A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

UNCLASSIFIED

AIRCRAFT: C-105

REPORT NO 7/0500/10

FILE NO

NO OF SHEETS: 38

TITLE:

STRUCTURAL TESTS

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PREPARED BY F.P. Mitchell DATE May 1956

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FORM 1316 A

INTRODUCTION

This report broadly deals with the static and fatigue test programme. The general philosphies behind the programme, and test results to date.

The test programme can be broken into five phases as follows:-

Phase 1 Preliminary Design Testing

This testing is not a contractual obligation but is required by the Company to establish the design. The main aspect of this testing is the time element. It must be carried out early in the design stages of the aircraft. As a result the test specimen are simiplified and usually differ in some respects from the final article.

Phase 2 Proof of Compliance

This series of tests will be conducted almost entirely on the static test aircraft and is engineered to meet the requirements of Specification MIL-S-5710.

Phase 3 Possible Component Fatigue Testing

At this time no definite plans are made along these lines. At present, it is assumed that specimen fatigue testing along with a static test specimen which has been well strain gauged, will suffice.

Phase 4 Fail Safe Testing

This type of testing is not an alternative to component fatigue testing but it does in some respects reduce the need for component fatigue testing. It is intended to use the remains of the static test article. Some testing of this type has been completed, and will be discussed later in this report.

Phase 5 Elevated Temperature Testing

The growing importance of heat in aircraft structures requires more testing and development of a research nature. Creep and transient temperatures causing induced thermal stresses are important problems requiring extensive testing coupled with theoretical analysis.

PHASE 1 PRELIMINARY DESIGN TESTING

Both static and specimen fatigue testing is done during this phase. Where the problem is primarily one of stability or buckling, static tests have been used. Where the problem is primarily a problem of high stresses and stress concentrations, fatigue testing has been used.

In some cases the first test specimen proved to be quite satisfactory; however, in many of these tests development work was required and the final test specimen differed considerably from the original. In the case of fatigue testing, if the original specimen were considered inadequate, new and redesigned specimen were ordered, built, and tested. All specimen changes were then incorporated into the aircraft design.

For ease of assessment this phase of testing is broken down into aircraft components as follows:- Fuselage, Centre Section, Wings, Fin and Control Surfaces.

FUSELAGE

Intake Duct RT 08-242

A series of tests were inaugurated on the round portion of the .032 aluminum alloy duct. Test specimen include $13\frac{1}{2}$ ft. of the intake duct built to production standards by the Production Dept. (See Fig. 1).
The intake duct is subject to high pressures and depressions, pressures in flight and depressions during engine ground run-up. The problem of high depressions in long intake ducts during engine run-up has been a particularly difficult problem facing aircraft designers using large jet engines. The weight of such long large diameter ducts is very large, and a very close assessment, both analytically and test wise was considered essential.

Two identical specimen were ordered and tested. The point of initial buckling was the prime objective of the test and it was necessary to establish the difference between a duct that had been pressurized to 10 psi., and one that had come straight from manufacturing. Initial pressurizing blows the duct round and removes the worst of the flats and manufacturing discrepancies.

It was proven that initial pressurization of the duct to 10 psi., did in fact raise the point of initial buckling to a satisfactory level. The significant aspect here is in regard to panel flutter. If the panels were allowed to buckle at too low a point, they would almost certainly come apart due to panel flutter.

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PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

As a result of these tests it has been decided to pressurize each production duct to 10 psi., prior to engine run-ups. It may be possible to discard this process in favour of engine programming during ground run-ups and take-offs on later production aircraft.

Other objects of the duct tests are as follows:-

1. Leak rate.
2. To substantiate the strength of the duct under limit pressures.
3. To substantiate the strength of the duct under ultimate pressures.
4. The fail safe characteristics of the duct under pressure loads.

The results and conclusions are as follows:-

(a) The leak rate using normal rivetting techniques was unsatisfactory. As a result, all rivetted joints will be glued on production and static test aircraft.
(b) The duct satisfactorily withstood limit pressures. A factor was included to take into account the effect of temperatures.
(c) The duct satisfactorily withstood ultimate pressures with a factor included for temperatures.
(d) Fail safe characteristics (see Page 35).

Magnesium Fuselage Skin Panels in Shear RT 08-243

The large area of relatively low stressed skins on the fuselage forward of Sta.485 is most efficiently covered by a low density material. Magnesium was the obvious choice but the magnesium would still be required to work to a high degree of tension field. Very small (3/32 dia.) countersunk rivets have been used to eliminate the need for dimpling. Magnesium however increases the point of initial buckling and also increased the panel flatness. This is particularly important in regard to the panel flutter problem. (See Figs. 2 & 3)

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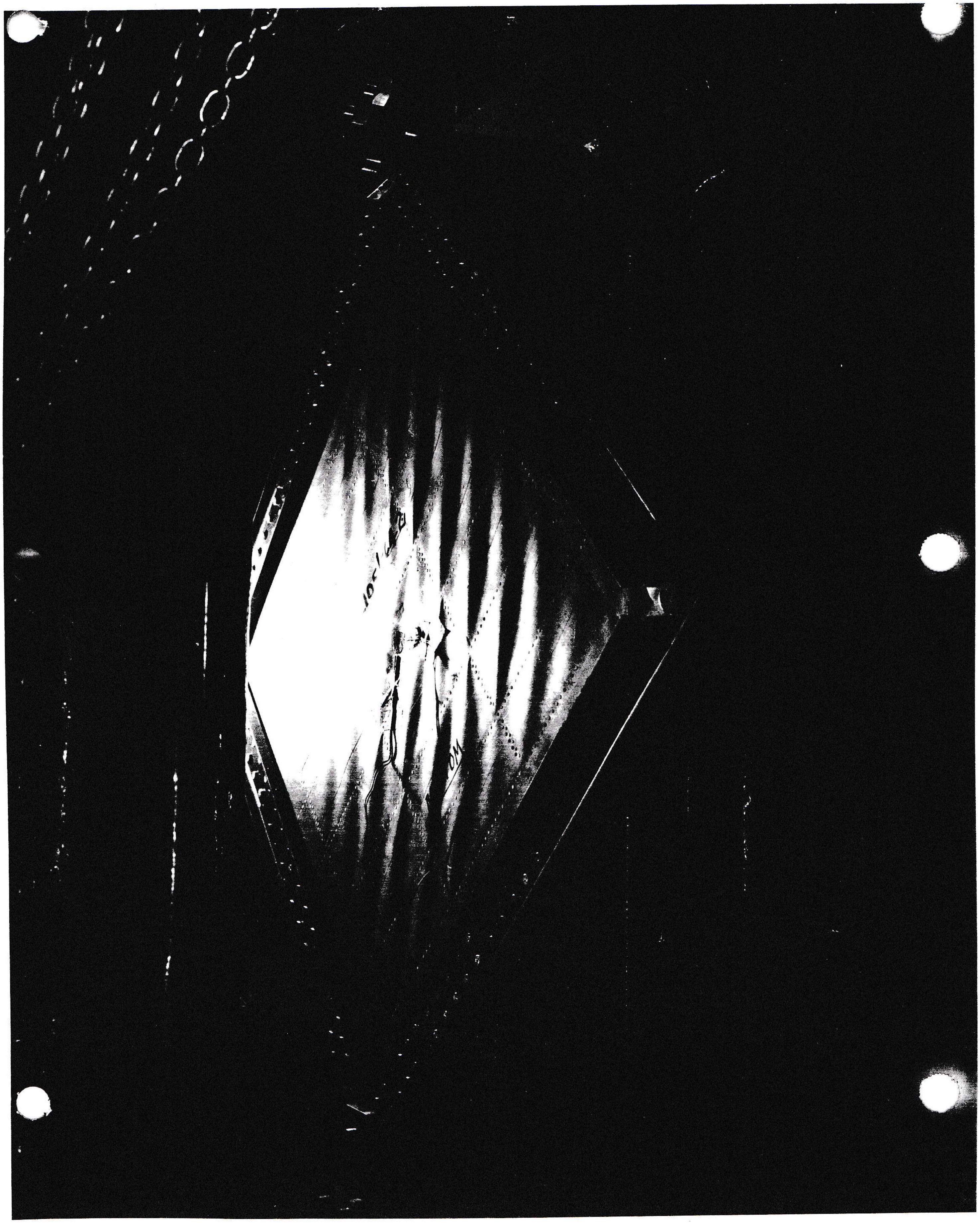
Sheet 4



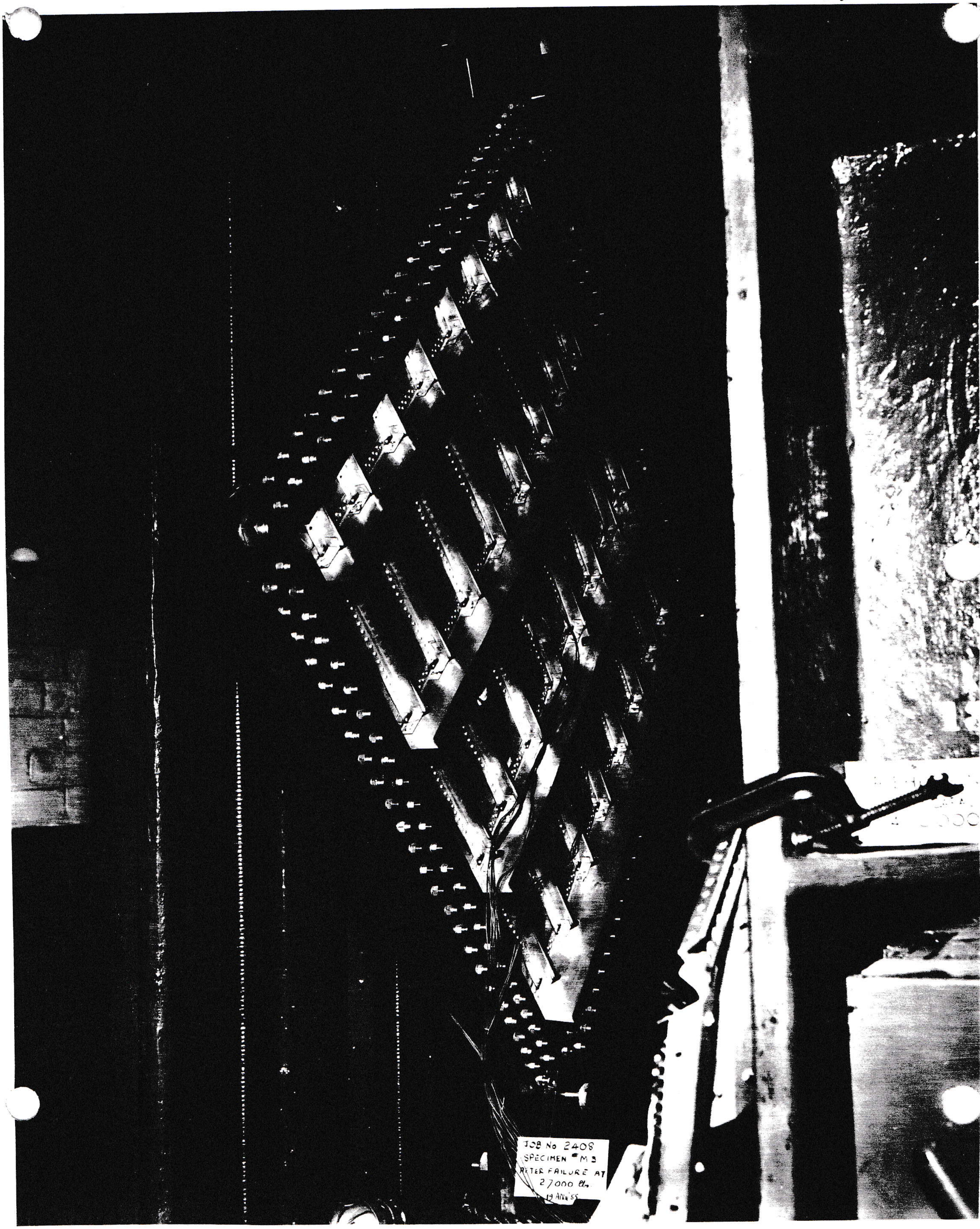


Fig. 5

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

A series of shear panel tests was inaugurated, with the following objectives:-

1. To establish the point of initial buckling.
2. To establish the strength of the stringers and frames that act to carry tension field loads.
3. To establish the limit allowable of the panels.
4. To establish the ultimate strength of the panels.
5. To establish a satisfactory rivet pitch.

