

ST. NO.
U.D.C. 32664
AUTH.

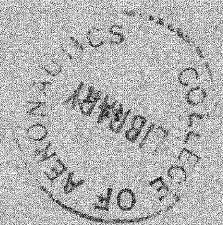


THE COLLEGE OF AERONAUTICS
CRANFIELD

RECENT AEROELASTIC INVESTIGATIONS AT THE
COLLEGE OF AERONAUTICS

by

D. J. Johns



A
32664A



CoA Aero Note No. 168

January, 1965

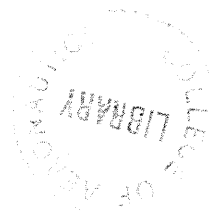
THE COLLEGE OF AERONAUTICS

CRANFIELD

Recent aeroelastic investigations
at The College of Aeronautics*

- by -

D.J. Johns



* Based on a Lecture given to the Coventry Branch of the Royal Aeronautical Society on February 19th, 1964.

Contents

	<u>Page No.</u>
List of symbols	1
1. Introduction	2
2. Panel, Membrane and Wing Instabilities	2
2.1 Supersonic Flutter of Rectangular Plane Isotropic Panels	2
2.2 Supersonic Flutter of Elliptic, Plane Isotropic Panels	3
2.3 Supersonic Flutter of Circular Cylindrical Shells	4
2.4 Subsonic Divergence of Rectangular Plane Panels	5
2.5 Subsonic Flutter of Rectangular Plane Membrane	6
2.6 Supersonic Divergence of Rectangular Flat Plate Aerofoils	6
2.7 Static Aeroelastic Behaviour of Doubly Swept Wings (M - Wings)	7
3. Investigations Associated with the Morane Saulnier M.S. 760 'Paris'	7
3.1 Elevator Flutter of a T-Tail Aircraft	7
3.2 Static Aeroelastic Problems	9
3.3 Wing-Aileron Flutter	11
3.4 Flutter of Aircraft Aerials	11
4. Investigations on Design Project Studies	12
4.1 Supersonic Delta Wing Airliner (1960)	12
4.2 V.T.O.L. Freighter (1961)	13
4.3 Supersonic Canard Airliner	13
4.4 General Purpose Freighter (1963)	15
5. Helicopter Problems	15
5.1 Analyses Including Aerodynamic Forces	15
5.2 Helicopter Ground Resonance	16
6. References	16
Figures	

List of symbols

a	speed of sound
b	maximum spanwise dimension of unswept elliptic panel
D	flexural rigidity ($Eh^3/12(1-\nu^2)$)
E	Youngs modulus
f	tension parameter ($N_{xx}/\sigma L^2\omega_1^2\pi^2$)
g	structural damping coefficient
h	panel thickness
K	flutter parameter ($\rho V^2 L^3 / MD$)
L	panel chord - maximum chordwise dimension of unswept elliptic panel
M	Mach number
N_{xx}	chordwise stress resultant
q	pitch rate
R	aerodynamic damping parameter ($\rho a / \sigma \omega_1$)
V	airspeed (suffices e and i signify 'equivalent' and 'indicated')
W	lateral deflection
x, y	chordwise and spanwise co-ordinates
α	body incidence
β	foreplane flap incidence
γ	% critical damping
ξ	aileron angle
η	elevator angle
ρ	air density
σ	mass per unit area
ω_1	panel fundamental frequency in chordwise bending
ω	panel flutter frequency
ν	Poisson's ratio

1. Introduction

Since the College of Aeronautics was founded in 1946 aeroelasticity has been the subject of much research by both students and staff in the Departments of Aircraft Design and Aerodynamics.

It is not the present intention to review the entire period since 1946, but, rather, to concentrate on the period 1958/63 during which the author has been associated with the College. In this period aeroelastic research has been concentrated on four main themes, viz:

- (a) panel, membrane and wing instabilities;
- (b) investigations associated with the Morane-Saulnier M.S.760 'Paris';
- (c) investigations on Design Project Studies, and
- (d) helicopter problems.

