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Structures Technical Memorandum 341

FLUTTER SUBSTANTIATION TESTS ON A TRANSAVIA FL-12/T-300 AIRTRUK

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Approved for Public Release

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Summary

A resonance test and subsequent flight tests have been conducted on a Transavia Airtruk. The natural modes and frequencies of vibration were measured in the ground tests, and attempts made to induce flutter during flight tests. The results of these tests are presented.
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1. INTRODUCTION

The Transavia Airtruk has been in production for 10 years, more than 100 having been sold, and there is no record of loss due to structural failure in this time. In order to expand overseas markets, particularly in North America, full F.A.A. certification was required and the Aeronautical Research Laboratories were requested to assist with the flutter substantiation tests.

The aircraft has been frequently modified since the first production model and the aircraft tested was as detailed in Table 1, with major dimensions shown in Fig. 1(a). For the ground resonance tests, the aircraft was set up on partially inflated 1 in wheels with the nose suspended, on a soft spring system, from the factory roof structure. With fuel tanks 50 per cent full and the hopper empty, the rigid body modes of pitch, roll, and heave were 1.4 hertz, 4.2 hertz and 4 hertz respectively.

The aircraft was complete and serviceable, with all components present in their correct locations.

The control surfaces were clamped to the main airframe for all the airframe modes except for the mode at 39.4 hertz, and freed for all the control surface measurements except tab rotation.

The undercarriage had the hydraulic dampers replaced with fixed struts, but the dampers were attached to the wheels to retain the correct mass distribution.

The same aircraft was used in the flight tests. All control cables were at the lower service limits of tension. No attempt was made to simulate in-service backlash increases.

2. TEST EQUIPMENT AND PROCEDURES

2.1 Ground Resonance Tests

A maximum of ten electro-magnetic vibrators was used for these tests. These vibrators were nominally identical having a maximum thrust of 138 newtons and diaphragm stiffness of approximately 5 kilonewtons/metre. The vibrators were initially located at the extremities of the wing front and rear spars, the port front spar and starboard rear spar of each tailplane, and in a horizontal direction at the lowest point of each fin.

The vibrators were mounted on adjustable height stands and attached to the aircraft structure through light-weight, telescopic rods using ball-jointed connectors bonded to the skin with a quick-setting adhesive. Relocation of vibrators was found to be necessary
for the wing torsion modes where the rear spar was very close to the
nodal line, and for the control surface modes. For the wing torsion
modes the two vibrators on the rear spars were relocated towards the
wing trailing edges, and for the control surface modes vibrators were
generally located at the trailing edge of the control surfaces.

The vibrators were driven by high output-impedance power-
amplifiers to ensure that the applied forces were always in phase,
irrespective of the motion of the structure. Frequency control was
provided by a variable oscillator.

Accelerometers were located on the structure directly over
each vibrator attachment and at six other locations, on the tailplanes
and fins, to provide an indication of the phase of the response relative
to the oscillator output signal. The sixteen accelerometer signals,
and a signal from the oscillator in quadrature with the driving signal,
were passed through a bank of phase-matched low-pass filters having
a roll-off frequency of 45 hertz. This was to remove distortions in
the signals caused by local panel resonances and other noise. The
signals were then fed into a Resolved Component Ratiometer (RCR) and
a multi-channel oscilloscope. The RCR operates by resolving each
accelerometer signal into its components in phase and quadrature with
a reference signal, summing the similar components, displaying the
average value of in-phase and quadrature components, and displaying
the ratio of the quadrature to the in-phase components of the response.
Using a reference signal in quadrature with the forcing signal makes
this ratio small at resonance. A pure resonant mode would provide a
ratio of zero, whereas, if a reference was used in phase with the
force, the ratio would approach infinity and overload the display
of the ratiometer.

Likewise, the use of the quadrature reference signal provides
Lissajous figures on the oscilloscope which are straight lines at
resonance. Use of a signal in phase with the force would provide
circles at resonance, which would make tuning rather difficult.

As an aid to determination of the resonant frequencies of
the airframe, a random noise signal, of bandwidth 0 to 50 hertz, was
fed into a vibrator at three consecutive locations - wing tip, tail-
plane tip, and fin tip. The response of the accelerometer directly
over the vibrator was analysed, together with the random signal, on
a dual-channel real-time spectrum analyser. In Fig. 2 the transfer
function of response, with reference to random input, is displayed
and the peaks annotated with the modes that were subsequently found
close to the frequencies indicated. The only modes found which did
not show up on these responses were wing torsion, and some of the
boom bending and torsion modes. At each of the resonant frequencies
indicated by the spectrum analyser, the force levels in all the
vibrators were manually adjusted until the acceleration responses at all the monitoring stations were as close as possible to being in quadrature with the exciting force. This was determined by observation of the Lissajous figures, and the RCR display, to obtain a condition with all figures as close as possible to straight lines, and the ratio displayed being a minimum. With a complex structure it is generally not possible, with a limited number of force inputs, to obtain a pure resonant mode.