Positive conclusions are not possible at this time, due to the non-availability of the proper magnesium sheet (ZE-41). However, some results were attained as follows:-

1. The point of initial buckling was attained. Initial buckling on the aircraft will occur at slightly better than 2g.
2. The stringers and frame strength was established as satisfactory.
3. The method of calculation was confirmed, enabling the limit load allowable to be accurately estimated.
4. Ultimate strength enabled confirmation of the method of calculation.
5. The very small countersunk (3/32) rivets at .5" pitch was found to be satisfactory.

This testing is to be continued using the proper magnesium, at room temperature and at elevated temperatures.

Magnesium Compression Panel Tests RT 08-379

Two panels representing the upper surface of the fuselage were manufactured and tested. The specimen were designed to represent three fuselage former bays (ℓ = 33") with the stringers supported at 11" intervals by channels designed to represent the former stiffness. The panels were identical except for a different rivet pitch. See Page 7 and Fig. 5 on Page 8.

The skin stringer, former combination proved to be entirely satisfactory with the maximum space of rivets tested.

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MALTON · ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO
SHEET NO 7

AIRCRAFT: C.105.

FUSELAGE STRINGER - SHEET COMPRESSION TEST (MAGNESIUM SHEET)

PREPARED BY: J.D. O'D
DATE: 14-12-54
CHECKED BY

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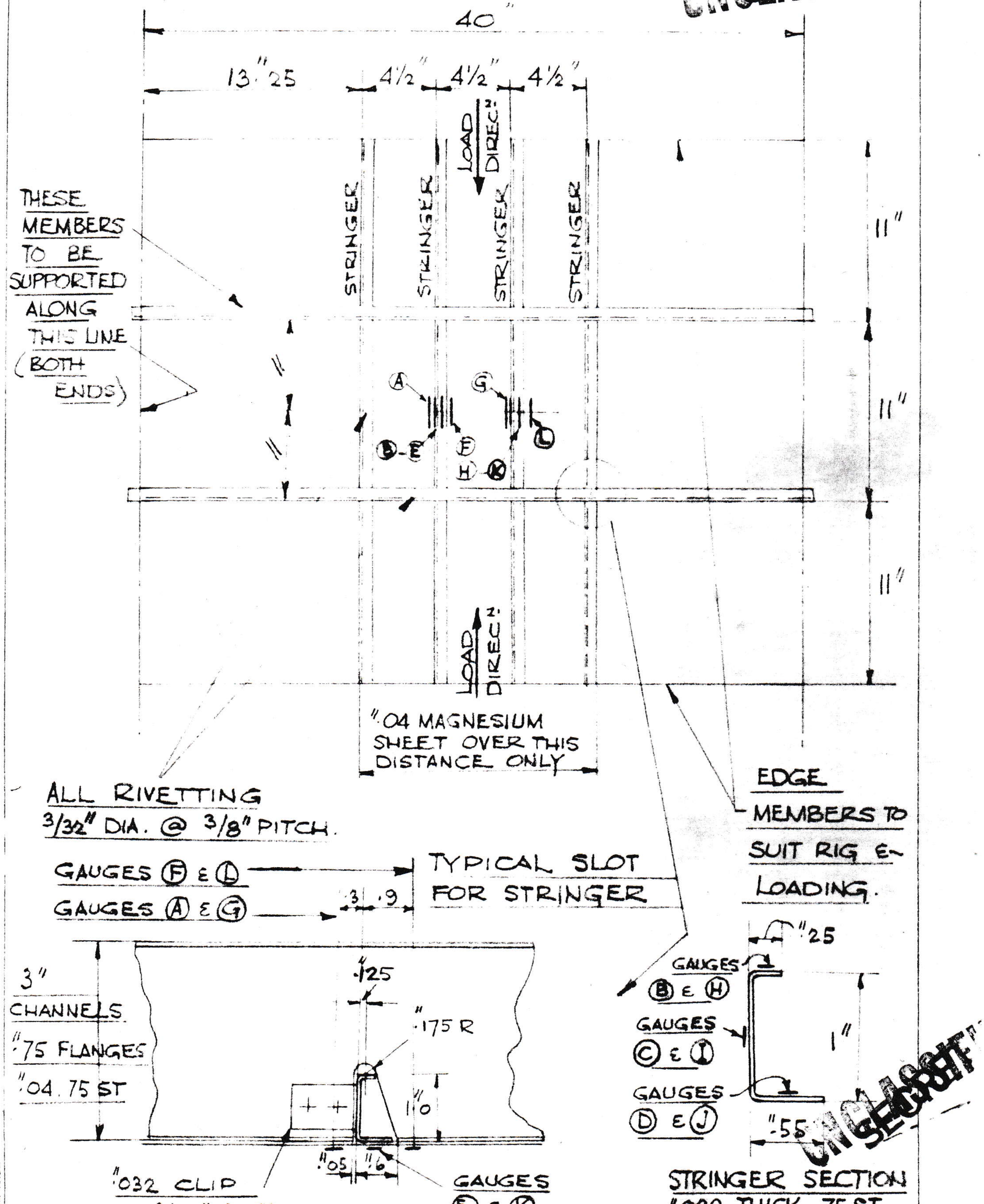

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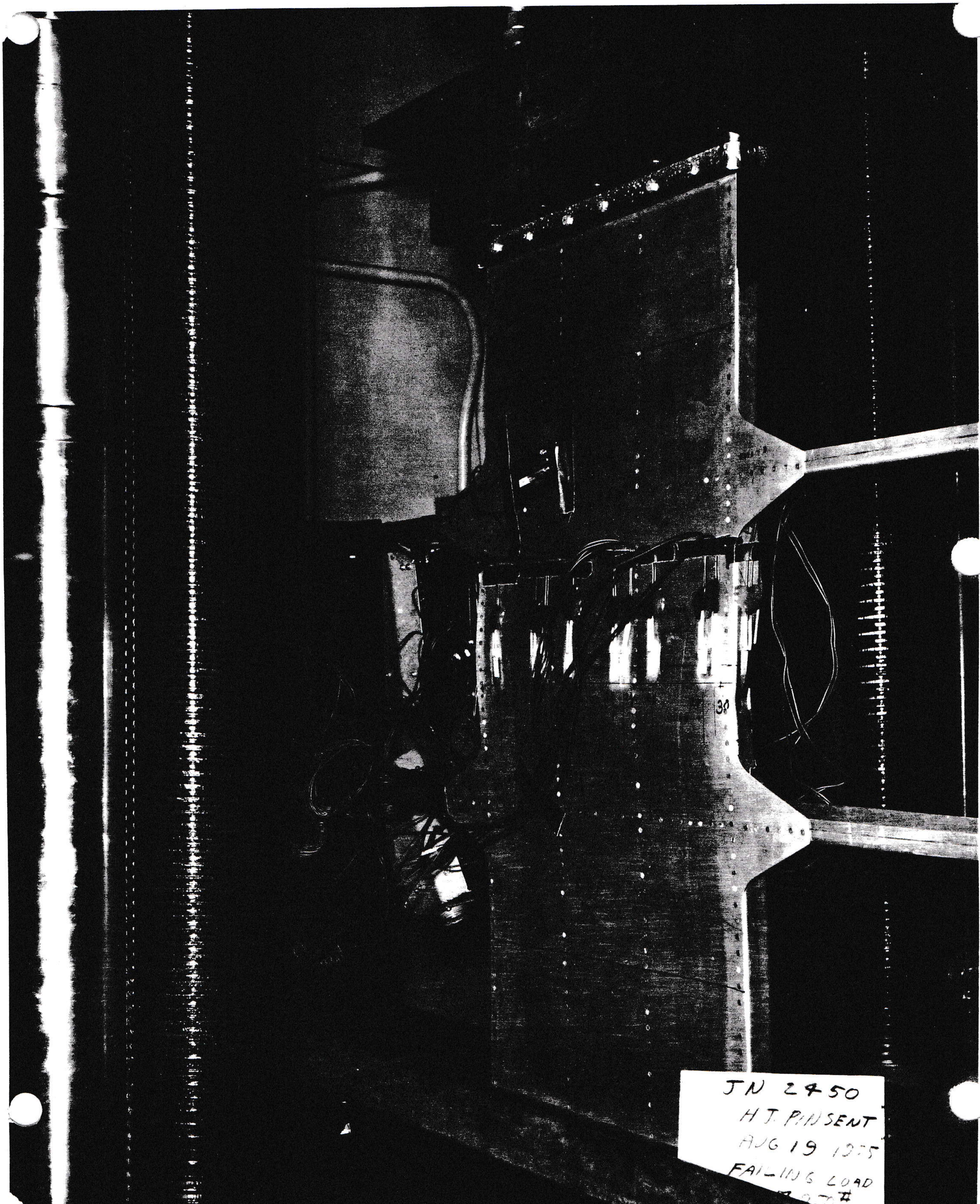

Fig. 5

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Fuselage Former Stiffness Test RT 08-244

The stiffness of the fuselage formers in the tank area is an important parameter in the fuselage load analysis and was found to be very difficult to analyse accurately. This is due to the complex shape of the former with odd shaped corners. Page 8 shows the frame that was tested.

Test results enabled a method of analysis to be developed that would give reliable stress distributions and stiffnesses.

Fuselage Fuel Tank RT 08-389

The fuselage tank is designed as a triple bubble with struts balancing the loads at the intersection of the perimeters. This scheme is discussed in the C-105 Structures Report. End bulkheads are designed with vertical and horizontal beams supported at shelves and by the tank skins.

A single tank specimen was designed and manufactured, incorporating the worst features of the final design. The tank skins are .032 aluminum alloy with rivetted joints. Tank liners are used in the actual aircraft.