The list of references covers the entire period up to 1963 and contains references dealing with such related topics as flutter model construction and unsteady aerodynamics. These are included for the sake of completeness since such investigations often bear directly on the more specific problems of aeroelasticity. It should also be mentioned that there has been considerable research on vibration testing, impact loading, etc., but such references are not included.

No detailed comparisons have been attempted between the research described and similar work done elsewhere - for these the appropriate references must be studied. It must be emphasised however that the research performed whilst being certainly worthwhile for its own sake also enables the students to become familiar with research methods - a primary aim of the College.

2. Panel, Membrane and Wing Instabilities

2.1 Supersonic Flutter of Rectangular Plane Isotropic Panels

Most of the plane, panel flutter investigations to date have considered rectangular panels, and the effects of various edge conditions, external flow conditions, etc. have all been examined. The published analyses differ considerably in the aerodynamic theories used, which include linearised supersonic flow theory, a quasi-steady approximation to this theory, and static theory in which all time-dependent effects are neglected.

The justification for this last approach was examined by the author (Ref. 1¹) in an analysis using linear piston theory for two-dimensional panels of finite chord. A two- and four- mode analysis was made employing the

¹The major part of Ref. 1 was written whilst the author was employed at Sir W.G. Armstrong Whitworth Aircraft Ltd., Coventry during 1957.

Galerkin procedure and the results showed that at Mach numbers above 2 aerodynamic damping can be neglected when structural damping is zero.

In most analyses to date the influence of structural damping has been neglected, presumably on the assumption that the result would be conservative. It has however been shown (Ref. 2¹) that at speeds where piston theory is applicable, hysteretic structural damping may be destabilising i.e. when the stiffness terms are factored by the quantity $(1 + ig)$ where g is the structural damping coefficient. Some of the analytical results are summarised in Fig. 1, where the critical flutter parameter K^2 is plotted against an aerodynamic damping parameter, R , for various values of g and membrane tension in the panel, $\pi^2 f$. It is seen that the significance of R , when $g = 0$ is very small, as mentioned above, but for $g \neq 0$ the variation of K^2 with R is much more pronounced. The destabilising effect of structural damping and membrane compression is clearly seen. Note: $\pi^2 f = -1$ corresponds to Euler buckling.

The possibility of wing/panel flutter coupling was examined in Ref. 3 in a brief binary analysis of a simplified wing/panel configuration. The modes assumed were wing twist and fundamental bending of the chordwise panel array simply supported on spanwise stiffeners. Linear piston theory was used for the aerodynamic forces which were assumed to act only on the flexible panels. For most practical wing structures the likelihood of wing/panel flutter coupling was shown to be small but for configurations with high wing torsional frequency and low panel bending frequency (i.e. frequency ratio $\rightarrow 1$) such a coupling should be examined.

2.2 Supersonic Flutter of Elliptic, Plane Isotropic Panels

Since most of the published literature has related to rectangular plane panels an analysis was made (Ref. 1) dealing with panels of elliptic planform. The analysis assumed linear piston theory, zero membrane stresses and zero structural damping and a two-mode Galerkin procedure was adopted using 'static' deflection modes for a clamped edge panel.

Now it is well known that it is customary in flutter analyses to use the in-vacuo vibration modes of the structure. For the configuration considered general analytical expressions for the modal shapes for varying ellipticity were not available and the 'static' modes used corresponded to the deflection shapes of the panel when subjected to both a uniform, and a linearly varying, pressure across the panel chord. For the special case of a circular panel it has since been shown by comparison with analyses using the known vibration modes, that the use of simple 'static' deflection modes is justified. This significant result can probably be generalised for other panel configurations.

¹Ref. 2 was written with a former colleague at Bristol Aircraft Ltd.

Further analysis have since been made by students at the College which are reported in Ref. T.11 and T.14. These studies and a later analysis of the effect of sweepback on the flutter of clamped edged elliptic panels are reported on in more detail in Ref. 4.

Thus, it is shown for the clamped edged panel that structural damping and compressive membrane stresses are destabilising and the effect of sweepback is stabilising or destabilising depending on whether the panel 'aspect ratio' (b/L) is less than, or greater than, unity initially. In general the addition of a concentrated mass is stabilising but may be destabilising when the additional mass is added close to the antinode of the overtone mode. However, the addition of a concentrated mass as a means of stabilisation is not as effective as increasing the panel thickness.