Modes were found at all the frequencies indicated on the spectrum analyser. Other modes were found at frequencies not indicated initially. Modal measurements were made with a single movable accelerometer attached to the structure, by a suction cup, successively at each point of the array of measuring stations. The layout of the measurement points on the aircraft is indicated in Fig. 1(b) in which half the aircraft is shown. The distribution of measurement points was symmetrical about the centre line. The actual locations are listed and dimensioned in Table 2.

The accelerometer signal was fed to a digital transfer function analyser, which displayed the in-phase and quadrature components of the response with reference to the forcing signal. Modal damping measurements were obtained from records of the decay rates of the vibration when the exciting signal from the oscillator was disconnected.

On completion of the airframe vibration mode measurements, the vibrators were disconnected from the structure, and the control surfaces freed by removing all clamps. The airframe was then rigidly supported at the wings, booms and tailplanes using contour boards on stands with the airframe held onto them with bags of sand. This was done to permit measurement of the uncoupled control surface modes. Accelerometers and vibrators were relocated to suit the individual control surfaces, and resonant modes excited as before, using random input and the spectrum analyser to detect resonances. The oscillator was then used to fine-tune each resonant vibration mode. The aileron, elevator and rudder resonances were obtained by attaching one or more vibrators to the trailing edges of the respective control surface. The tab rotational frequencies were obtained by clamping the elevator to the tailplane and attaching the vibrator to the elevator. The only attachment to the tab was an accelerometer on the trailing edge. A random signal was fed into the vibrator and the transfer function of the tab was determined on the spectrum analyser for the three positions of each tab. The tabs are of the spring loaded type.

The elevator control is connected to a downspring which holds the control column in a full forward position when released. To hold the column central required several restraining rubber cords. The test on the elevators was repeated with a single rubber cord on the control column to try and simulate a hand held situation, and also with the downspring removed completely.
2.2 Flight Tests

Lightweight accelerometers, having a mass of 8 grams each, were fixed to the aircraft structure at thirteen locations as listed in Table 3. Wherever possible, the accelerometers were fixed inside the structure. The accelerometers on the elevator and rudder were mounted externally. In all locations the accelerometers were fixed to the structure with a quicksetting epoxy resin adhesive. On the external surfaces they were also faired with tape to improve airflow. Cables were run, under covers and along wing trailing edges, to amplifiers located in the passenger compartment. This compartment also housed the power supplies, DC to AC inverter, and a 14-track instrumentation tape-recorder. A remote control station and microphone were installed in the pilot's compartment.

Two test flights were made, to cover a total of 12 test points. This involved making several dives from approximately 3400 metres to 1200 metres altitude and holding each required airspeed steady while the control stick was sharply struck in both fore and aft, and sideways directions, to induce a sudden disturbance on the surfaces. The test points, and air speeds are listed in Table 4.

The records were replayed into a multi-channel chart recorder to observe the actual time histories at each test point. In order that the traces were not swamped by noise signals outside the range of interest, the signals were passed through a bank of low-pass filters having a roll-off at 45 hertz. Samples of this time history are presented in Fig.15.

To investigate the frequency response of the structure to the general turbulence present at different speeds the recorded data were analysed on a Fourier analyser, two channels at a time. The spectra obtained for the speeds of 145 and 175 knots are presented in Figs. 16 to 18. In order to obtain the impulse response of the structure, the Random Decrement procedure was used. This procedure is based on methods described in Ref. 1. Two techniques were used to produce Figs 19 to 22 which illustrate the random decrement signatures at different parts of the structure. Initially, the data were passed through a low-pass filter set successively at 10, 15, 20, 25 and 30 hertz to examine responses in each of these ranges, the highest frequency present being dominant in each case. The signal was then digitised at a rate of 1000 samples per second and passed through a microprocessor to provide an acceleration decrement signature of 0.5 second duration. Fig. 22 is the only figure presented which used this technique. The results at the lower frequencies were distorted due, it is thought, to the poor cut-off characteristics of the analogue filters used, and the closeness of various vibration frequencies. To improve matters, digital filters were used after the analogue data had been digitised at the rate of 1280 samples per
second using a time history 1.6 seconds long. At each test point and location three such time histories were used to provide the data which were then analysed using a random decrement programme on a DEC PDP-10 computer to produce Figs. 19 to 21.