The tank specimen was subjected to a limit and ultimate pressure of 18.5 psi and 27.8 psi respectively. The tank satisfactorily withstood both limit and ultimate pressures.

The tank was then subjected to a cycling pressure of from 0 to 18.5 psi. After approx. 10,000 cycles, cracks appeared in the tank door (See ATR 2457/2). Doors were repaired and throat washers added under the heads of the bolt to reduce the bending stresses in the flanges. At 25,000 cycles failure occurred in the top hat sections acting as a beam on the bulkhead.

This life was considered satisfactory.

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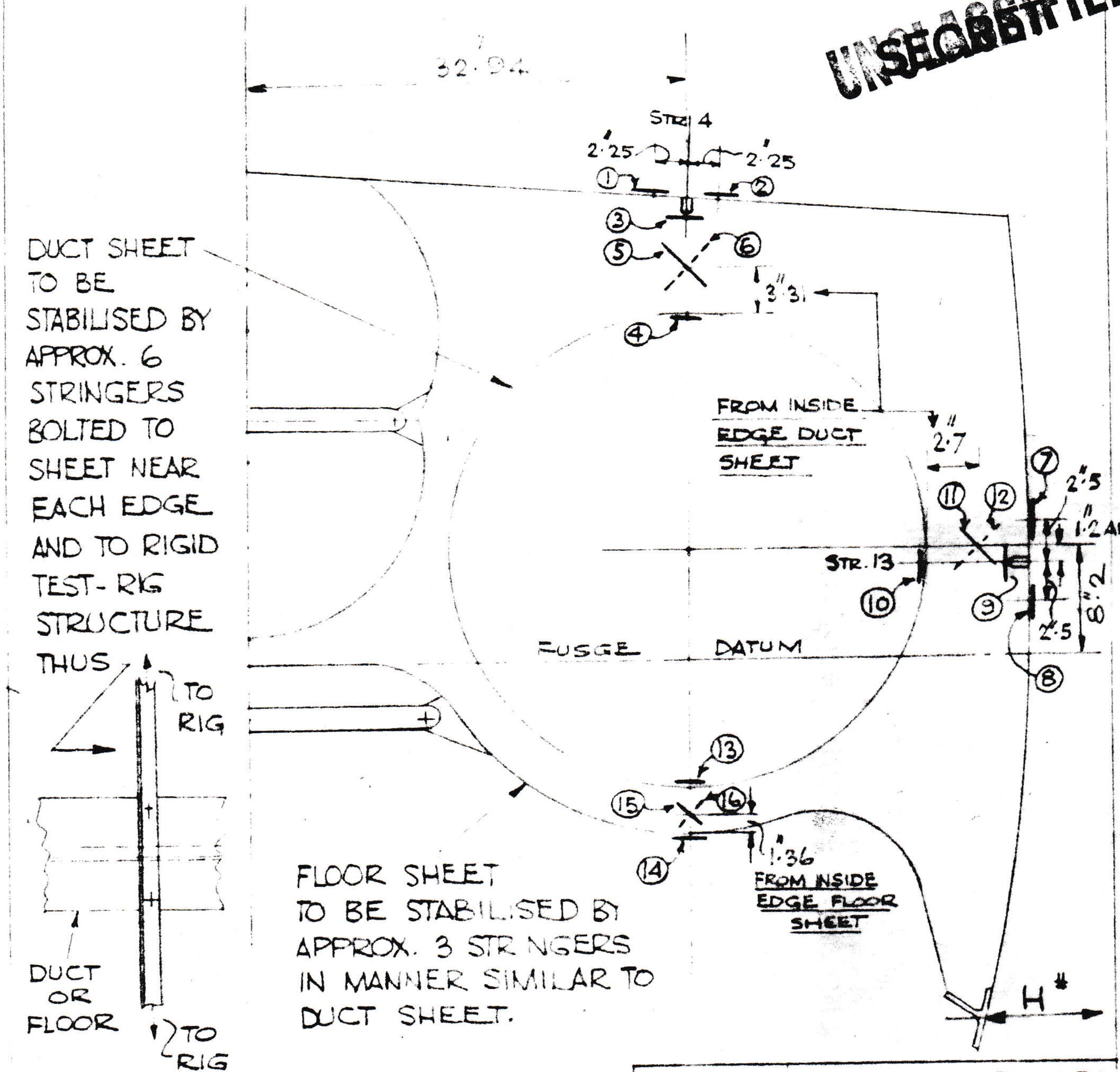
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C. 105.	ADDENDUM TO FUSELAGE FORMER STIFFNESS TEST.	REPORT NO	SHEET NO 10
		PREPARED BY: J.D.O'D	DATE: 19-11-54.
		CHECKED BY	DATE



1/ STRAIN-GAUGES SHOWN THUS 1
2/ GAUGES 1 2 7 8 & 14 ARE ON OUTER FACE OF OUTER SHEET PARALLEL TO FORMER
3/ GAUGES 3 & 9 ARE AT EDGE OF STRINGER SLOT AND PARALLEL TO OUTER PROFILE.
4/ GAUGES 4 10 & 13 ARE ON INNER FACE OF DUCT SHEET PARALLEL TO FORMER.
5/ GAUGES 5 6, 11 12, 15 16 ARE AT 45° TO DIMENSION LINES IN SKETCH AND EACH ONE OF A PAIR IS ON

GAUGE	EXPECTED STRESS
1 & 2	1740 PSI TENSION
3	2620 PSI TENSION
4	550 PSI COMPN.
5 & 6	160 PSI SHEAR.
7 & 8	14000 PSI TENSION
9	18180 PSI TENSION
10	14320 PSI COMPN.
11 & 12	2840 PSI SHEAR.
13	28030 PSI COMPN.
14	12020 PSI TENSION.
15 & 16	1310 PSI SHEAR.

VALUES OF STRESS ARE FOR
H = 1000 # ACTING INBOARD

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Windscreen & Pilot's Canopy Glass
Temperature Shock Tests RT 08-250

Preliminary tests have been requisitioned to evaluate the effect of thermal shock on the windscreen and pilots canopy glass. Two types of glass panel interlayers were proposed - a vinyl interlayer and a silicon interlayer. The silicon interlayer has as yet not been available, and all testing has been on panels with a vinyl interlayer.

It is expected that the silicon interlayer would be more satisfactory from a thermal shock point of view. The vinyl interlayer has the rather bad characteristic of having a large change in consistancy over our temperature range. At low temperatures it is very hard and at high temperatures it is quite soft. It is possible upon freezing a heated panel for the vinyl to freeze with the panels not in the equilibrium state. This would cause high local stresses in the glass panels sufficient to cause fracturing of the glass.

Because of the unpredictable nature of glass, cycling of the critical thermal shock case is mandatory.

The object of the test is to set up the panel in a representative manner and apply inside and outside temperatures in a manner representing the most critical thermal shock case. This whole procedure to be cycled 2000 times.

Although some preliminary shock tests have been carried out, the proper temperature cycling has not yet been started.

Glass Panel Strength Tests RT 08-489

In order to evaluate the glass panel variability factor, two panels were tested under equilibrium temperature conditions, one at room temperature and one at a case representing the 250° F boundary layer temperature.

In lieu of American requirements the test requirements of AP.970 Chapter 725 have been used as a guide. AP.970 requires a variability factor of 3 for tempered glass.

/continued ......

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Test results are as follows where variability factor is the test failing pressure divided by 2 times the working pressure of 6 psi.

At Room Temperatures
Failing pressure - 47 psi Ref. ATR 2515/2

Variability Factor $\frac{47}{2 \times 6} = 3.92$

At Elevated Temperatures
Failing pressure = 35 psi Ref. ATR 2515/1

Variability Factor = $\frac{35}{2 \times 6} = 2.92$

This appears satisfactory although it will be necessary to do some tests on production type panels. Photograph on Page 12 shows failed glass panel.

CENTRE SECTION (Sta. 485 Aft)

Stiffness of Light Formers RT 08-384

The prime purpose of this test was to confirm calculations regarding the stiffness of the light frames. As explained in the C-105 structures report, it is necessary, due to the distortion effects of the wing to have an accurate estimate of the frame stiffness. The structural aim for these frames was to keep the stiffness as low as possible, compatible with static load requirements, and the allowable stresses as high as possible. This is particularly difficult where light gage rolled sections are used.

The measured stiffnesses agreed very well with calculations, and allowed the calculations of induced stresses to proceed on a sound basis.

The secondary purpose of this test was to check the stability of the former flanges. A deflection of 1.63" was achieved before failure of the frame which was far short of the 2" deflection required. The series of tests conducted clearly showed the importance of good workmanship. Although the 2" deflection is theoretically possible it was decided to use the 1.63" test result as a practical limitation.
Frames were altered accordingly.
Photograph on Page 14 shows a test set-up.

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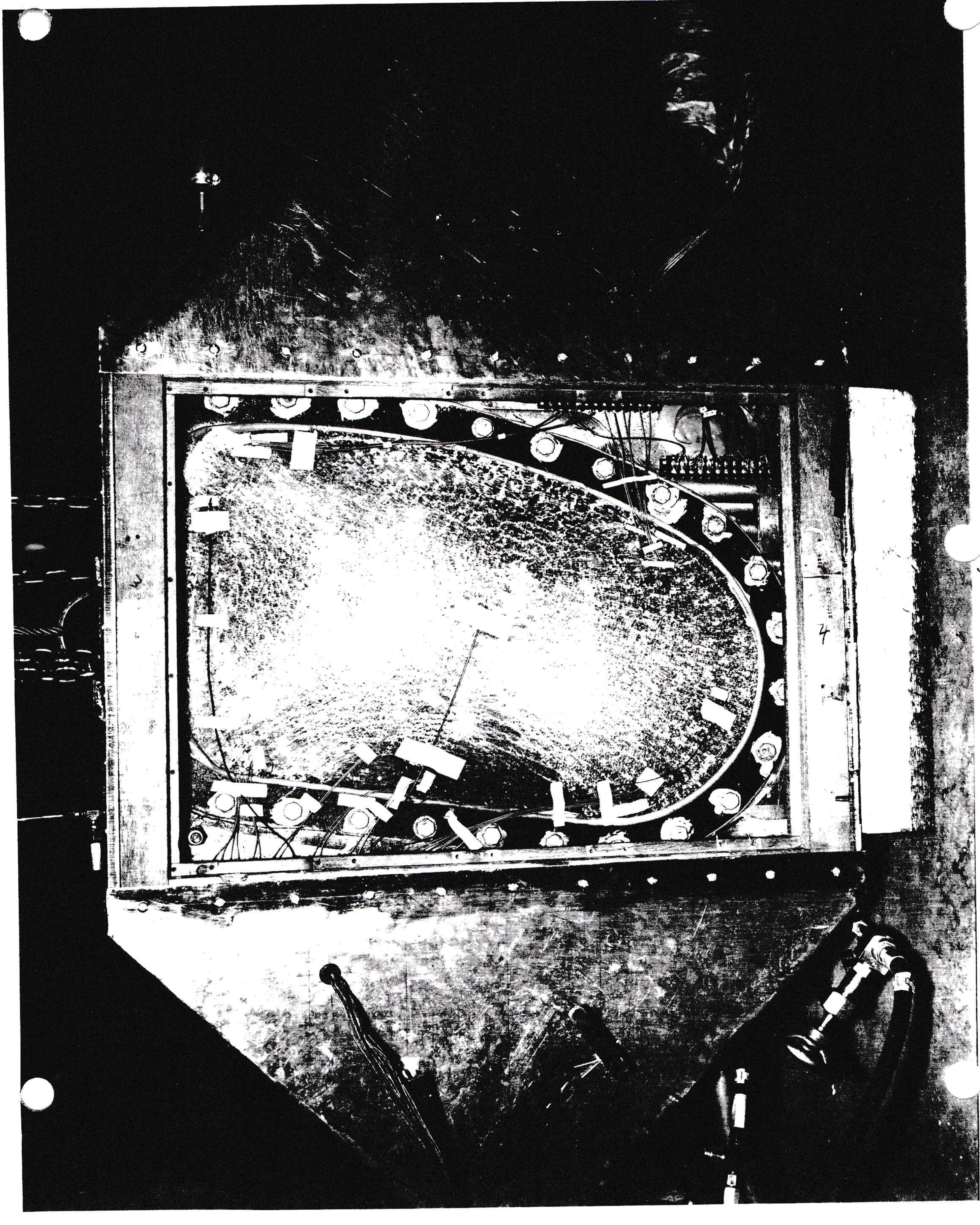