Results for circular panels with varying degrees of edge restraint are also given. There is a 20 per cent difference in critical panel thickness between simply-supported and fully clamped edges.

An experimental investigation of the flutter of circular panels is reported in Ref. T.11 and T.14. The findings have been summarised in Ref. 5. The tests were made in the continuous running $9'' \times 9''$ supersonic tunnel at the College, at a fixed Mach number of 2, on aluminium panels having an unsupported area of 6 in. diameter. The panel mounting apparatus is shown in Fig. 2. Inductance type pick-ups were used to measure the panel vibrations but since vibrations were experienced on all the panels tested and at all stagnation pressures it was uncertain whether panel flutter has in fact occurred. However, by a comparison of the predicted and measured 'flutter' frequencies it is believed that the thinnest panel (0.011 in.) did flutter at a high stagnation pressure (about 12 p.s.i.).

These tests were only exploratory in nature but they enabled some of the problems of instrumentation and interpretation to be tackled.

2.3 Supersonic Flutter of Circular Cylindrical Shells

For ring stiffened shells with small ring spacing the use of a travelling wave approach to flutter analysis can only be justified if the flutter wavelength (axial) is small in comparison with the spacing - and it has therefore been customary to use a standing wave analysis. Such an approach was followed in Ref. 6. Three simultaneous differential equations were derived for the equilibrium of a shell element in terms of the axial, circumferential and radial displacement components, and by using linear piston theory for the radial aerodynamic forces the flutter problem was examined.

By neglecting tangential inertia forces in the shell and considering a binary flutter analysis a particularly simple closed form result is obtained which indicates that, if aerodynamic damping is also neglected, the axisymmetric flutter mode is the most critical (Ref. 7). It is now

felt that tangential inertia forces should be retained in the analysis, since, particularly for modes of large axial or circumferential wavelength, the effect on the predominant resonant frequencies may be large. In fact many authors have tended to assume also that aerodynamic damping could be neglected - as it can be for plane panels at high supersonic speeds. For cylindrical shells however the flutter frequencies are much higher and aerodynamic damping must be included since it depends directly on the flutter frequency (Ref. 8). It follows (Ref. 5) that the significant coupled modes in a binary analysis using piston theory are those having a low sum of the squares of the modal natural frequencies as well as a low difference in these squares. For practical shells this corresponds to modes with a large number of circumferential waves.

It was shown in Ref. 6 by an axisymmetric flutter mode analysis that the use of a travelling wave form of the radial displacement component gives similar results as a standing wave form for shells of finite length. It has since been shown that the converse is also true, viz. a standing wave binary analysis, using an asymmetrical expression for the radial displacement component, reduces, if one assumes a large number of circumferential waves, to the same form as obtained from a travelling wave analysis.

2.4 Subsonic Divergence of Rectangular Plane Panels.

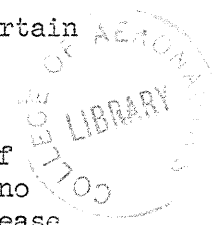
The static aeroelastic instability of two dimensional panels in a subsonic flow has been investigated by several authors and it was suggested that the phenomenon of 'panel divergence' might have occurred in practice.

In several experimental investigations conducted at the College (Ref. T.11, T.15, T.19) 'panel divergence' was never experienced but certain observations can be made on the results obtained.

Thus, in Ref. T.15 a panel was tested having nominally clamped end conditions and subject to a constant tensile load. For given values of this load it was found that, whilst at low dynamic pressures there was no marked deformation of the panel, at a higher pressure a noticeable increase in the deformation occurred similar to a 'divergence' of the panel. The experimental relationship found between this critical dynamic pressure and the tensile force in the panel had the same form as simple theory would predict.

In Ref. T.19 the panel was assumed to have an initial curvature and the ratio of final deflection (at a given airspeed) to the initial deflection was calculated as a function of tensile load in the panel, in-plane stiffness of the panel support structure and dynamic pressure. The correlation obtained with experiment was encouraging but not conclusive.

All the above studies have been described in greater detail in Ref. 5.