3. TEST RESULTS

3.1 General

The vibration mode shapes for all the measured flexible modes are listed in Table 5 and plotted in Figs. 3 to 14. In all instances only the response of the structure in quadrature with the exciting force is plotted. For clarity of presentation, not every measured point is plotted, and in cases where the vibration amplitudes on certain components of the aircraft structure were negligible, measurements were not recorded.

3.2 Details of Mode Shapes

Mode at 5.14 Hz. (Fig. 3)

This mode is the antisymmetric lateral bending of the booms. Because of the high tailplane some torsion is also introduced. There is also slight antisymmetric bending of the wings.

Mode at 6.07 Hz. (Fig. 4)

This is antisymmetric vertical bending of the booms. The motion of the wings is essentially rigid body motion.

Mode at 9.7 Hz. (not illustrated)

This is the rotation frequency of the elevators. The control spring had been removed and a rubber cord was used to hold the control column central. It was considered that this best simulated actual flying conditions. The response of each elevator was independent of the other and could be made to oscillate in-phase or in opposition.

Mode at 9.9 Hz. (Fig. 5)

This is symmetric torsion of the booms. The two booms are slightly different, and with the close proximity of the antisymmetric mode at 10.5 Hz some difficulty was experienced initially in separating the two modes.

Mode at 10.5 Hz. (Fig. 6)

This is the antisymmetric torsion mode of the booms. This mode was excited mainly by forces on the starboard tail with a slight restrainin
force on the front tail whereas the mode at 9.9 Hz. was excited solely from forces on the port tail. The main wing is indicating symmetric bending in spite of the antisymmetric nature of the reaction torque applied at the boom connection. This is no doubt due to the close proximity of the fundamental wing bending mode at 11.4 Hz.

**Mode at 11.4 Hz. (Fig. 7)**

This is fundamental wing bending coupled with symmetric boom torsion. The boom torsion is the main cause of the sharp change in bending at the intersection of boom and wing.

**Mode at 14.2 Hz. (not illustrated)**

This is elevator rotation with the control column fixed. The column, with its control spring, was lashed in two directions using rubber cords. The damping measured in this mode is quite high; in excess of 5 percent of critical damping. The elevator rotation could be in-phase or anti-phase depending upon direction of force input. This mode appears very similar to the mode at 9.7 hertz.

**Mode at 14.4 Hz. (Fig. 8)**

This is the wing antisymmetric bending mode. Antisymmetric bending of the booms occurs with rotation of tailplanes.

**Mode at 21.6 Hz. (not illustrated)**

This is anti-phase rotation of the two rudders with the control column free. This mode was excited with a force on the port rudder only.

**Mode at 23.4 Hz. (not illustrated)**

This is symmetric rotation of the ailerons with the stick free.

**Mode at 30.8 Hz. (not illustrated)**

This is the symmetric bending of the stub wing, and was excited via force inputs to the main wings.

**Mode at 35.6 Hz. (Fig. 9)**

This is the symmetric wing torsion mode coupled with spanwise aileron bending.
Mode at 38 Hz. (Fig. 10)
This mode is essentially rudder bending about its hinge line, coupled with antisymmetric tailplane torsion. The rudder was clamped to the fin at the hinge line with a single clamp. The tailplane torsion is present because of the close proximity of the mode at 38.4 Hz. measured with the control surfaces free.

Mode at 38.4 Hz. (Fig. 11)
This is antisymmetric torsion of the elevator.

Mode at 39.4 Hz. (Fig. 12)
This mode is antisymmetric wing torsion and antisymmetric aileron torsion coupled with aileron bending. This mode was measured with the ailerons free to rotate and with the airframe on its soft support.

Mode at 43.2 Hz. (Fig. 13)
This is rudder bending coupled with some boom torsion which causes the tailplane to roll.

Mode at 46.5 Hz. (Fig. 14)
This mode is symmetric tailplane bending.

Mode at 48 Hz. (not illustrated)
This is the rotation frequency of the tailplane tab when in the maximum down position.

Mode at 53.6 Hz. (not illustrated)
This is the rotation frequency of the tailplane tab when in the mean position.

