in order to decide the optimum fairing shape required.

Position 3. Again, physically, the problems are not too difficult. Drag conditions would be rather more serious but this is simply a matter of detail design applied to the horn/fuselage attachment, and of course to the horn itself since we now have a ram-air condition. Aerodynamic effects on the test wing should not be quite so critical as in positions 1 and 2, however, there will certainly be some effect which would clearly be much more difficult to minimise with the horn mouth facing upstream. The test programme could again be invoked however, to study the severity and methods of keeping interference to an acceptable level.

3.2 Aircraft Services

'Lincoln' aircraft RF.342, immediately prior to purchase by the College on behalf of M.O.A., was used by D. Napier and Sons Ltd., for de-icing research. A Turbomeca 'Palouste' 3 is installed in the bomb bay, housed within a large randome blister originally fitted when the aircraft was used for radar flight testing. This engine, which is essentially a gas turbine air compressor, has a power output in the form of compressed air bled off between compression and combustion stages. The air originally supplied spray nozzles on icing simulation equipment when the aircraft was operated by Napiers in the de-icing research role and at maximum engine R.P.M. (34,000) and standard sea level conditions, delivery is between 980 and 1860 c.f.m. at a pressure of between 41.5 and 43 p.s.i.g. Unfortunately, altitude compensation for fuel metering is not embodied on the unit, hence jet pipe temperature must be controlled with altitude to keep within the specified limits. This can only be done by throttling back as altitude increases and Napiers estimate that a mass flow of 1.10 lbs/sec. (326 c.f.m.) at a delivery pressure of 26 p.s.i.g. would be available at the test altitude of 10,000 ft.

Curves[7] supplied by Pye-Ling Ltd. indicate that, at this gauge pressure and with an air mass flow of 260 c.f.m. the acoustic output from the transducer would be 1650 watts measured at the mouth of the standard 'Ling' progressive wave tube horn previously referred to (7" × 7" mouth, 6 ft. long, low frequency cut off 50 c.p.s.). There are no figures available from Pye-Ling to indicate actual output at the transducer under these reduced pressure conditions however, since the unit is claimed to generate in excess of 2000 acoustic watts and since the Pye-Ling figures are based on this power being available at the mouth of the horn under test conditions then, with our proposed horn and reduced output of 1650 acoustic watts, we should still achieve a S.P.L. of the order of 159.5 db. at the mouth assuming comparable horn efficiencies.

The figures quoted are of course for the general case and it is possible that we could have a unit on the bottom of the acceptance limit. If we therefore assume this condition and extrapolate from the curves, the minimum transducer performance would still give approximately 158 db at the horn mouth. There will also be a further reduction due to relative density conditions at 10,000 ft. giving a final generated S.P.L. of the order of 155 db.

Considering the maximum horn distance of 21 ft. from the test wing (position 2 figure 1), discussions with R.A.E. Farnborough have indicated that an attenuation of 15-20 db could be expected over this distance, assuming a natural expansion for the sound. This therefore means that in the worst case we should achieve 135 db. at the wing surface, giving a reasonable margin in hand over the required 130 db. level. On this basis, horn positions 1 and 3 (figure 1) would also be satisfactory.

Electrical power requirements are fairly nominal. Input to the transducer is 15 V R.M.S. (21 V peak) at 6.0 amps R.M.S. maximum (90VA), whilst the aircraft can generate a total of 12 kw 28V D.C., approximately 6 kw of which is at present unused. There are, in addition, two 30 kVA alternators, installed by Napiers on the outboard engines and, again, used by them in the de-icing role. The units are not required for the routine current exercise hence this power source is also available for the noise tests.

We can conclude therefore that although available air supplies existing on the aircraft do not meet the maximum transducer demand, the order of S.P.L. which can be generated at the horn mouth gives an acceptable margin over the requirement on the basis of analysis. In order to substantiate the estimated figures, however, it is recommended that a free field test of the coupled transducer/horn facility should be carried out. This can be a relatively simple exercise and proposals for such a test are included in section 4. Electrical power supplies, on the other hand, are clearly adequate and do not present any problem.

3.3 Instrumentation and control

A diagrammatic layout is shown in figure 2. This indicates the maximum equipment which would be necessary for control and analysis of the complete facility as visualised. The high frequency driver (shown dotted) is a possible means of augmenting sound pressure levels at the higher frequencies using electrical power only. This is included merely to indicate that a method of 'boosting' is available should the requirement arise.

The proposal is that the main bulk of instrumentation units for the facility should be installed in a multibank racking immediately aft of the manometer. This will enable control of noise input, analysis and calibration to be in the hands of the flight test observer in that area. There will be some variations in observed frequencies at the wing surface between horn positions 2 and 3 (figure 1) due to the 'doppler' effect, however, in order to achieve specific frequency conditions, fairly continuous monitoring will be necessary in any event and can no doubt be resolved as a matter of control technique.