2.5 Subsonic Flutter of Rectangular Plane Membranes

An exploratory experimental investigation into this problem is reported in Ref. T.15. Two distinct types of flutter were observed for a given chordwise tension in the membrane. The first flutter occurred as a low frequency, small amplitude, travelling wave and the second, at a slightly higher speed, had a larger frequency and amplitude (Fig. 3). From the available test data, for membranes with their aspect ratio much less than unity, there appears to be a linear relationship between the membrane tension parameter (tensile force/span²) and dynamic pressure for both types of flutter.

2.6 Supersonic Divergence of Rectangular Flat Plate Aerofoils

Many papers have been published concerning the static aeroclastic stability of rectangular plate wings in supersonic flow. The usual method of investigation has been to specify the spanwise deflection mode precisely and to allow the chordwise distortion to be arbitrary. This is based on the assumption that only the chordwise mode is important in supersonic divergence. In surveying the literature on the subject it is clear that the effect of the choice of modal functions is so pronounced that aero-elastic instability could be either most likely or, as in some solutions, completely impossible.

Rhodes (Ref. T.18) has analysed this problem, using Ackeret's theory for the aerodynamics, in a Rayleigh-Ritz method. The deflection modes assumed are of the general polynomial form:

$$W = \sum_{m=0}^4 \sum_{n=2}^4 C_{mn} x^m y^n$$

Values of m and n up to 4 are used but clearly some values of C_{mn} are zero when the root boundary conditions of the wing ($y=0$) are satisfied.

Thus it can be shown that divergence only occurs at supersonic speeds for wing aspect ratios of less than unity and even then it is not a problem of practical concern. Further analyses are reported in Ref. T.18 using experimental and theoretical structural influence coefficients for flat plates of aspect ratio 1 and 2. The results show that as the plate stiffness decreases the lift of the elastic plate, for a given root incidence, also decreases and is never greater than the lift of the same rigid plate.

It must be emphasised that the above conclusions only relate to rectangular, flat plate wings clamped along their entire root chord. For wings having other profiles or planforms or different root constraints the same conclusions do not necessarily apply.

2.7 Static Aeroelastic Behaviour of Doubly Swept Wings (M - Wings)

A wing of M-planform has been considered as a possible compromise in design to avoid the undesirable aeroelastic properties of divergence and aileron reversal typical of swept forward and swept-back wings respectively. The choice of kink position, where the change of sweep occurs, is obviously of prime importance for optimum aeroelastic design.

Analyses are reported in Ref. T.35 for five wings having similar geometrical and elastic properties but different kink positions. Using a Rayleigh-Ritz approach, results are given for the divergence speeds and rolling powers of the various wings and compared with simple experimental studies. Both sets of results showed an increase of divergence speed as the kink position moved inboard. The results for the rolling power are less conclusive but indicated a decrease in rolling power as the kink position moved inboard.

3. Investigations Associated with the Morane Saulnier M.S. 760 'Paris'

3.1 Elevator Flutter of a T-Tail Aircraft

The College took delivery of a M.S. 760 'Paris' jet aircraft (Fig. 4) in December 1958 and in February 1960 after a stick force transducer had been fitted to the top of the control column an oscillation of the tail structure and elevators was observed. This oscillation was later recorded as an elevator vibration whose amplitude increased with increasing airspeed from 200 knots up to 320 knots above which speed it was considered unwise to proceed. On enquiry the makers confirmed that the flutter only occurred when a sufficient mass was added to the top of the control column.

It was decided to investigate the problem further as a student research study (Ref. T.13 and R.15) particularly since the aircraft has a T-tail. In the previous decade at least six aircraft with T-tails had been flown in Great Britain and at least two had experienced flutter of the fin and/or tailplane structure, viz. the Handley Page Victor and Gloster Javelin.

The subsequent theoretical investigations consisted of binary and ternary symmetric flutter calculations with, and without, the manual control circuit included. These showed the aircraft to be liable to flutter for mass distributions similar to that which existed at the time of the original incident. The actual modes used were (a) a wing bending mode that included some fuselage bending, with a frequency of about 10 c.p.s., (b) a fuselage vertical bending mode at about 18 c.p.s. and (c) elevator rotation. The results were in qualitative agreement with the observed effects.