Mode at 55.2 Hz. (not illustrated)
This is the rotation frequency of the tailplane tab when in the maximum up position.

3.3 Flight Tests

On the samples of time history produced in Fig. 15 the impulses applied to the ailerons are well defined, as are the responses on the wings and boom to the impulses. The impulses applied to the elevators are not obvious no doubt due to the damping effect of the long control
cables, which pass over several pulleys and two 90 degree changes in direction before reaching the elevator. The general level of random acceleration at the elevator, as measured by the accelerometers and shown in Figs. 15(b) and 15(d), is almost equal in amplitude to the impulses applied to the ailerons, and should have been sufficient to initiate flutter if the aircraft had crossed a flutter boundary. It should be noted that the vertical scale used in Figs. 15(c) and 15(d) is 10 times that used in Figs. 15(a) and 15(b). The accelerometer amplifier sensitivity was also changed following the first flight, as saturation of the amplifiers was suspected. The top record on Fig. 15(d) indicates that saturation was a problem with that particular amplifier even at the lower sensitivity. Figs. 16 to 18 show the frequency spectra of the accelerations at six locations and two air speeds. The most significant feature appears on Figs 16(a) and 16(b) where a sharp peak can be seen at approximately 21 hertz on both wing and aileron.

Detailed investigation of the records at other speeds and altitudes indicates that the vibration frequency and amplitude does not vary between speeds from 100 to 175 knots, or altitudes from 1230 to 2770 metres, and generally does not exceed 1 millimetre peak-to-peak at the aileron trailing edge. The oscillation disappeared at various times during the flights, particularly at times when the pilot may have been undertaking manoeuvres using the ailerons.

In Figs. 19 to 21 (produced by Random Decrement Analysis), which indicate the impulse response of the structure at six locations at a speed of 175 knots, it can be seen that vibrations around the tail region are damped. In Fig. 20(a) the wing and aileron show a relatively low damped oscillation at 10 hertz. This is due to leakage of the response of the wing bending mode of about 12 hertz, (see Fig. 16(a)), through the filter at 10 hertz. Fig. 20(b) shows this mode correctly at about 12 hertz with higher damping.

Fig. 20(d) shows the aileron having signs of a lightly damped oscillation in the 20 to 25 hertz range. To illustrate the aileron vibration further, Fig. 22 is presented to show the rapid decay from the initial peak response to the continuing limited amplitude oscillation. The two curves indicate that the frequency and damping of the aileron oscillation is independent of air speed in the range covered by the test.

4. DISCUSSION

The approach to determining the resonant frequencies of the structure by use of a spectrum analyser appears to have some shortcomings. The method adopted in this test, of using a random input and a transfer function of the response with reference to the input,
generally showed the regions in which resonances occurred, but was unable to separate close resonances, such as the symmetric and antisymmetric boom torsion modes, even when using the band expansion capabilities of the analyser. However, the frequency spectra shown in Fig. 2 indicate all the regions in which there were airframe vibration modes except the wing torsion modes. This omission was caused by placing the accelerometer and shaker close to the torsional nodal line. This would have been avoided if the wing-tip leading-edge had been used.

With the flight tests, the fact that the pilots cabin and the instrumentation compartment were well separated introduced the problem of monitoring the operation of equipment in flight. It is apparent that a telemetry system, with remote control of amplifier gains, would be an advantage in this type of work. This would also obviate the need to carry the tape recorder on the flight.

5. CONCLUSIONS

The aircraft has been ground tested, in relation to its airframe vibration characteristics, and all the expected modes of vibration measured. The flight tests have been carried out and every reasonable effort made to induce flutter at speeds up to 175 knots. The aircraft, in the configuration tested, has been found to be clear of flutter at air speeds up to and including 175 knots.
REFERENCES