Control of the 'Palouste' A.P.U. is from a station in the cockpit, behind the pilot approximately at the navigator's position (in the original aircraft service role). This will be in the hands of the flight engineer so that, clearly, intercommunication between the observer aft and the 'Palouste' operator will require to be immediate and positive. The normal intercommunication system should of course be adequate, however, a system of ident. lights between the two for the emergency case could be introduced fairly easily as a standby. In addition, it would be necessary to provide a further spill valve for the air supply, to be under the direct control of the observer such that he can select air as well as electrical power to the transducers at will, provided the 'Palouste' is running at the correct conditions.

The diagram is self-explanatory and a full list of equipment with probable cost and delivery status is included in section 6.

3.4 Effect of generated S.P.L. on crew

The crew in the cockpit are, at the moment, subjected to S.P.L's in the range 115/120 db. continuously in flight from the engine/propellor combination.[3] A generated S.P.L. of 155 db at the horn mouth is, as previously discussed, in excess of requirements by about 5 db. This means that the maximum S.P.L. we need generate in position 2 (figure 1) to achieve 130 db at the test wing would be of the order of 150 db. The attenuation, however, on which this is based is an estimated figure and it would therefore be politic to assume at this stage that the maximum output of 155 db would be necessary at this position. The 'Budworth' engine operator would be the crew member subjected to the highest level and, if we assume a drop of approximately 15 db through the fuselage, with a further drop due to distance attenuation, the level he will experience will be of the order of 135 db. Similarly, with the horn in position 1 (figure 1) the 'Budworth' operator would again be subjected to the highest level, however, in this case, the generated S.P.L. would be considerably less, hence the order is unlikely to be more severe than that resulting from position 2 (figure 1). Position 3 (figure 1) would have the greatest effect on the flight test observers in the manometer/instrumentation area, however, again the generated S.P.L. would be less and again it is unlikely that the 135 db figure would be exceeded. From the above estimation, fuselage sound-proofing will clearly be necessary in the critical areas, however this can be accomplished fairly easily and should result in insulation of the crew down to an acceptable level below the 130 db threshold. Ear-defender headsets are available and it is recommended that these should also be used, possibly coupled with throat microphones to provide the complete intercommunication facility.

The intensity levels in themselves, therefore, are not particularly serious, however, when coupled with the frequency spectrums necessary for the test programme, the problem becomes aggravated and it is extremely difficult to assess the precise effect. The recommended use of ear-defenders however, could be expected to provide an attenuation of between 25 db and 30 db and, when this is coupled with the fuselage sound-proofing, levels should be acceptable down to our lowest attainable frequency of 150 c.p.s.

A degree of fuselage sound-proofing is therefore a firm recommendation. The crew under these conditions would not be subjected to S.P.L's of any greater magnitude than they experience in the cockpit at the moment, indeed, the effect of generated S.P.L's on the basis of estimation, would be a good deal less.

3.5 Effect of generated S.P.L. on structure and equipment

3.5.1 Aircraft structure

Fuselage panels in the region of the horn mouth are estimated to have a resonant frequency of between 168 and 238 c.p.s. On the basis of a total two hour test period, endurance would be 1.5×10^6 cycles in the low frequency case giving a mean allowable stress level of 20,000 p.s.i. In practice, the source would produce stresses of the order of 1500 p.s.i. and, allowing for stress raisers in the worst condition, factoring by 4 would still give a substantial reserve. If we assume that the frame modes are different from the skin mode then the levels would be increased and the factor correspondingly reduced, however, it is felt that this is unlikely. There is, in addition, a degree of alleviation since the horns are sited at the point of maximum fuselage curvature.

Items of the 'Lincoln' structure which are most suspect are the frame cutout/stringer cleat connections which are more likely to suffer from fatigue cracking problems than the skin/stringer combination, also, since the panel resonant frequencies are fairly low, additional damping will probably not be necessary. Representative panel testing should however establish whether reinforcing/damping is required but on the basis of analysis, no serious modifications would be necessary and, in any event, a strict inspection check would be carried out in the critical areas between each flight.

Tests have in fact been carried out by Southampton University on a flat 'Lincoln' fuselage panel specimen. This was subjected to intensity levels up to 150 db with a 5 hour total endurance at frequencies between 88 and 1400 c.p.s. in the acoustic testing laboratory.[9] The specimen was still undamaged on completion of the programme however, at the maximum intensity level of 150 db the panel vibrated rather viciously and, since the type of edge restraint was not really representative of conditions obtaining on the actual aircraft, there are some reservations re full acceptance of the results. Further fuselage panels are available and Southampton have very kindly offered the use of their facility should additional testing be required. It is recommended that the offer should be accepted and further tests carried out with perhaps more representative edge fixing and possibly strain gauging to establish actual stress levels in confirmation of the above analysis. Proposals are included in section 4.