Fig. 6

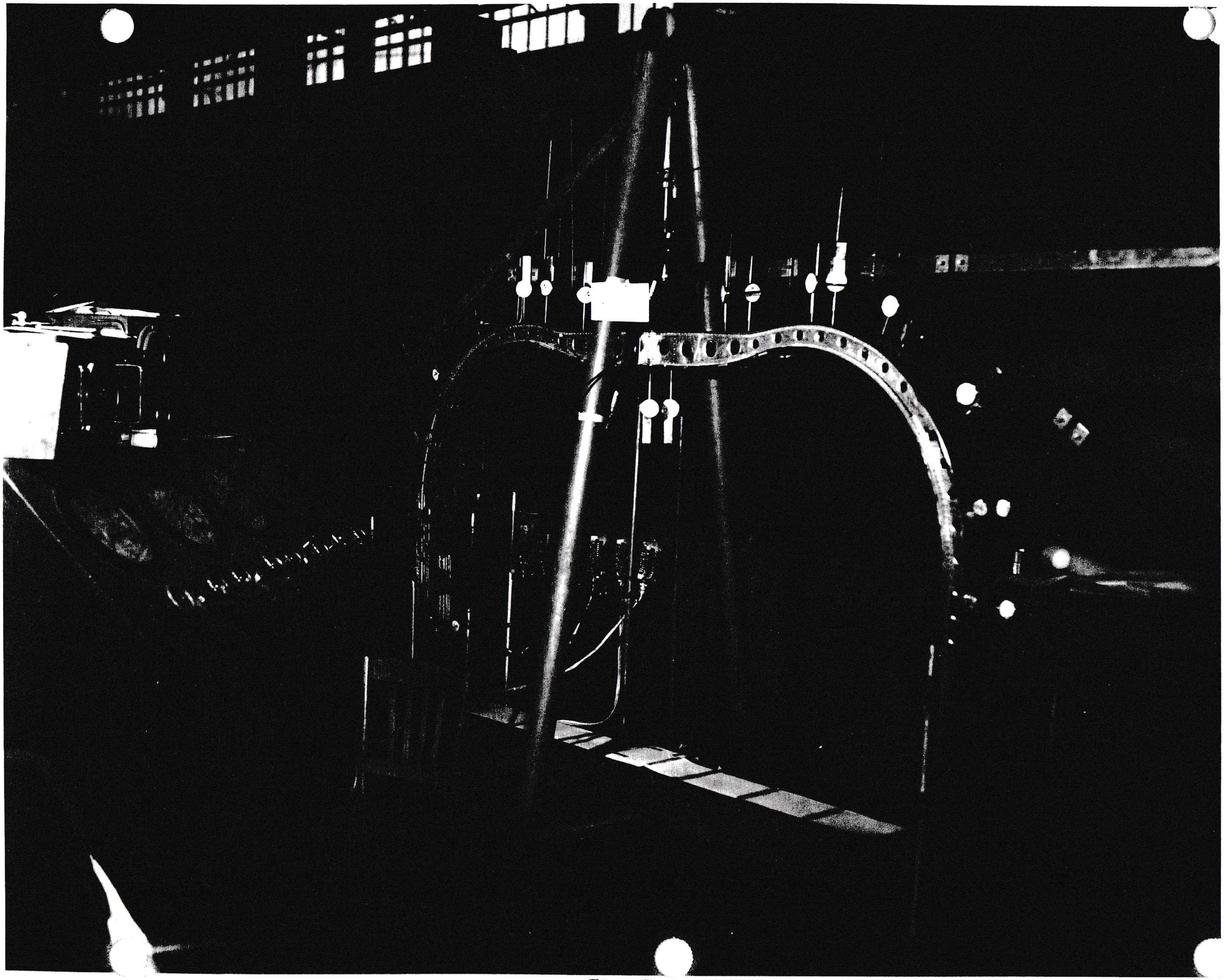

Fig. 7

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Light Frame Stabilizing RT 08-454

Further tests were conducted on the light frame sections to evaluate the rolling of the flanges caused by the radius of the frame. This effect of rolling of the flanges reduces the stability of the frame. Although limit conditions were met, the ultimate failure of the frames fell about 10% below the objective.

The light frames were altered to show a positive margin using test results. Photograph on Page 17 shows the test set-up.

Main Frame Stabilizing RT 08-485

As in the above test, the rolling effect of the frame flanges causes local stresses and reduces the stability of the frame booms.

A specimen representing a section of a heavy machined frame was manufactured and subjected on test to a bending moment.

From tests, a failing stress of 45,000 psi was attained. This very closely approximated the calculated crippling stress.

No design changes were required. Photograph on Page 18 shows test set-up.

Side Skin Shear Panel Tests RT-08-243

The side skins aft of Sta.485 are more highly loaded than the skins forward at Sta.485. Not only are the panels a much heavier gauge (.051 and .064, - 75ST clad) but the aspect ratio of the panel is very large. The loads on the edge members are quite severe on this type of panel. Testing was required primarily because of the edge member. Two types of edge members were tested - an angle extrusion and a lipped rolled section. Both edge members proved to be satisfactory, allowing the skin to work up to a nominal shear flow of 31,000 psi. One panel failed prematurely, due to excessively large rivets attaching the skin to the edge members. Fig. 10 shows panel tested.

Side Skin Access Door Shear Test RT 08-476

This test is further to the side skin shear panel tests. It uses the same rig and the same size panel, but the panel incorporates a screwed on access panel (approx. 10" x 10").