## TABLE 1(a)
### DETAILS OF AIRCRAFT FOR GROUND RESONANCE TEST

<table>
<thead>
<tr>
<th>Aircraft type</th>
<th>Transavia PL-12/T-700 Airtruk</th>
</tr>
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<tbody>
<tr>
<td>Serial No.:</td>
<td>G996</td>
</tr>
<tr>
<td>Registration:</td>
<td>VH-UOT</td>
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<tr>
<td>Engine:</td>
<td>Lycoming Type IO-540-K1AS</td>
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<tr>
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<td>Serial No: L-18871-48A</td>
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<tr>
<td>Propeller:</td>
<td>Hartzell Type HC-C3YR-1RF/P8466A-2R</td>
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<td>Serial No: DY903</td>
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<tr>
<td>Hopper configuration:</td>
<td>Wet/Dry Doors</td>
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<tr>
<td>Flap Control:</td>
<td>Standard manual</td>
</tr>
<tr>
<td>Modifications incorporated:</td>
<td></td>
</tr>
<tr>
<td>No. 140</td>
<td>Horn balance elevators</td>
</tr>
<tr>
<td>No. 141</td>
<td>Bevel Trailing Edge Ailerons</td>
</tr>
<tr>
<td>No. 143</td>
<td>Elevator Damp Springs</td>
</tr>
<tr>
<td>Total mass approx.</td>
<td>1000 kilograms</td>
</tr>
<tr>
<td>Centre of Gravity approx.</td>
<td>420 millimetres aft of main wing leading edge</td>
</tr>
</tbody>
</table>

## TABLE 1(b)
### AIRCRAFT CONFIGURATION FOR FLIGHT TESTS

| Mass of aircraft plus oil, fuel, pilot, parachute, battery pack, tape recorder, amplifiers, etc. | 1408 kilograms |
| Maximum permitted mass of aircraft                                                               | 1742 kilograms |
| Centre of Gravity of aircraft at take-off.                                                       | 0.532 metres aft of datum |
| Limits of Centre of Gravity                                                                      | 0.417 to 0.589 metres aft of datum |
| Moment about Centre of Gravity                                                                   | 7350 Newton-metres |
| Control surface balances:                                                                        |                  |
| Trim tab                                                                                          | 0.3 kilograms T.E. heavy |
| Elevator                                                                                            | 0.14 kilograms L.E. heavy |
| Aileron                                                                                            | 1.59 kilograms T.E. heavy |
| Rudder                                                                                             | 0.75 kilograms T.E. heavy |
### TABLE 2(a)

**LOCATION OF WING MEASURING STATIONS**

<table>
<thead>
<tr>
<th>$\eta$</th>
<th>$\xi$</th>
<th>0.015</th>
<th>0.202</th>
<th>0.635</th>
<th>0.765</th>
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<td>+0.995</td>
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<td>W1B</td>
<td>W1C</td>
<td>W1D</td>
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<td>-0.995</td>
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<td>W12C</td>
<td>W12D</td>
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<td>+0.844</td>
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<td>W2B</td>
<td>W2C</td>
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<td>W10E</td>
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<td>W4B</td>
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<td>W4E</td>
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<tr>
<td>-0.470</td>
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<td>W9A</td>
<td>W9B</td>
<td>W9C</td>
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<td>W9E</td>
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<td>W6A</td>
<td>W6B</td>
<td>W6C</td>
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<td>-0.131</td>
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<td>W7A</td>
<td>W7B</td>
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</table>

$\xi$ is proportion of wing chord aft of wing leading edge

$\eta$ is proportion of wing semi-span from aircraft centreline
### TABLE 2(b)

LOCATION OF TAILPLANE MEASURING STATIONS

<table>
<thead>
<tr>
<th>η</th>
<th>0.042</th>
<th>0.375</th>
<th>0.625</th>
<th>0.986</th>
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<tbody>
<tr>
<td>+ 0.875</td>
<td>PT1A, ST1A</td>
<td>PT1B, ST1B</td>
<td>PT1C, ST1C</td>
<td>PT1D, ST1D</td>
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<td>- 0.875</td>
<td>PT4A, ST4A</td>
<td>PT4B, ST4B</td>
<td>PT4C, ST4C</td>
<td>PT4D, ST4D</td>
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<td>+ 0.484</td>
<td>PT2A, ST2A</td>
<td>PT2B, ST2B</td>
<td>PT2C, ST2C</td>
<td>PT2D, ST2D</td>
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<tr>
<td>- 0.484</td>
<td>PT3A, ST3A</td>
<td>PT3B, ST3B</td>
<td>PT3C, ST3C</td>
<td>PT3D, ST3D</td>
</tr>
</tbody>
</table>

ξ is proportion of tailplane chord aft of tailplane leading edge.
η is proportion of tailplane semi-span from boom centreline.