3.5.2. Test wing structure

We would not expect any problems with structural aspects of the test wing. Composite skins etc. should be more than adequate to withstand generated levels at the endurance limits of the proposed programme, however, it is difficult to calculate the effect on such items as internal ducting and 'araldite' filler used at the leading edge/wing skin joint either of which, if damaged, could result in serious disruption of the laminar flow experiment. Whilst a detail analysis of effects on ducting etc. has not yet been carried out, it is thought unlikely that significant damage would result from the proposed S.P.L./frequency combination. A test programme is not really feasible since the actual wing structure would require to be represented in any specimen and this would clearly be a major exercise. Alternatively, the actual wing itself could be used but a) there is no access to internal structure, in the case of ducting, and therefore no means of inspection before and after test and b) subjecting the wing to higher endurance levels increases the risk of damage accordingly.

In general, a test programme is thought to be neither feasible nor justified for this particular component. The feeling is that damage is unlikely but that a more detailed analysis of the possible effect on both ducting and filler should be carried out prior to mounting the actual experiment.

3.5.3 Radio and instrumentation equipment

Again, it is difficult to assess the precise effect on equipment. Some types of radio, for example, have rather bad characteristics which could lead to microphonous problems. However, as previously discussed, it will be necessary to introduce a degree of sound-proofing in the fuselage in any case, with corresponding benefits applicable to both crew and equipment. In view of this, levels inside the fuselage should not be excessive and it is felt that a 'suck it and see' philosophy is justified, backed perhaps by some consultation with manufacturers prior to flight when exact levels in the fuselage are determined on ground test.

4. Test recommendations to confirm feasibility

4.1 Aerodynamic (Section 3.1)

Proposed horn positions are indicated in figure 1 and it is suggested that a test programme could be initiated during the currency of the present flight tests.

The programme would consist of design and manufacture of a mock-up transducer/horn unit (which could be used as a basis for the actual exponential horn) to be mounted on the fuselage at the revelant positions, coupled with wool-tuft installation in the critical areas. Flow patterns could then be studied visually (using the existing T.V. camera) and an assessment made of effects on the test wing.

If it is clear that these effects are too serious to be tolerated then a fairing would require to be designed. The simplest method of arriving at an optimum shape would be by means of a wind tunnel programme, coupled with subsequent installation on the aircraft, again on a mock-up basis, to confirm results.

The above programme could proceed immediately with an estimated total cost of £550 if a fairing is required or £260 if the initial flight test indicates that this is not necessary.

4.2 Transducer/horn facility (Section 3.2)

The degree of attenuation under free-field conditions has been estimated in order to establish expected S.P.L's at the test wing surface, however, there are a number of variables such as performance of the coupled transducer/horn unit which are difficult to assess precisely. Although the best available advice has been taken on this (Structures Dept. R.A.E. Farnborough) it is considered that, since a simple test could be carried out to substantiate the estimate, it would be politic to do so.

The exponential horn design will already be available but a frame mounting of some description will be required, together with an additional frame mounting of the measuring equipment at required distances and attitudes. A transducer can be obtained on loan from either Pye-Ling Ltd. (who have expressed their willingness to co-operate) or Southampton University and both electrical and measuring equipment are available at the College. Air supplies and services are also available.

Total cost is estimated to be £500.

4.3 Structural (Section 3.5.1)

A scrap 'Shackleton' Mk.1 has been located at R.A.E. Farnborough and arrangements made with the senior stores officer to retain the fuselage for an agreed period (end of March 1965) against a possible test requirement. The skin/stringer/frame combination on this aircraft is similar to that of the 'Lincoln' and representative specimens could be cut from the fuselage for test purposes.

As previously noted in the above section, Southampton University have offered the use of their siren facility to carry out further tests and the proposal is that the College should obtain and prepare specimens, in conjunction with Southampton, and make arrangements for testing to proceed.

It is difficult for the College to estimate the total cost of such a programme since arrangements would no doubt be made on a direct basis between M.O.A. and Southampton for their part of the work. The College commitment would be confined to selecting and, presumably, strain gauging of the specimens and liaising with Southampton prior to and during the actual programme.

Estimated cost of the College is £220. Total estimated costs, therefore, for feasibility tests are:-

a)	Aerodynamic	£550 max.	£260 min.
b)	Transducer/horn	£500	
c)	Structural	£220	

giving a total of £1270 max. or £980 min.

All the above test programmes could proceed immediately authority is given.

5. List of equipment required

1) Transducers - 2 off
2) Random noise generator
3) Sine oscillator
4) Equaliser
5) Pre-amplifier
6) Power amplifier
7) Monitor and interlock
8) Pressure switch
9) Microphones - 6 off
10) Microphone selector (2 way)
11) Band pass filter ($^1/_3$ octave)
12) Spectrometer
13) Piston phone for microphone calibration
14) Recorder
15) Oscilloscope
16) Air filter
17) Flow meter
18) Spill valve
19) Electrical equipment (inverters, control gear etc.)
20) Head sets.

Items 1-13 are supplied by Pye-Ling Ltd. who have quoted a delivery status of two/three months and an overall cost estimate of £6900 assuming that two transducers are required. (One unit is at present in transit from the U.S.A. which we will obtain on loan for test purposes - we are under no obligation to buy at this stage).