The flight tests were most comprehensive. Firstly, the flutter characteristics of the basic aircraft, with empty wing tip fuel tanks, were established. Secondly, the effects of variations in many parameters were examined viz. control column inertia, aircraft centre of gravity, main

fuel tank content and wing tip fuel tank content.

These flights showed that the three most important parameters affecting the flutter of the aircraft are control column inertia, nose ballast and main fuel tank content. In fact the variation of in-flight damping with main fuel tank content was most complex.

The last series of flight tests was concerned with proving the safety of the aircraft, for variations in the same parameters as before, following the necessary modifications to prevent elevator flutter. The flutter calculations had shown the necessity for increased elevator mass balance - from 5.5 lb to 7.5 lb - where the elevator underbalance arose directly from the fact that the aircraft had a T-tail. It had been noticed previously that although the elevators themselves were statically balanced when out of the aircraft, and normally rested nose down when in the aircraft, they were in fact underbalanced. This was due to the elevator control run up the fin providing an underbalancing moment of 18 lb.in. When the complete circuit connected this was not apparent as the control column in the cockpit slopes back and so, statically, provides an overbalancing moment that over-rides the effect of the control run. However, when the aircraft is oscillating the displacement amplitude is much greater at the control run in the fin than at the control column. Thus the apparent static overbalance becomes a dynamic underbalance for the elastic modes considered, and it was the mass of the control run up the fin which provided the inertia coupling to cause flutter.

Some interesting results have been obtained from this investigation which suggest that a destabilising effect due to the internal damping of the aircraft may be occurring.

The main tank capacity is 950 litres. When it contains 600 litres the in-flight damping is markedly reduced compared to the case of a full tank. With 400 litres the damping is much the same but further reduction to 200 litres causes the damping to increase again. These trends are shown in Fig. 5 for the aircraft with zero nose ballast but with the stick force transducer fitted. The frequency ranges from 17.0 to 17.5 c.p.s. and the aircraft state was identical (apart from the nose ballast) with that in which it fluttered at 240 knots. Therefore Fig. 5 shows implicitly the effect of nose ballast on flutter - in that flutter did not occur for the condition described - and also the complex effect of main fuel tank content. Both effects could be due to the changes in mass - whereby frequency coincidence was increased; or to changes in mode - whereby the aerodynamic coupling of the elevator forces in the fuselage bending mode was increased. An alternative explanation could be that changes in internal damping are occurring with variations in nose ballast or in main tank fuel content to produce the results observed in Fig. 5. Some support for this hypothesis is found by examining the variation of the free surface area with volume of the fuel in the main tank (Fig. 6). It is easily seen that this variation follows a similar pattern to that observed from the

in-flight damping measurements except that the latter would appear to vary inversely as the free surface area. Energy dissipation in a standing wave system should be proportional to the free surface area of the fuel. That the inverse occurs could be due to the destabilising effect of the internal damping, viz., in addition to the stabilising dissipation, phase changes are caused which are most destabilising.

A closer examination of this effect has been recently made and will be reported on more fully later (Ref. T.23). Some results are shown in Fig. 7 where the variation of damping (in a ground vibration test) with main fuel content is given. The series of curves for different values of exciter current show the significant effect of this parameter on the measured damping at resonance in the frequency range 17.5 - 18.0 c.p.s. At higher exciter currents (and hence exciter forces) the fuel surface wave motions produce lower values of damping; this could be due to the surface waves breaking at the higher force levels.

It can also be seen that there is a general (but not marked) tendency for the damping to vary as predicted i.e. to increase and decrease with corresponding changes in the free surface area. The behaviour around 50% tank fullness of the curves for lower exciter currents can be explained by the fact that, at this fullness, the fuel free surface coincides with a horizontal baffle over about 50% of the area.

3.2 Static Aeroelastic Problems

The main purpose of the investigation reported in Ref. T.22 was to determine both by ground and flight tests the influence of aeroelastic distortion on certain control characteristics of the 'Paris' jet aircraft.