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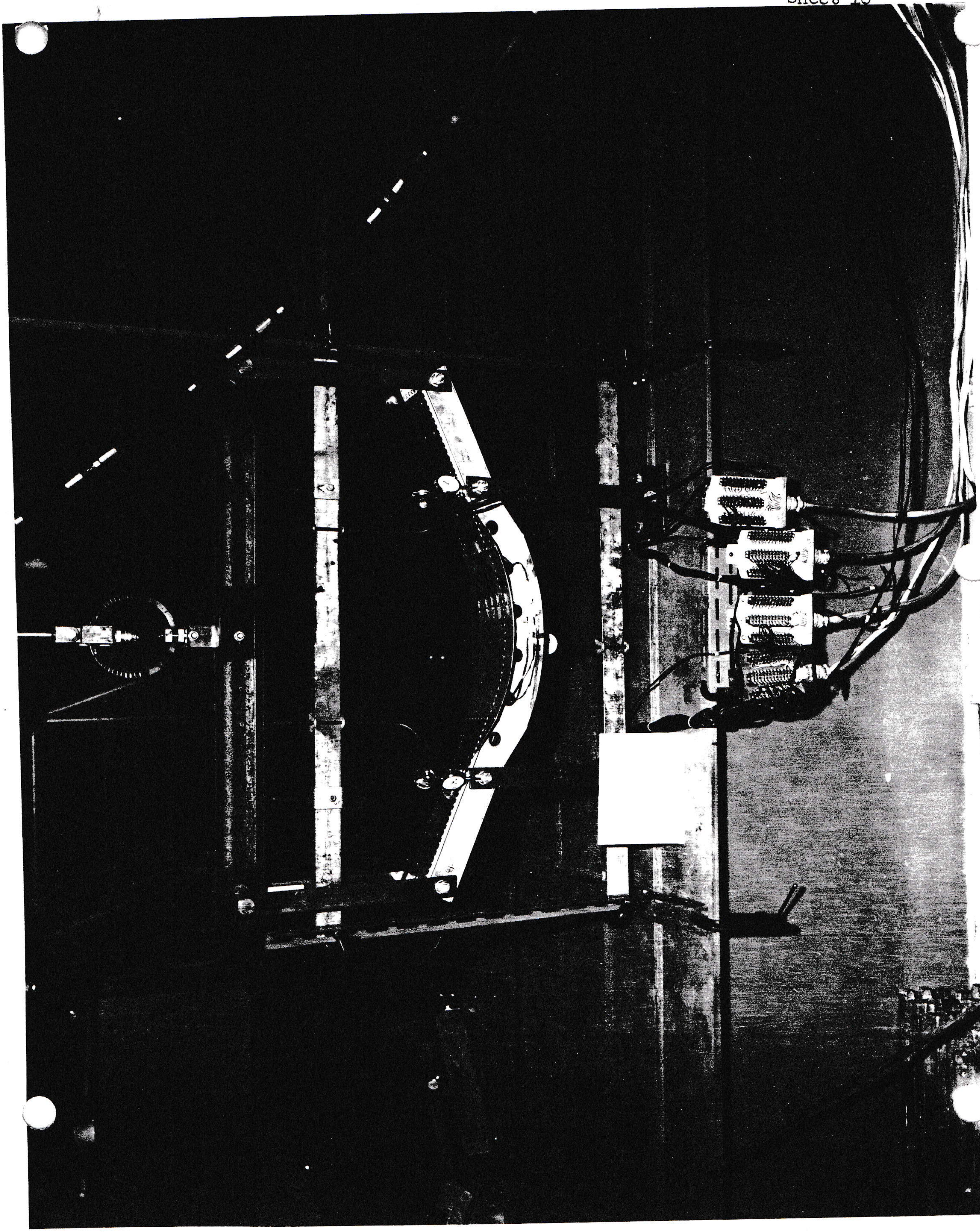
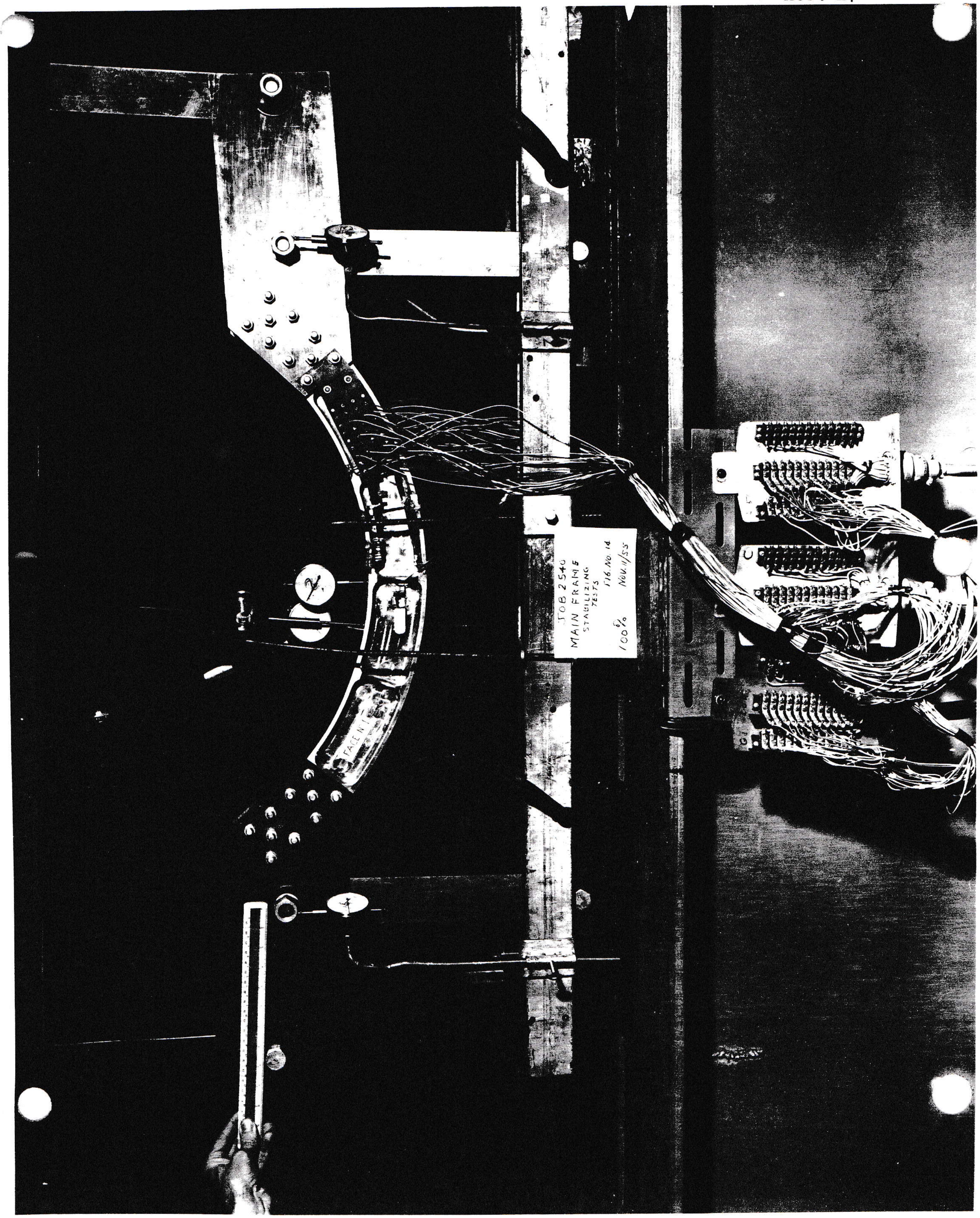
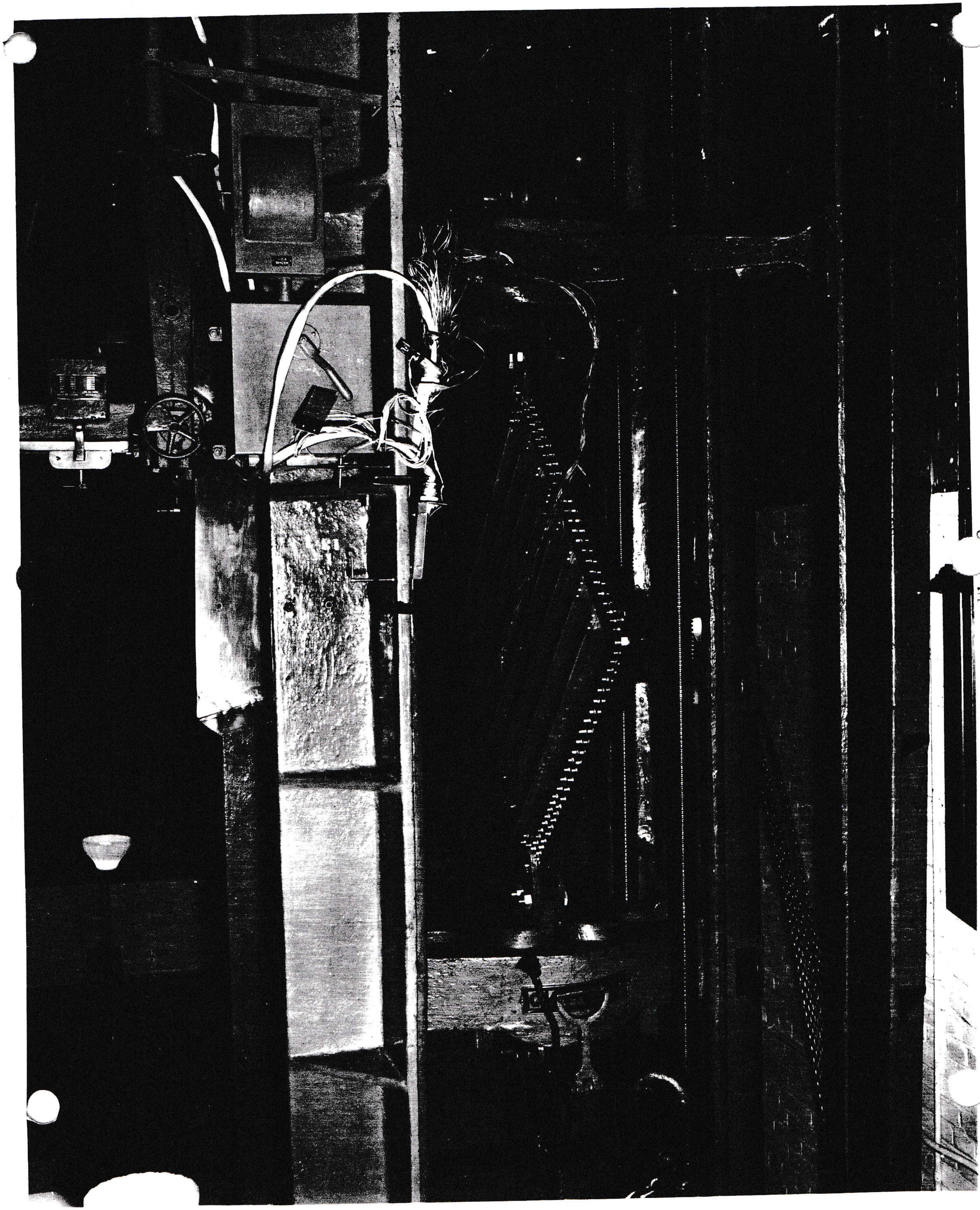


Fig.

Sheet 17



Sheet 18



PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

The object of test is to assess the effect of a stressed panel on the overall stress distribution. The strains in the door will be less than the side skins due to slippage in the bolted joints. Also, the effects of load cycling on the strain distribution is to be obtained.

Representative edge members are used.

Engine Intake Duct (Floating Assembly) RT 08-310

The aims and objectives of this test are identical to the fuselage intake duct tests. The test specimen will be exactly representative of the aircraft. At this time it is suggested that this test could very well fall within the scope of Phase No. 2 testing.

Fatigue Test of Light Former Joint
at Lower Longeron RT 08-279

Both the heavy and light frames are cut by the lower longeron. This has caused a rather difficult detail problem in splicing frame shears and bending moments across the joints. Local offsets and flange joggles reduce the fatigue strength.

The specimen to be tested will incorporate a short section of the longeron and part of the frame. Results should be applicable to all light frame joints at the lower longeron.

Fatigue Test of Heavy Former Joint
at Lower Longeron RT 08-278

As above.

Longeron Joint at Sta. 485 - Fatigue Test RT 08-268

This joint is the structural attachment of the lower longeron between the nose fuselage and the centre section. Ultimate design load for this joint is 116,740 lbs. Joint is made up of stepped titanium splice plates attached to 75 ST extrusions.

Fatigue tests are considered necessary to properly assess this joint.

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AVRO AIRCRAFT LIMITED MALTON ONTARIO TECHNICAL DEPARTMENT		REPORT NO.	
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		CHECKED BY	DATE

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Fatigue Test on Wing to Fuselage Hinge RT 08-4[illegible]

The problem of a hinge carrying shear load is that due to adverse tolerances the load distribution on the lugs will be uneven causing some lugs to be overstressed.

Several sections of hinges were manufactured to drawing tolerances and fatigue tested in a machine. The hinges proved to be satisfactory.

Engine Shroud Test RT 08-560

The engine shroud is subject to pressures and depressions in flight, and at the same time it will be distorted in shape by the flexing of the wing and centre section frames.

The section of shroud being tested is 66" long and is made up of .018 Al. Alloy with light stiffeners. The stiffeners are very shallow in depth due to space limitations.

The stability of these stiffeners when subjected to suction while distorted is critical. The manufacturing tolerances could also have a significant effect.

The test rig is so designed as to apply pressures and distortions in a representative manner.

Upon completion of suction tests, the specimen will be tested to cyclic variation of pressure (18.0 psi to 0) and deflections.

Multipost Stiffened Box Beams as used on the Inner & Outer Wings RT 08-333 and RT 08-240

It would be well here to briefly describe the problem.

The use of multipost stiffening results in greater structural efficiency as opposed to ordinary longitudinally stiffened sheet or multiweb stiffening when the initial conditions require that the covers must withstand load in both directions in the plane of the cover as well as shear. The posts are obviously also well suited in helping to contain pressure normal to the cover.

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PHASE 1 PRELIMINARY DESIGN TESTING (contd.) UNCLASSIFIED

Since, in a well designed posted box, the compression cover will buckle with longitudinal nodes along the stringers, this type of construction behaves up to buckling the same as a multiweb box. The main difference between the two is their behaviour after buckling. Even when the buckling stress is low, which infers a high peaking stress over the stringer, it is quite practical to have a posted structure behave as well as a multiweb structure, since the failure will almost always be one of local instability, or stringer effectiveness in containing the skin buckling.

In order to have the skin and stringer in a conventional stringer stiffened skin work to the higher stresses, which indicate higher efficiencies, the stringer must be very robust, be very closely spaced, and have a reasonably short column length. This type of stiffening, then, requires that a great deal of the bending material works in one direction only, and also necessitates rather large cutouts in the ribs to provide for the stringers.

The behaviour of the post stiffened skin can be predicted up to buckling using NACA TN 3118. The post-buckling behaviour can be roughly checked considering the following points:-

1. Reduction of buckling stress due to interaction of shear and chordwise stresses with the longitudinal stresses.
2. Compression stability of the stringer-skin column between posts under the peaked stresses after buckling.
3. Local stability of the elements of the stringer, paying particular attention to bending stresses due to normal pressure.
4. The effectiveness of the stringer as a shear panel stiffener.