Items 14 and 15 are already available on the College whilst items 16, 17 and 18 can be obtained well within the overall delivery time scale at an approximate cost of £200. Item 19 would consist mainly of embodiment loan units and it is possible that item 20 can also be obtained from embodiment loan sources; if however, for the purpose of this estimate, we assume that item 20 has to be purchased, the total cost would be £240 based on 8 off sets, with delivery, again, approximately 3 months.

All other items are either comparatively minor or can be obtained using embodiment loan procedures. Total estimated equipment cost therefore, is approximately £7,400 within an overall delivery period of three months.

6. Cost and time-scale estimate

	£
Design labour	1450
Manufacturing labour	2320
Materials	300
Equipment	7400
Flying (30 hours over 9 month period)	10200
	£21670

This figure does not include the feasibility tests which have been costed separately.

The design and installation work could be completed within a three/four month period from receipt of contract, assuming that certain equipment could be ordered in advance of this date.

7. Discussion

On the basis of analysis, it is clear that estimated requirements for sound pressure levels at the test wing surface could be met with a sufficient margin in hand to guarantee against an adverse combination of variables. The expected level of 135 db does of course require confirmation and a nominal test programme is the only realistic method of substantiating this figure.

Sufficient electric and pneumatic power is available on the aircraft to meet the noise facility requirement and a study of the possible effects of generated noise levels and frequencies on the aircraft structure tends to indicate that this will not be a serious hazard, again subject to confirmation by means of a test programme. Effects on crew and equipment can be minimised by insulation of the fuselage (not a difficult exercise) down to an acceptable level; no greater in fact than the levels already existing.

The necessary transducer facility to achieve required S.P.L's and frequency bands is available, together with control and monitoring equipment, within a reasonable time-scale. It is however, rather expensive but could no doubt be used constructively on other programmes after completion of the noise tests.

8. Conclusions

Flight tests to study the effect of noise on the laminar flow fin currently being tested in 'Lincoln' aircraft R.F.342 are feasible, subject to confirmation of certain estimated figures and effects by means of a nominal test programme. We estimate that the experiment would require a total flying time of 30 hours to achieve an actual acoustic testing time of 2 hours within this figure.

References

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3. Trayford, R.S. — External noise level on 'Lancaster' P.A.474. College of Aeronautics Note No. 138 February 1963.

4. Hyde, D. — Some thoughts on a proposed in-flight experiment to study the effect of external noise on a swept laminarised wing. College of Aeronautics Note No. 145 April 1963.

5. Pye-Ling Ltd. — Bulletin No. EPT 94A-264 Model EPT-94A Electro-pneumatic transducer.

6. Pye-Ling Ltd. — Bulletin No. EPT-CUR-764A. Model EPT-94A Electro-pneumatic transducer. Typical performance curves.

7. Ling Electronics Inc. — Engineering data report No. EDR-19 January 1963.

8. Clarkson, B.L. and Pietrusewicz, S.A. — The University of Southampton, Random siren facility. A.A.S.U. Report 204. December 1961.

9. - — The University of Southampton, Acoustic test laboratory report on 'Handley Page laminar flow fatigue specimens'. November 1962.