### TABLE 2(c)

LOCATION OF FE', RUDDER AND BOOM MEASURING STATIONS

<table>
<thead>
<tr>
<th>B.S.</th>
<th>W.L.</th>
<th>2540</th>
<th>3810</th>
<th>4220</th>
<th>4440</th>
<th>4950</th>
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<tr>
<td>- 305</td>
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<tr>
<td>- 75</td>
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<td>0</td>
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<td>+ 355</td>
<td></td>
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<td>+510</td>
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</table>

PF4A, SF4A
PF4B, SF4B
PF4C, SF4C
PF3B, SF3B
PF3C, SF3C
PF2B, SF2B
PF2C, SF2C
PF1B, SF1B
PF1C, SF1C

B.S. - Body Station in millimetres from main wing root leading edge.
W.L. - Water line in millimetres from boom centreline.
TABLE 3

LOCATION OF ACCELEROMETERS FOR FLIGHT TESTS

<table>
<thead>
<tr>
<th>Location Description</th>
<th>Measuring Station(s)</th>
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<tbody>
<tr>
<td>Port aileron - measuring station W1E</td>
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<tr>
<td>Port-wing front spar - measuring station W11B</td>
<td></td>
</tr>
<tr>
<td>Port-wing rear spar - measuring station W11C</td>
<td></td>
</tr>
<tr>
<td>Starboard-wing front spar - measuring station W2B</td>
<td></td>
</tr>
<tr>
<td>Starboard-wing rear spar - measuring station W2C</td>
<td></td>
</tr>
<tr>
<td>Port fin - measuring station PF2B</td>
<td></td>
</tr>
<tr>
<td>Port fin - measuring station PF4B</td>
<td></td>
</tr>
<tr>
<td>Port boom - 0.5 metres aft of PB2V</td>
<td></td>
</tr>
<tr>
<td>Port tailplane - inboard of PT1B</td>
<td></td>
</tr>
<tr>
<td>Port elevator - measuring station PT4D</td>
<td></td>
</tr>
<tr>
<td>Port elevator - measuring station PT1D</td>
<td></td>
</tr>
<tr>
<td>Port rudder - measuring station PF2C</td>
<td></td>
</tr>
<tr>
<td>Port rudder - measuring station PF4C</td>
<td></td>
</tr>
</tbody>
</table>

(Refer Fig. 1(b) for locations of measuring stations)

TABLE 4

FLIGHT TEST POINTS

<table>
<thead>
<tr>
<th>Test point</th>
<th>I.A.S. Knots</th>
<th>Altitude metres (feet)</th>
<th>Date flown</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>106</td>
<td>2770 (9000)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>2</td>
<td>117</td>
<td>2000 (6500)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>3</td>
<td>128</td>
<td>1230 (4000)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>4</td>
<td>128</td>
<td>2770 (9000)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>5</td>
<td>138</td>
<td>2000 (6500)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>6</td>
<td>148</td>
<td>1230 (4000)</td>
<td>10th July 1981</td>
</tr>
<tr>
<td>7</td>
<td>145</td>
<td>2770 (9000)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>8</td>
<td>152</td>
<td>2000 (6500)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>9</td>
<td>160</td>
<td>1230 (4000)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>10</td>
<td>160</td>
<td>2770 (9000)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>11</td>
<td>166</td>
<td>2000 (6500)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>12</td>
<td>175</td>
<td>1230 (4000)</td>
<td>12th July 1981</td>
</tr>
<tr>
<td>Description of Mode</td>
<td>Natural Frequency Hz</td>
<td>Damping % Critical</td>
<td>Ratiometer Rdg. Q/R</td>
</tr>
<tr>
<td>---------------------------------------------------------</td>
<td>---------------------</td>
<td>--------------------</td>
<td>---------------------</td>
</tr>
<tr>
<td>Antisymmetric lateral bending of booms</td>
<td>5.14</td>
<td>2.2</td>
<td>0.26</td>
</tr>
<tr>
<td>Antisymmetric vertical bending of booms</td>
<td>6.07</td>
<td>2</td>
<td>0.05</td>
</tr>
<tr>
<td>Elevator rotation - stick free and downspring removed</td>
<td>9.7</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Symmetric torsion of booms</td>
<td>9.9</td>
<td>-</td>
<td>0.07</td>
</tr>
<tr>
<td>Antisymmetric torsion of booms</td>
<td>10.5</td>
<td>1.7</td>
<td>0.08</td>
</tr>
<tr>
<td>Symmetric wing bending with symmetric boom torsion</td>
<td>11.4</td>
<td>1</td>
<td>0.08</td>
</tr>
<tr>
<td>Elevator rotation - stick fixed</td>
<td>14.2</td>
<td>&gt;5</td>
<td>-</td>
</tr>
<tr>
<td>Antisymmetric wing bending with antisymmetric boom torsion</td>
<td>14.4</td>
<td>2.9</td>
<td>0.05</td>
</tr>
<tr>
<td>Rudder anti-phase rotation - stick free</td>
<td>21.6</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Aileron symmetric rotation</td>
<td>23.4</td>
<td>2.4</td>
<td>-</td>
</tr>
<tr>
<td>Stub wing symmetric bending</td>
<td>30.8</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Symmetric wing torsion with aileron bending</td>
<td>35.6</td>
<td>2.5</td>
<td>0.2</td>
</tr>
<tr>
<td>Rudder bending with tailplane antisymmetric torsion</td>
<td>38</td>
<td>2.2</td>
<td>0.15</td>
</tr>
<tr>
<td>Elevator antisymmetric torsion</td>
<td>38.4</td>
<td>1.4</td>
<td>-</td>
</tr>
<tr>
<td>Antisymmetric wing torsion and antisymmetric aileron torsion with aileron bending</td>
<td>39.4</td>
<td>2.4</td>
<td>-</td>
</tr>
<tr>
<td>Rudder bending</td>
<td>43.2</td>
<td>1.5</td>
<td>0.11</td>
</tr>
<tr>
<td>Symmetric tailplane bending</td>
<td>46.5</td>
<td>1.8</td>
<td>0.1</td>
</tr>
<tr>
<td>Tab rotation - max. down position</td>
<td>46</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Tab rotation - mean position</td>
<td>53.6</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Tab rotation - max. up position</td>
<td>55.2</td>
<td>-</td>
<td>-</td>
</tr>
</tbody>
</table>
FIG. 1(a) GENERAL OUTLINE OF AIRCRAFT WITH MAJOR DIMENSIONS
FIG. 1(b) LOCATIONS OF MEASURING STATIONS (only half shown here)
RESPONSE AT WING TIP TO RANDOM INPUT