However, these checks are only approximate and must be corroborated by test. The tests prove the following points:-

1. The rib spacing is sufficient to have the theory of TN 3118 apply.
2. The support given by the posts and tension cover continues to be sufficient after buckling.
3. The torsional stiffness of the stringer and the bending stiffness of the post are satisfactory to prevent torsional instability of the stringer.

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PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

4. The theory of TN 3118 can be extended to more than one row of posts between spars.

The post stiffened boxes that were tested were capable of working to an average pure compression stress of 50,000 psi. These boxes had an equivalent skin gauge including skin, stringers and posts of .246 inches. A comparable skin having an equivalent gauge of .254 inches with only longtiudinal stiffening was found by previous testing to be capable of working to a stress of 42,000 psi. This shows an increase in efficiency of 12%.

The posts do not in themselves improve the shear carrying capacity of the skin but since this type of stiffening allows less longitudinal stiffening, more material for the same weight can be put in the skin, thus improving the shear capacity.

Three outer wing boxes were tested with varying combinations of applied torque and bending to fully evaluate the interaction effects. During these tests the posts were reduced in size. It also became apparent that the stringer section could be slightly reduced and a fourth box using a reduced area stiffener was tested and found to be satisfactory.

Two inner wing boxes have been tested with and without fuel pressures. Results from both panels were satisfactory, and no development work was found to be necessary. Photograph on Page 24 shows a box being tested.

Inner & Outer Wing Compression Tests RT 08-230

Preceding the full box beam tests, several compression panel tests were conducted to closely evaluate the column stability characteristics. Data obtained from these tests aided the design of the box beams. Fig. 12, Page 25, shows the type of skin buckle obtained.

Elevator Stiffness and Limit Load Test RT 08-497

An elevator and trailing edge mounted on a flexible beam representing the wing, was manufactured primarily for control system testing.

However, it was important to carry out load tests before the system testing to check the load distribution in the elevator links. If the load distribution was unsatisfactory it would invalidate the control system testing.

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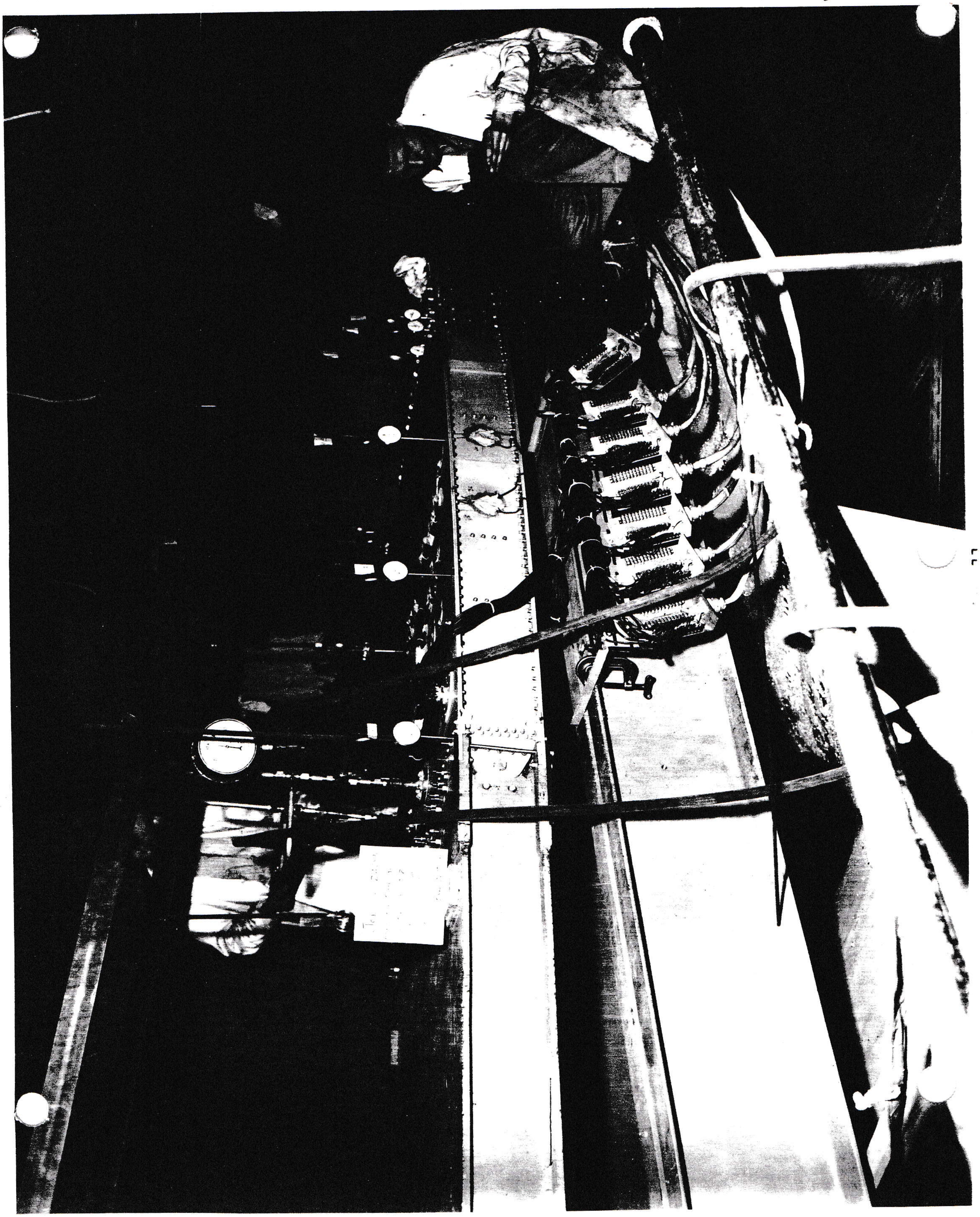

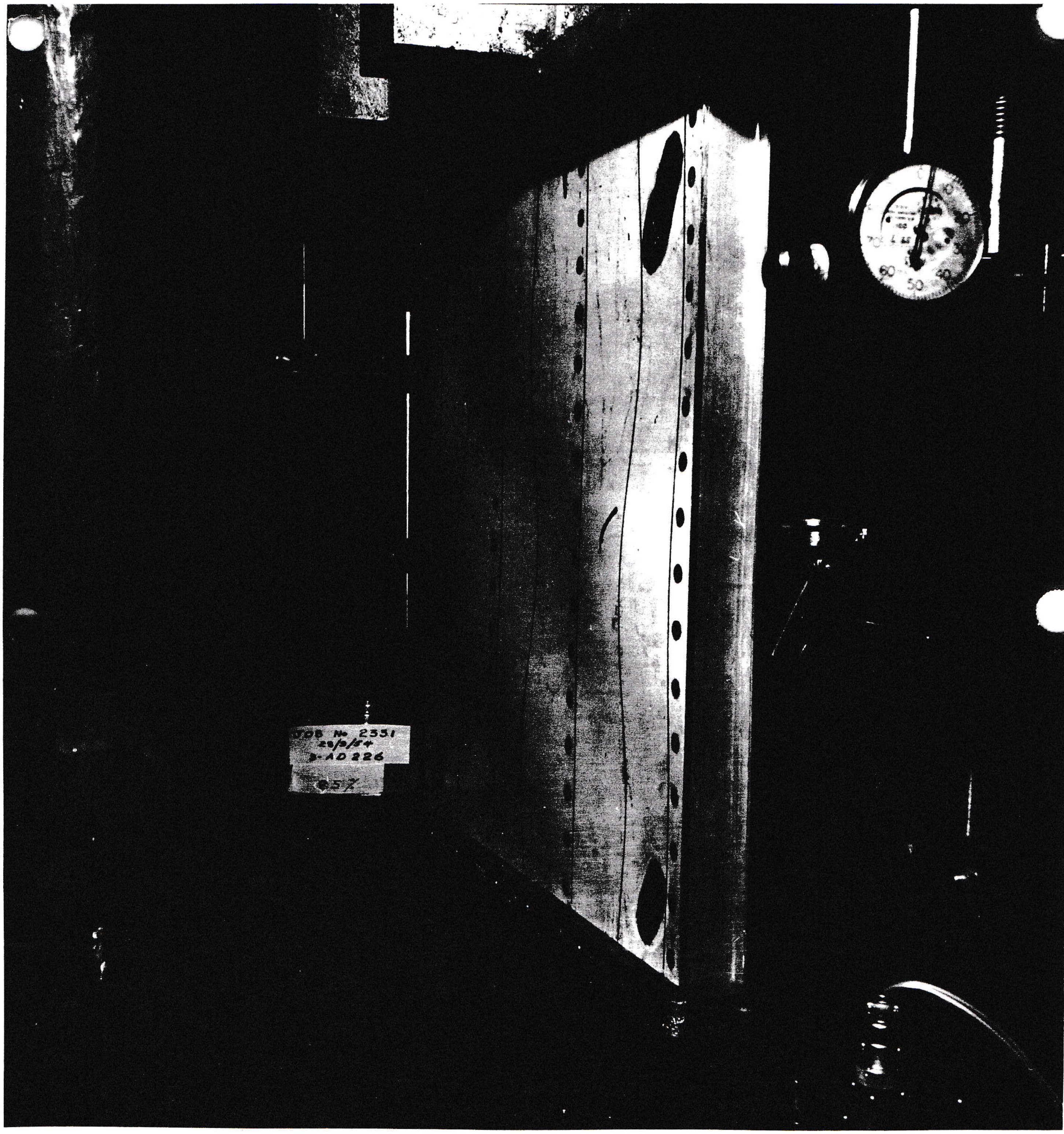

Fig. 12

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

In the initial seriesof tests it was found, from strain gauges, that the inboard link was more heavily loaded than it should be. This was most unsatisfactory from the point of view of bearing wear.

The inboard steel lever was realized to be much too stiff, being rather hurriedly designed for the test specimen. This extra stiffness in this lever had the effect of increasing the load in the inboard link. The lever was reworked to remove the excess material, and a slightly revised aerodynamic load distribution was also used. This revised distribution catered to some tip loss, which helped to reduce the load level in the inboard link.

The tests were resumed and the resulting load distribution among the links was considered satisfactory. Tests were carried out with the wing bent and with elevator neutral and deflected. Photograph on Page 27 shows the test set-up.

The limit load strength of the elevator, trailing edge and linkage, proved to be satisfactory.

Aileron Stiffness & Limit Load Test RT 08-497

The same tests as were carried out on the elevator will be conducted on the aileron and aileron trailing edge.

Shear Test of High Strength Fasteners (some with sealing grooves) RT 08-365

The wing torque box is fastened by a multitude of fasteners. The performance of these fasteners, especially in regard to deformation under limit loads is of the utmost importance to the satisfactory structural performance of the wing. Any undue bolt slip or joint deformation could cause permanent set to the wing of a severe nature.

The test rig was designed as a large circular plate with the joint being tested at the periphery. This type of test set up applies pure shear to the joint.