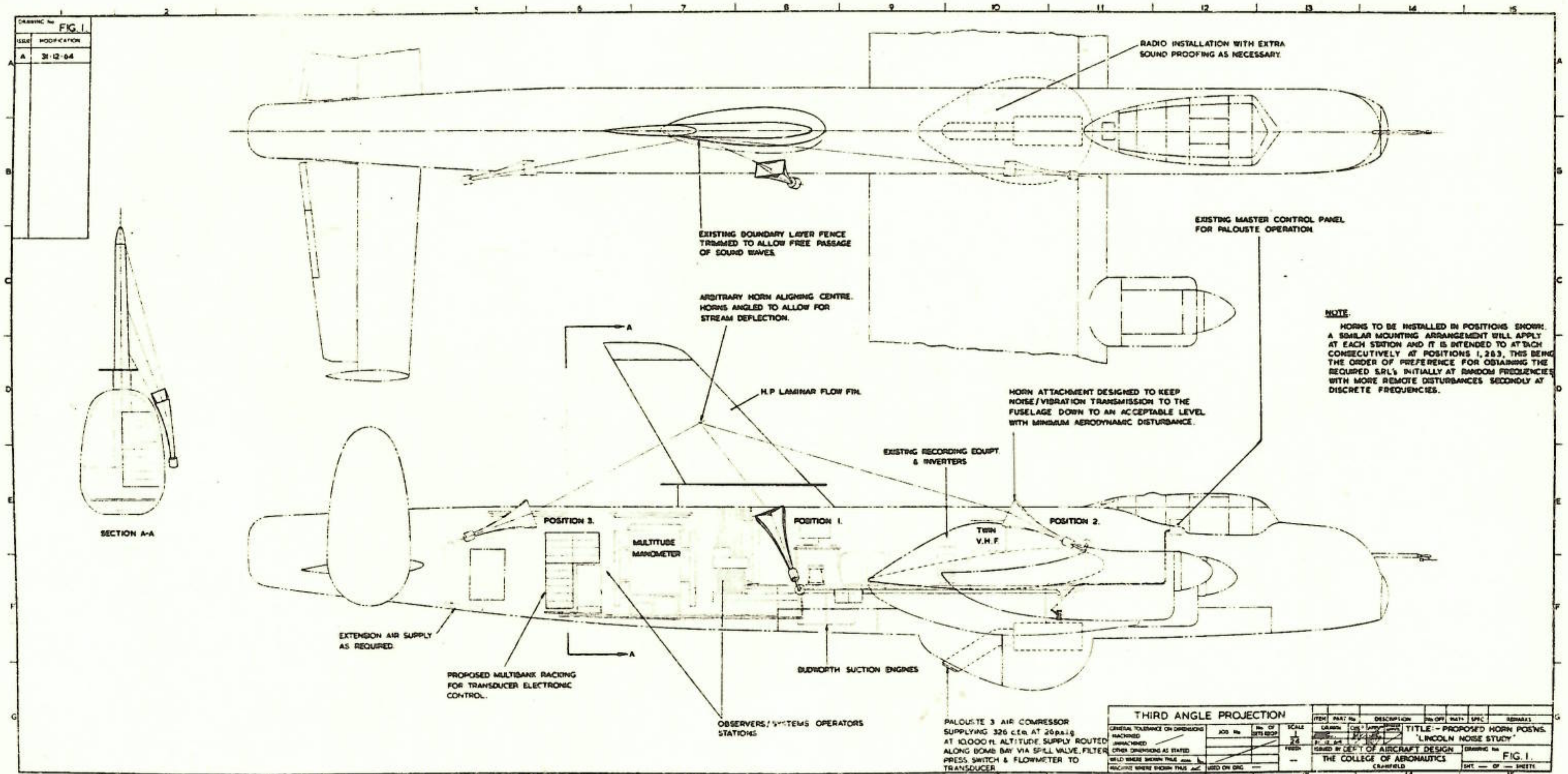
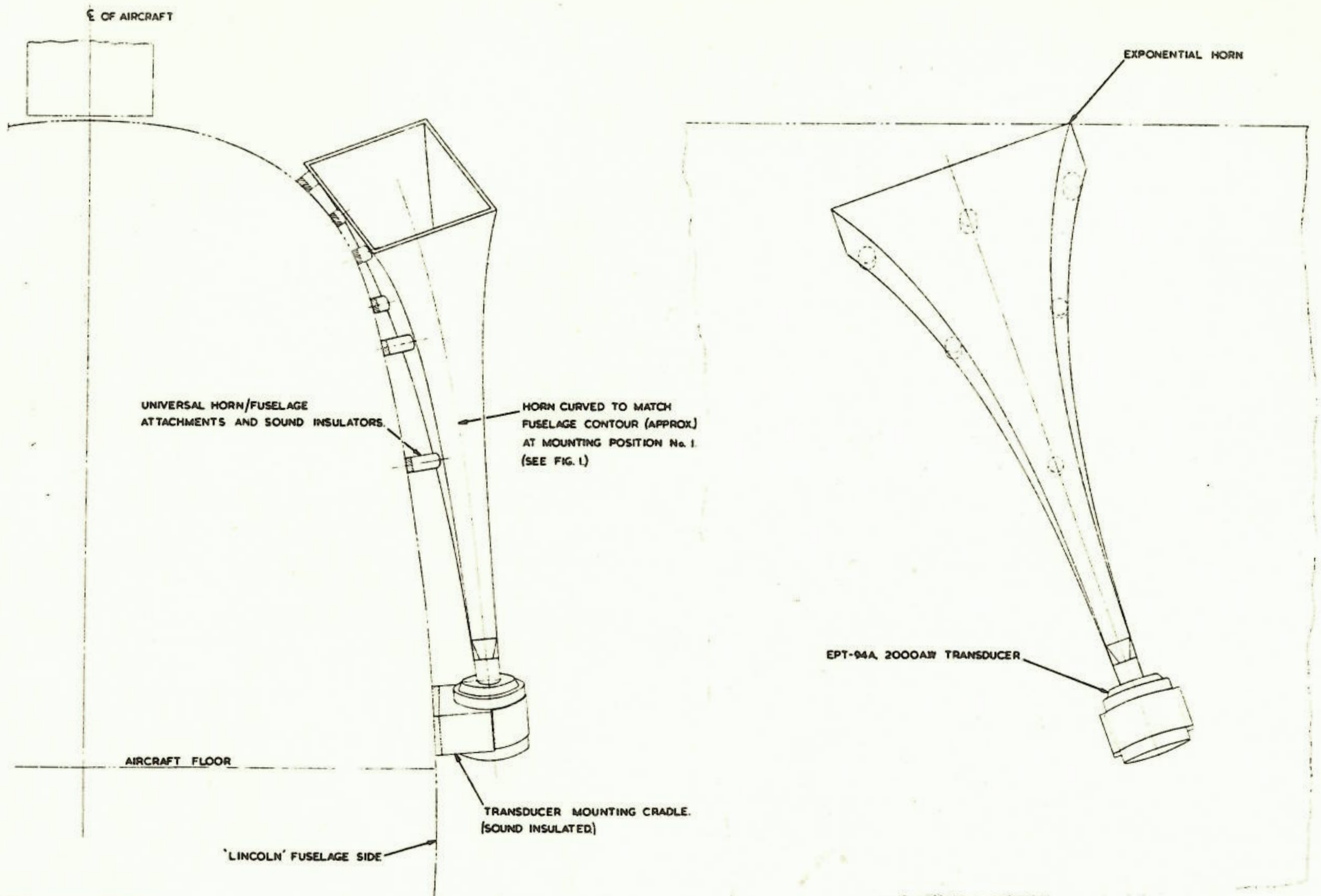