FIG. 2(a) RESPONSE TO RANDOM INPUT
RESPONSE AT TAILPLANE TO RANDOM INPUT

![Response Spectrum Graph]

**Fig. 2(a) Response to Random Input**
**FIG. 3 (a) MCDE AT 5.14 Hz**
FIG. 3(b) MODE AT 5.14 Hz
FIG. 4(a) Mode at 6.07 Hz
FIG. 4(b) MODE AT 6.07 Hz
FIG. 5(a) MODE AT 9.9 Hz
FIG. 6(a) MODE AT 10.5 Hz
FIG. 6(b) MODE AT 10.5 Hz
FIG. 7(a) MODE AT 11.4 Hz
FIG. 5(b) MODE AT 14.4 Hz
FIG. 9 (b) MODE AT 35.5 Hz
FIG. 10(a) WOLF AT 38 Hz
Fig. 11(b) Mode at 3.4 Hz
FIG. 13(a) MODE AT 43.2 Hz
STBD BOOM VERTICAL

PORT BOOM VERTICAL

FIN/STBD BOOM LATERAL

PORT BOOM LATERAL

W1

W3

W5

W8

W10

W12

FIG. 13(a) MODE AT 43.2 Hz
FIN/RUD LATERAL

STBD TAILPLANE VERT

WING VERTICAL

STBD WING VERTICAL

PORT TAILPLANE VERT

A LEADING EDGE
○ TRAILING EDGE
△ HINGE LINE ON ELEVATOR
□ BALANCE HORN

FIG. 14(a) MODE AT 46.5 Hz
STBD BOOM VERTICAL
PORT BOOM VERTICAL
STBD BOOM LATERAL
PORT BOOM LATERAL

W1
W3
W5
W8
W10
W12

ST2
ST3
ST4

PFT 011/020 AT 364 HK
FIG. 15(a) TEST POINT 2 - 117 KNOTS I.A.S.
Port boom
vertical
0.2g/cm

Port tailplane
0.1g/cm

Port elevator
0.1g/cm

Port aileron
0.1g/cm

Port wing
Front spar
0.1g/cm

FIG. 15(b) TEST POINT 2 - 117 KNOTS I.A.S.
FIG. 15(c) TEST POINT 12 - 175 KNOTS I.A.S.
FIG. 15(d) TEST POINT 12 - 175 KNOTS I.A.S.
AIRTRUK 145 KTS. AILERON/WING