All types of fasteners and joints used on the inner and outer wing have been tested, and limit and ultimate allowable loads obtained. This data was used to size all the joints subject to shear.

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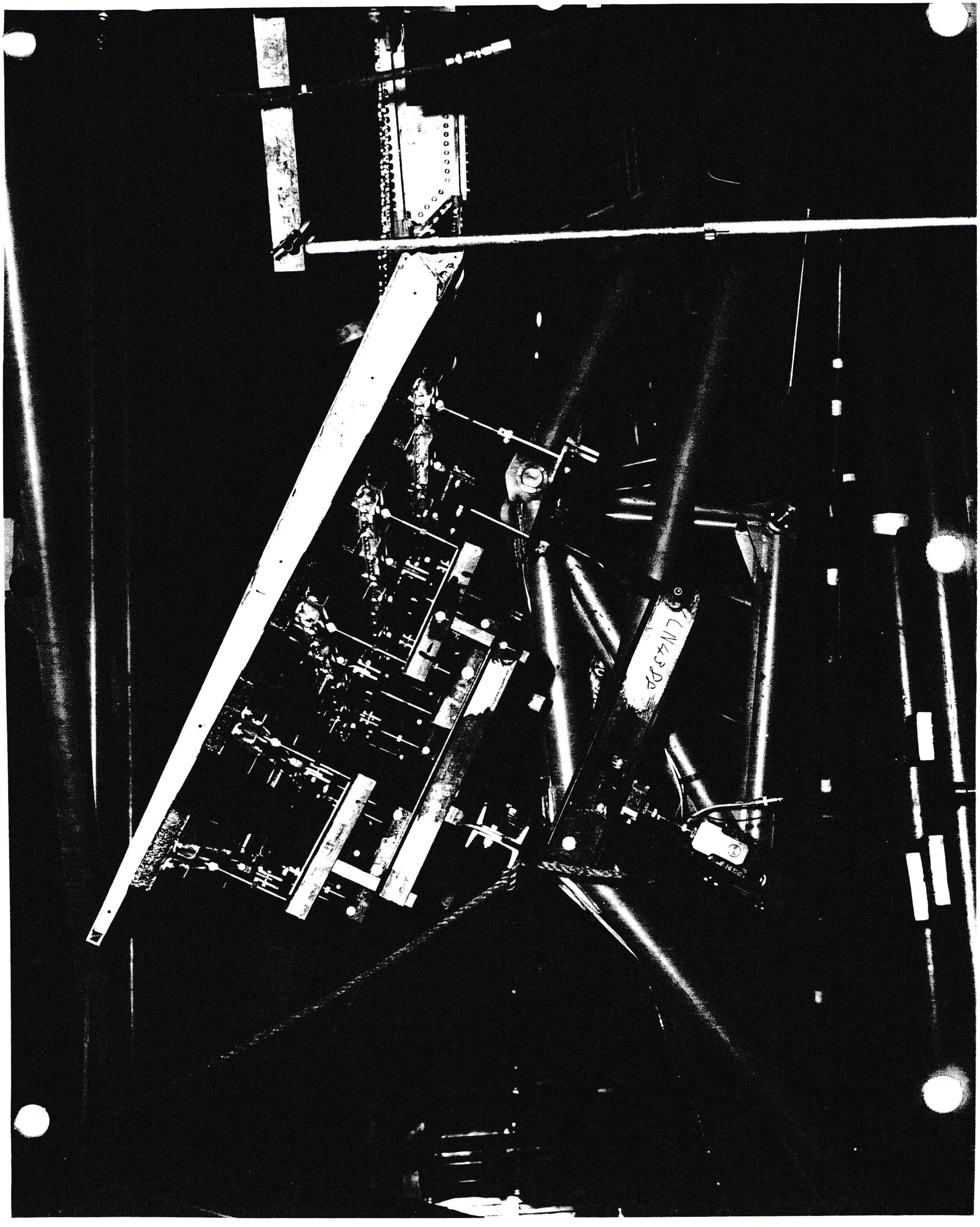
LN43PP

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Static & Fatigue Tests - Skin Splices RT 08-276

It was realized early in the design stages of the wing that the ability of the skin lap joints to take chordwise loads was very much in doubt. Early fatigue tests of typical joints confirmed our fears, and as a result a programme was set up to fatigue test various types of joints using various types of fasteners.

As a result of these tests, design allowables were obtained which when used to stress the joint, would give a satisfactory fatigue life.

The C-105 Structures Report gives a more detailed description of types of joints tested and the resulting S-N curves. This report does not however go into the effect on the joint of various fasteners. Report 7/0500/9 gives full details of this series of tests.

Static & Fatigue Tests of Transport Joint RT 08-261

A fatigue test specimen will be designed and manufactured, representing a 5" width of the transport joint. The specimen will be designed for testing in a fatigue machine.

No testing has as yet been carried out.

This is in line with the basic policy to specimen fatigue test critical joints.

Fatigue Tests of Elevator Links RT 08-262

Seven links with bearings will be fatigue tested in a fatigue machine at room temperature and at elevated temperatures. The purpose of this test is to assess the fatigue strength of the lug bearing combination.

Engine Mount Fittings - Fatigue Testing

Due to the critical nature of fatigue in engine mounting structures most of the fittings will be machine fatigue tested.

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PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Some of the links due to design requirements in connection with engine removal cannot be designed to adequate fatigue standards. Fatigue testing in these cases will be necessary to establish a replacement life. Where the fitting is not easily replaceable, it obviously must meet the required standards. Several fittings will be tested but since no results are available at this time, the tests will not be enumerated here.

Strength Test of Typical Outer Wing Rib RT 08-546 & 08-551

This test complies with the requirements of Spec. MIL-S-5710, Para. 4.5. It has been included in this phase of testing since it is required to substantiate the design preceding the main aircraft structural tests.

Five ribs have been designed incorporating the worst structural features of both the inner and outer wings.

Tests are required to assess the effects of stringer cut-outs, splices, rib cut-outs for equipment, and to comply with the requirements.

FIN

Fin Posted Box RT 08-241

Several posted boxes representing the fin torque box were manufactured and tested in the same rig used for the inner and outer wing torque boxes. The specimen represented in all respects the actual fin box, i.e., ribs, stringers, skins and posts. The skins were not exactly representative since the fin box uses taper rolled skins. However, two skin gauges were used - .188 and .156. This covers fairly well the more critical area of the fin. Considerable development work did take place to obtain the most efficient combination of stringers and post. Photograph on Page 29 shows a fin box under 90% of ultimate load.
Final boxes tested proved to be entirely satisfactory. Interaction of shear and bending moment was considered satisfactory.

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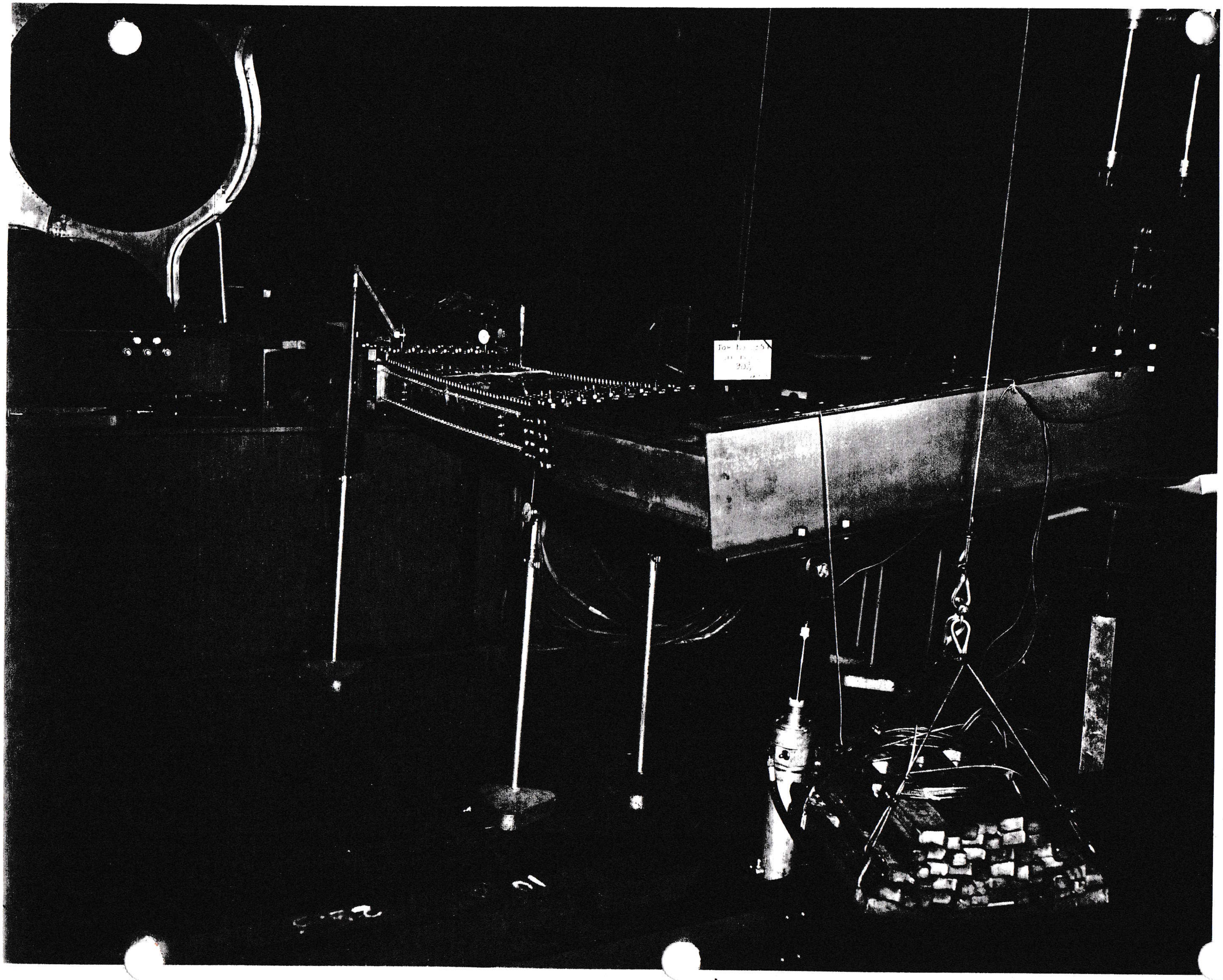

Fig. 14

PHASE 1 PRELIMINARY DESIGN TESTING (contd.)

Fin Rib Shear & Bending Strength Test RT 08-496

Two specimen representing the range of fin ribs were tested in order to comply with the requirements of MIL-S-5710, Para. 4.5. and to check the crippling of the web with the stringer cut-outs and rib lightening holes. The fin ribs are much less robust than the outer or inner wing ribs and much more susceptible to panel crippling failures.

The ribs satisfactorily withstood the design limit and ultimate.

Failing loads were approximately 8% higher than expected.

Rudder Stiffness & Limit Load Tests RT 08-474

This test is similar to the strength tests carried out on the elevator. Again it was a matter of using the control system test set-up for preliminary structural tests. Due to the time element these tests may be carried out on the actual static test aircraft. This is considered to be satisfactory.