FIG. I.
ISSUE MODIFICATION
A 31-12-64
RADIO INSTALLATION WITH EXTRA SOUND PROOFING AS NECESSARY.
EXISTING BOUNDARY LAYER FENCE TRIMMED TO ALLOW FREE PASSAGE OF SOUND WAVES.
EXISTING MASTER CONTROL PANEL FOR PALOUSTE OPERATION.
ARBITRARY HORN ALIGNING CENTRE. HORNS ANGLED TO ALLOW FOR STREAM DEFLECTION.
NOTE.
HORNS TO BE INSTALLED IN POSITIONS SHOWN. A SIMILAR MOUNTING ARRANGEMENT WILL APPLY AT EACH STATION AND IT IS INTENDED TO ATTACH CONSECUTIVELY AT POSITIONS 1, 2 & 3, THIS BEING THE ORDER OF PREFERENCE FOR OBTAINING THE REQUIRED S.P.L's INITIALLY AT RANDOM FREQUENCIES WITH MORE REMOTE DISTURBANCES SECONDLY AT DISCRETE FREQUENCIES.
H.P. LAMINAR FLOW FIN.
HORN ATTACHMENT DESIGNED TO KEEP NOISE/VIBRATION TRANSMISSION TO THE FUSELAGE DOWN TO AN ACCEPTABLE LEVEL WITH MINIMUM AERODYNAMIC DISTURBANCE.
EXISTING RECORDING EQUIPT & INVERTERS
SECTION A-A
POSITION 3.
MULTITUBE MANOMETER
POSITION 1.
TWIN V.H.F.
POSITION 2.
EXTENSION AIR SUPPLY AS REQUIRED.
PROPOSED MULTIBANK RACKING FOR TRANSDUCER ELECTRONIC CONTROL.
OBSERVERS/SYSTEMS OPERATORS STATIONS
BUDWORTH SUCTION ENGINES
PALOUSTE 3 AIR COMPRESSOR SUPPLYING 326 c.f.m. AT 20 p.s.i.g AT 10,000 ft. ALTITUDE. SUPPLY ROUTED ALONG BOMB BAY VIA SPILL VALVE, FILTER PRESS. SWITCH & FLOWMETER TO TRANSDUCER
THIRD ANGLE PROJECTION
SCALE 1/24
TITLE:- PROPOSED HORN POSN'S. 'LINCOLN NOISE STUDY'
ISSUED BY DEPT OF AIRCRAFT DESIGN
THE COLLEGE OF AERONAUTICS CRANFIELD
DRAWING No FIG. I.

DRAWING No	FIG. 2
ISSUE	MODIFICATION
A	31.12.64



VIEW LOOKING FORWARD

SIDE ELEVATION SHOWING HORN ON STARBOARD SIDE.

NOTE:

SEE FIG. 3 FOR DIAGRAMMATIC LAYOUT OF PROPOSED NOISE SOURCE.
SEE FIG. 4 FOR HORN DESIGN.
TOTAL ASS'Y WEIGHS APPROX. 85 lbs.