0.100 g RMS

FIG. 16(a) FREQUENCY RESPONSE AT 145 KNOTS I.A.S.
AIRTRUK 175KTS. AILERON/WING

0.100 g RMS

ACCELERATION

FREQUENCY

50.8 Hz

WING

FIG. 16(b) FREQUENCY RESPONSE AT 175 KNOTS I.A.S.
AIRTRUK : 45KTS. ELEVATOR/TAILPLANE

0.100 g RMS

FREQUENCY

TAILPLANE

50.0 Hz

FIG. 17(a) FREQUENCY RESPONSE AT 145 KNOTS T.A.S.
AIRTRUK 145KTS. Rudder/FIN

0.100 g RMS

ACCELERATION

FREQUENCY 60.0 Hz

Fig. 13(a) Frequency response at 145 knots I.A.S.
AIRTRUK 175KTS. RUDDER/FIN

0.100 g RMS

FIG. 18(b) FREQUENCY RESPONSE AT 175 KNOTS I.A.S.
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 10 Hz

**ELEVATOR**
- No of Averages = 112
- No of Pts in Sample = 512

**TAIL**
- No of Averages = 116
- No of Pts in Sample = 512

FIG. 19(a) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 15 Hz

ELEVATOR


TAIL

FIG. 19(b) RANDOM DECREMENT SIGNATURE

NO OF AVERAGES = 164
NO OF PTS IN SAMPLE = 512

NO OF AVERAGES = 147
NO OF PTS IN SAMPLE = 512
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 20 Hz

ELEVATOR

NO OF AVERAGES = 178
NO OF PTS IN SAMPLE = 512

TAIL

NO OF AVERAGES = 199
NO OF PTS IN SAMPLE = 512

FIG. 10(c) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 25 Hz

ELEVATOR

NORMALISED AMPLITUDE

TIME SECS

NO OF AVERAGES = 204
NO OF PTS IN SAMPLE = 512

TAIL

NORMALISED AMPLITUDE

TIME SECS

NO OF AVERAGES = 262
NO OF PTS IN SAMPLE = 512

FIG. 19(d) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 15 Hz

AILERON

NO OF AVERAGES = 117
NO OF PTS IN SAMPLE = 512

WING

NO OF AVERAGES = 139
NO OF PTS IN SAMPLE = 512

FIG.20(b) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 20 Hz

FIG. 20(a) RANDOM DICREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 25 Hz

[Diagram of normalized amplitude with labeled data points]

NO OF AVERAGES = 189
NO OF PTS IN SAMPLE = 512

[Diagram of normalized amplitude with labeled data points]

NO OF AVERAGES = 193
NO OF PTS IN SAMPLE = 512

FIG. 26(a) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 10 Hz

---

**Rudder**

- Normalised Amplitude
- Time: 0.40 time secs
- No of Averages: 88
- No of pts in sample: 512

---

**Fin**

- Normalised Amplitude
- Time: 0.40 time secs
- No of Averages: 106
- No of pts in sample: 512

---

FIG.21(a) RANDOM DECREMENT SIGNATURE
AIRTRUK AT 75 Knots.... LOW-PASS FILTER AT 15 Hz

**Rudder**

- No of Averages: 127
- No of Points in Sample: 512

**Fin**

- No of Averages: 145
- No of Points in Sample: 512

**Figure 21(b) Random Decrement Signature**
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 20 Hz

**RUDDER**

- Time: 1.40
- Number of Averages: 188
- Number of Points in Sample: 512

**FIN**

- Time: 1.40
- Number of Averages: 171
- Number of Points in Sample: 512

**FIG. 21(c) RANDOM DECREMENT SIGNATURE**
AIRTRUK AT 175 Knots .... LOW-PASS FILTER AT 25 Hz

Rudder

NORMALISED AMPLITUDE

TIME SECS

NO OF AVERAGES = 182
NO OF PTS IN SAMPLE = 512

Fin

NORMALISED AMPLITUDE

TIME SECS

NO OF AVERAGES = 194
NO OF PTS IN SAMPLE = 512

FIG. 11(4) RANDOM DECREMENT SIGNATURE
AIRTRUK 175/117KTS. AILERON

175 KNOTS

117 KNOTS

TIME

0.5 SEC

FIG. 22 RANDOM DECREASED SIG. NATURES (FILTERED AT 30 HERTZ, LOW PASS)
AUSTRALIA

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Central Office

Chief Defence Scientist
Deputy Chief Defence Scientist
Superintendent, Science and Technology Programmes
Controller, Programmes and Analytical Studies
Defence Science Representative (U.K.) (Doc Data sheet only)
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A resonance test and subsequent flight tests have been conducted on a Transavia Airtruk. The natural modes and frequencies of vibration were measured in the ground tests, and attempts made to induce flutter during flight tests. The results of these tests are presented.
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0–8