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PHASE 2 PROOF OF COMPLIANCE

Most of these tests will be carried out using the full aircraft static test specimen. The purposes of these tests are as follows:-

1. To comply with the requirements of MIL-S-5710.
2. To substantiate load distribution.
3. To confirm stiffnesses and distorted shapes.
4. To confirm limit load requirements.
5. To confirm ultimate load requirements.

The C-105 aircraft must be tested in the proper environment i.e., it must be a complete aircraft. The interaction effects and the integral nature of fuselage, centre section, wing and fin absolutely dictate this policy. There can be no satisfactory component testing as we were able to do on the C-100 aircraft. A possible exception is the dive brake.

Although these tests are primarily to prove the structural integrity in line with the requirements, the initial tests have been simplified in order that some testing will be completed before initial flight trials. These initial tests as described in the following text are planned to clear the aircraft for initial flight testing only.

Tests are described in the planned order of procedure.

Main U/C Spring Back Static Test RT 08-246

A large area of the wing around the undercarriage cut-out is critical for the undercarriage spring back case as well as the landing gear itself. Although in importance this test is secondary to the rolling pull-out case, it is definitely required before flight. Since the aircraft rigging for the R.P.O. case will support the aircraft for the landing case, this test does not seriously disrupt the R.P.O. case. A six week set back of the R.P.O. case estimated, is considered satisfactory.

Approximately limit load will be applied, and strain gauge readings, and wing and undercarriage deflections will be recorded.

Complete proof of compliance tests will be conducted at a later stage in this programme.

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PHASE 2 PROOF OF COMPLIANCE (contd.)

Rolling Pull-Out Case RT 08-245

See Reports 7/0500/7 Issue 2 - Main A/C Static Tests
7/0500/8 - Test Loads R.P.O. Case

Since the above referenced reports record in detail the tests, it will be sufficient here to discuss only basic philosophies of the test.

The most difficult analytical problem is the interaction of wing, centre section, and fin. This area of structure then represents the greatest threat to structural integrity, and therefore must be tested first. Also a test of this nature strains the complete structure to a reasonably high factor of its design load. For example, parts of the wing will only be loaded to about 50% of limit load, but this is still far better than testing the symmetric cases, which would test the wings to limit load, but would not apply any load to the fin.

A completely balanced and final aerodynamic R.P.O. case was not available. Again certain simplifying assumptions were made by the aerodynamic department to obtain a balanced aircraft. To get some testing done early was the prime motivating force.

Cockpit and Fuel Tanks will be pressurized.

Intake ducts will not be pressurized.

Airload distribution over the fuselage and centre section will not be well represented. This is only significant in the structure aft of Sta. 485 and below the wing where the effect of airload distribution and internal pressures are significant.

Symmetric Cases (Limit Load Tests)

It is planned that two symmetric cases will be tested following the R.P.O. case. Cases will be a pitch case and a no-pitch case with different centre of gravities. These tests are in the detail planning stage.

A multitude of deflection and strain gauge readings will be recorded.

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PHASE 2 PROOF OF COMPLIANCE (contd.)

Sufficient automatic recording equipment has been ordered to allow us to analyse completely the full set of strain gauge readings between each load level assuming that we will actually test one load level a day.

Aircraft Drop Tests RT 08-247 & 08-246

It is proposed that final proof of compliance in regard to the undercarriage - structure combination will be demonstrated by drop tests of the complete aircraft designed to represent the actual design rate of velocity.

The most difficult analytical problem in regard to the undercarriage is the problem of dynamic coupling of the undercarriage with the wing structure. If we were to static test the gear-wing we would have to apply calculated loads that would not be checked. The obvious answer is to drop the gear which will check the dynamics as well as the gear-wing strength.

Drop testing the gear alone will give completely incorrect results since the flexibility of the wing cannot be simulated.

The aircraft will have to be dropped at least three times to cover the full range of cases. Wheels will be spun-up to properly represent the spin-up cases. Steel plates or equivalent will be used to obtain the proper coefficient of friction.

Many tests of a more minor nature will be carried out on the test article before the ultimate load test such as the following:-

Cockpit Proof & Ultimate Pressure Test RT 08-251

It also may be possible to do some cyclic pressure testing between the main aircraft static tests.

Tail Parachute Limit Load Test

A simple load will be applied at critical angles.

Dive Brake Test RT 08-255

This test need not be done on the static test aircraft.

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PHASE 5 ELEVATED TEMPERATURE TESTING

This phase of testing will continue to be more important as the speed range of the C-105 A/C is extended. A considerable amount of testing will be necessary to clear the aircraft to Mach. 2, but a much larger amount of research testing will be required to clear the A/C for higher Mach. numbers. The effect of creep and induced stresses due to transient temperatures will require close attention.

Transient Heating of Wing Fuel Tank Model RT 08-357

This is the first of this type of testing. The torque box to be tested is similar to the static test inner wing boxes.

The objective is to determine the heat flow through the box and the resulting thermal effects.

Heat will be applied under very close time control and temperatures and stresses measured by the use of continuous trace thermocouples and strain gauges.

PHASE 2 PROOF OF COMPLIANCE (contd.)

Wheel Well Pressurization

Gear will be retracted and doors closed and sealed if necessary. Wheel well will be pressurized to a very low value. This will apply representative loads on the structure over the wheel well.

Main Gear Door Strength & Stiffness RT 08-258

Need not be carried out on the static test aircraft.

Nose Gear Door Strength & Stiffness RT 08-259

Need not be carried out on the static test aircraft.

Ramp Strength & Stiffness Test

This test must be carried out on the static test specimen. Equal and opposite loads will be applied to each ramp. Cockpit will be pressurized.

Main Aircraft Ultimate Static Test

No definite plans have been made for this test. It will not be carried out until some flight test data is available. This obviously places the test at a much later date.

The question also arised whether it would not be better to fail the aircraft by repeated loads rather than by a single ultimate load.

The writer would advise this latter approach. However, since this would be a major deviation from the specifications, we would have to obtain complete written agreement from the R.C.A.F.

PHASE 3 COMPONENT FATIGUE TESTING

The undercarriage is the only item under the existing plan that will be fatigue tested. This is due to the complexity of detail and the use of a super high heat treat steel.

A single gear will be tested. Development tests may be necessary depending on the results of the first tests.

Further component fatigue testing on the aircraft structure will be considered following the static and flight tests.

PHASE 4 FAIL SAFE TESTING

The C-105 A/C uses the fail safe concept as much as possible. The basic concept of fail safe structure is to design a structure in such a way so that any damage will be localised. Redundant structures are generally good fail safe structures. In pressure vessels, small aspect ratio panels will tend to localise cracks preventing disastrous "rips".

This concept is new so that very little experience is in hand. It is planned to use the "remnants" of the static test article to conduct testing of this type and to gain much needed experience along this line, as well as to improve the fail safe characteristics of the C-105 A/C

Intake Duct - Fail Safe Test RT 08-242

The only test conducted to date of a fail safe nature was on a section of intake duct that had been used for pressure and depression tests.

The test consisted of pressurising the intake duct to limit pressure and firing 50 calibre bullets through the duct at strategic points. The pressure in the duct receded in a satisfactory way and no disastrous failure of the duct occurred.

The photograph on Page [illegible] shows the damaged duct.

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Sheet 36

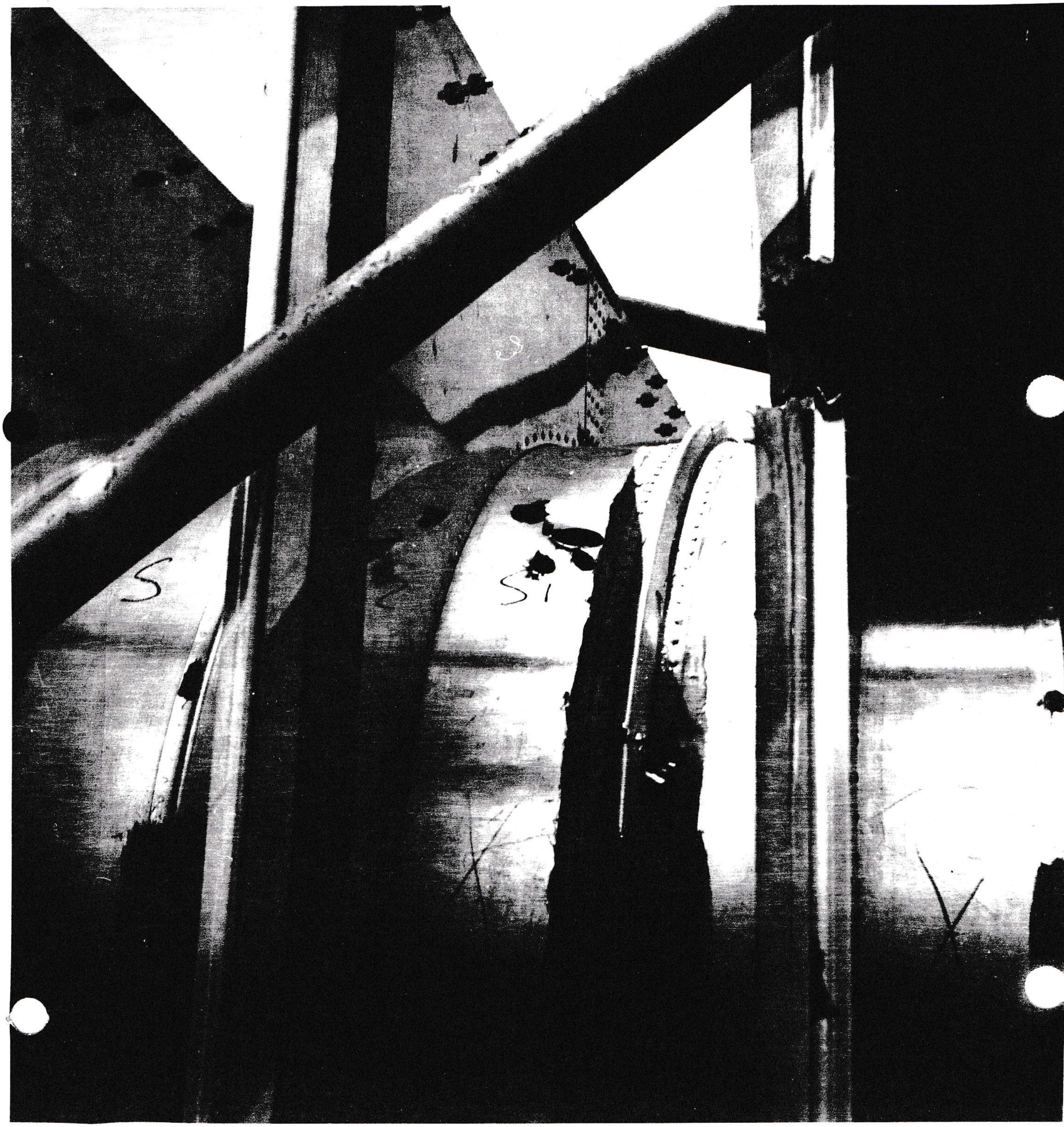

Fig. 15