THIRD ANGLE PROJECTION

ITEM	PART No	DESCRIPTION	No OFF	MAT'L	SPEC	REMARKS

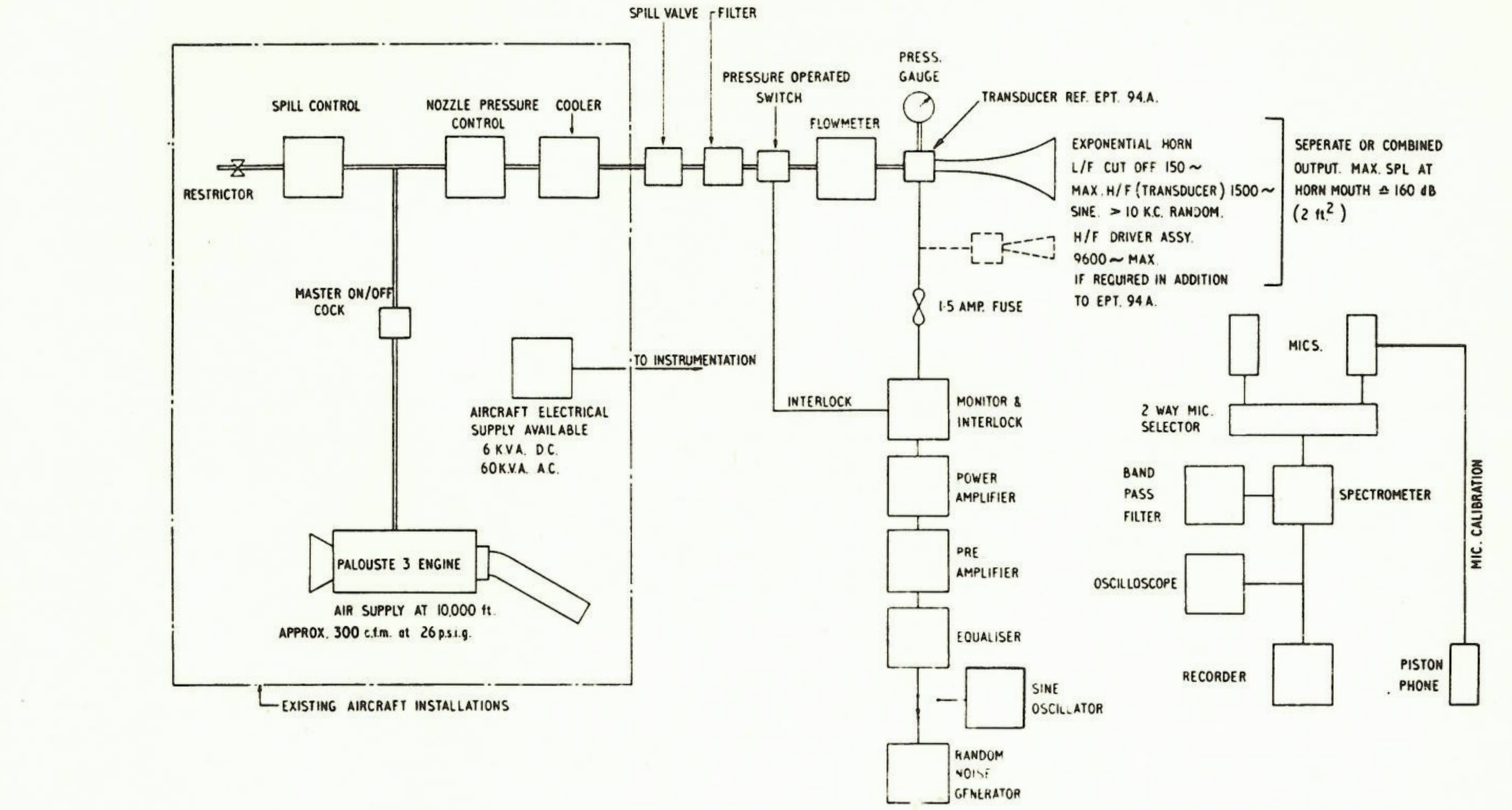
GENERAL TOLERANCE ON DIMENSIONS: MACHINED / UNMACHINED / OTHER DIMENSIONS AS STATED
WELD WHERE SHOWN THUS
MACHINE WHERE SHOWN THUS
JOB No — No OF SETS REQ'D —
USED ON DRG. FIG. 1.
SCALE —
FINISH —
DRAWN 31.12.64 CHK'D APPV'D STRESS APPV'D

TITLE:- PROPOSED HORN INST'N. 'LINCOLN NOISE STUDY'

ISSUED BY DEP'T OF AIRCRAFT DESIGN
THE COLLEGE OF AERONAUTICS
CRANFIELD

DRAWING No FIG. 2
SHT — OF — SHEETS

SKM 60-8

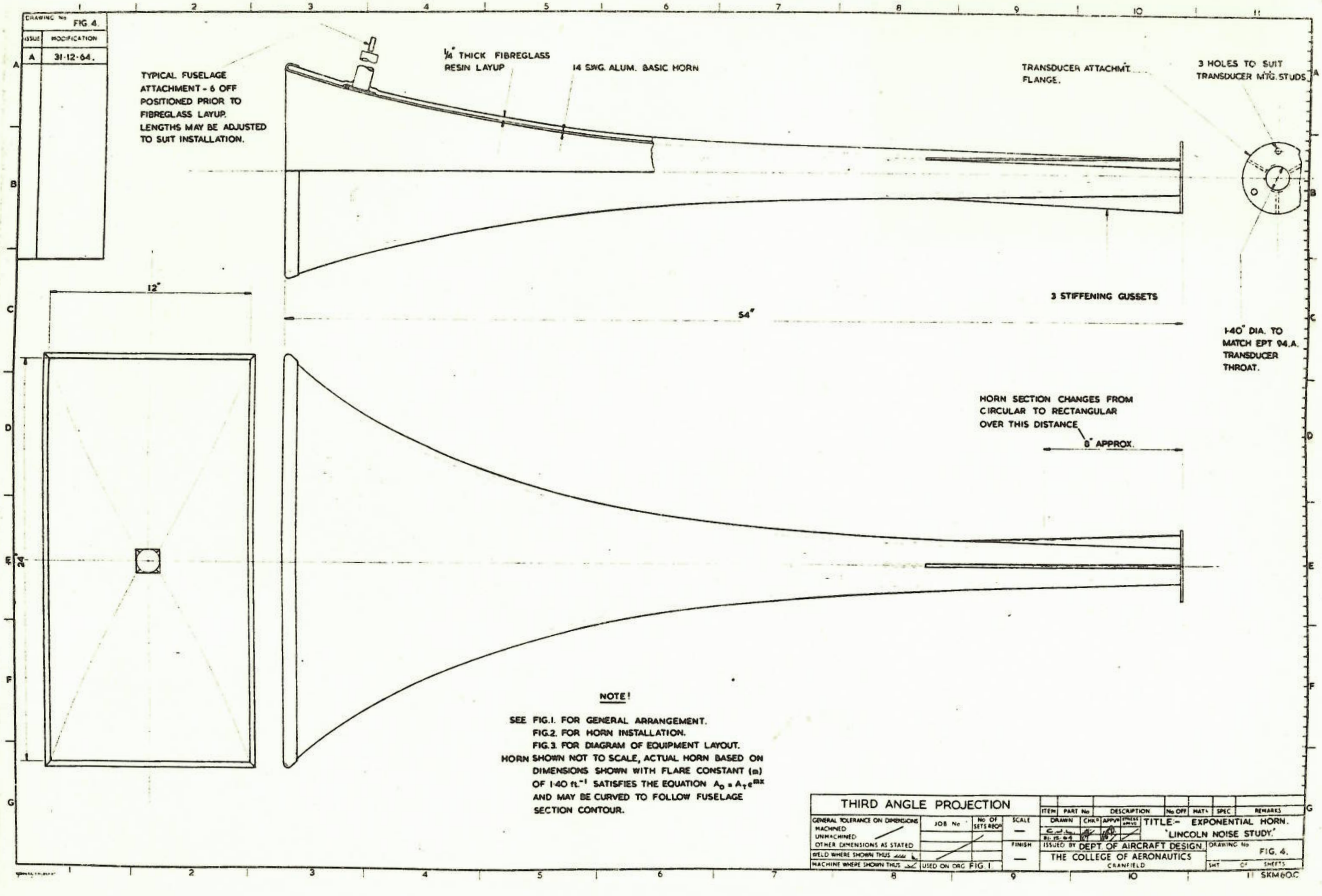


NOTE. ABOVE IS THE MAXIMUM EQUIPMENT NECESSARY FOR FLIGHT TRIALS.

DRG. NO. S.K.M. 60.A. FIG.3.

DIAGRAMMATIC LAYOUT OF PROPOSED NOISE SOURCE - LINCOLN

COLLEGE OF AERONAUTICS
DEPARTMENT OF AIRCRAFT DESIGN
DRN. R.E.L. DECEMBER 1964.


DRAWING No FIG. 4.
ISSUE MODIFICATION
A 31-12-64.
TYPICAL FUSELAGE ATTACHMENT - 6 OFF POSITIONED PRIOR TO FIBREGLASS LAYUP. LENGTHS MAY BE ADJUSTED TO SUIT INSTALLATION.
1/4" THICK FIBREGLASS RESIN LAYUP
14 SWG. ALUM. BASIC HORN
TRANSDUCER ATTACHMT. FLANGE.
3 HOLES TO SUIT TRANSDUCER MTG. STUDS
3 STIFFENING GUSSETS
12"
54"
1·40" DIA. TO MATCH EPT 94.A. TRANSDUCER THROAT.
HORN SECTION CHANGES FROM CIRCULAR TO RECTANGULAR OVER THIS DISTANCE
8" APPROX.
24"
NOTE!
SEE FIG.1. FOR GENERAL ARRANGEMENT.
FIG.2. FOR HORN INSTALLATION.
FIG.3. FOR DIAGRAM OF EQUIPMENT LAYOUT.
HORN SHOWN NOT TO SCALE, ACTUAL HORN BASED ON DIMENSIONS SHOWN WITH FLARE CONSTANT (m) OF 1·40 ft.⁻¹ SATISFIES THE EQUATION A0 = AT e^mx AND MAY BE CURVED TO FOLLOW FUSELAGE SECTION CONTOUR.
THIRD ANGLE PROJECTION
ITEM PART No DESCRIPTION No OFF MATL SPEC REMARKS
GENERAL TOLERANCE ON DIMENSIONS
MACHINED
UNMACHINED
OTHER DIMENSIONS AS STATED
WELD WHERE SHOWN THUS
MACHINE WHERE SHOWN THUS
JOB No
No OF SETS REQD
USED ON DRG FIG. 1
SCALE —
FINISH —
DRAWN CHKD APPVD STRESS APPVD
31-12-64
TITLE:- EXPONENTIAL HORN. 'LINCOLN NOISE STUDY.'
ISSUED BY DEPT. OF AIRCRAFT DESIGN.
THE COLLEGE OF AERONAUTICS
CRANFIELD
DRAWING No FIG. 4.
SHT OF SHEETS
SKM60C