Initially the relevant stiffnesses of the aircraft were measured which affected the rate of roll, rate of pitch and wing flexure - torsion flutter and these were used to predict their actual quantitative effect on the behaviour of the aircraft. The measured stiffnesses were also compared with criteria given in Av.P.970. Chap 500* (where applicable). The military requirements were chosen as the Paris is closer in design and performance to existing military aircraft from which it itself was developed than to existing civil aircraft.

It was concluded that all the wing stiffness criteria in A.v.P. 970 are satisfied.

The fuselage vertical bending stiffness is greater than that required but the tailplane torsional stiffness requirement is not met. However, the

combination of fuselage bending and tailplane twist probably satisfies the spirit of the requirements.

It was also found that both the aileron and elevator control circuits are too flexible compared with the requirements.

Subsequently, the necessary flight tests were made to provide the required comparable flight measurements and only with regard to the wing flutter problem were the results wholly inconclusive. However, this problem is still being pursued and will be reported on more fully later (Ref. T.23).

All the flight tests were conducted at 10,000 ft. and the wings and fuselage were instrumentated for measurements of rates of roll and pitch. Also measured were the control surface deflections and the movement of the control column (pilot's input). Roll accelerations were also measured, better results being obtained from an angular accelerometer in the fuselage than from wing tip accelerometers.

Fig. 8 shows a plot of maximum rate of roll per aileron angle against the inverse of speed squared. The different sets of results indicate an apparent dependence on the direction of roll and show a distinct non-linear relationship. Fig. 9 shows the same results plotted with respect to the control column deflection. This plot obviously includes the effect of control circuit stretch but, more interestingly, a linear relationship is now found which shows less dependence on the direction of roll. The corresponding predicted aileron reversal speeds are 620 knots E.A.S. and 550 knots E.A.S. respectively, where the former value is only 10 knots lower than the value obtained from Av.P. 970 (which also does not include control circuit stretch).

By using the measured values in flight of control column movement and aileron angle it was possible to calculate the effective control circuit flexibility. The agreement with the ground stiffness test result is excellent, viz., .65 degrees of aileron movement per mean aileron hinge moment (lb.ft) cf. with .63.

Static aeroelastic effects on longitudinal stability and control are much more difficult to assess as more parameters are involved, e.g. fuselage flexibility, tailplane twist and tailplane mounting flexibility, elevator flexibility and control circuit stretch. On the 'Paris' it proved possible to differentiate only between fuselage bending and tailplane twist effects and the corresponding flexibilities were determined from the flight test data and compared with ground test results, viz.

	Flight Test	Ground Test
Fuselage Flexibility	10.60×10^{-5} deg/lb	8.36×10^{-5} deg/lb
Tailplane Flexibility	17.15×10^{-5} deg/lb	17.90×10^{-5} deg/lb

The type of manoeuvre chosen to assess these aeroelastic effects (preferred out of a total of four considered) was the 'pull out' from a dive. The aircraft was trimmed to fly straight and level at the required speed at 10,000 ft. and the tailplane setting was recorded. Without altering this setting the aircraft was taken to between 12,000 and 14,000 ft, depending on the required 'pull-out' speed, and a dive started. The 'pull-out' was judged to level out at exactly 10,000 ft. and the dive had to be steep enough to ensure a steady pitch rate by the time the plane was horizontal at 10,000 ft. The same engine power was maintained during trim and 'pull-out' to prevent any effects of thrust eccentricity on pitching moment. This manoeuvre was difficult but not impossible to attain. A gyroscope measured pitch rate and a force balance instrument measured pitch acceleration - the latter was not successful. An expression for the pitch rate of an aircraft (q) allowing for fuselage (k_4) and tailplane (k_6) flexibilities is

$$\frac{-q}{\Delta\eta V^2} = \frac{k_2(1 - k_6 V^2)}{k_1 V - k_0 V + k_4 V^3}$$

where k_0 , k_1 , k_2 include rigid aircraft aerodynamic terms and $\Delta\eta$ is the change in elevator angle. Fig. 10 compares the flight test data with predicted data based on ground test results, for the elevator effectiveness defined as in the equation above. The experimental results at the high speed end of the range are known to be suspect.

3.3 Wing-Aileron Flutter

This study is currently in progress (Ref. T.23) and the theoretical investigations have almost been completed. These analyses are intended to determine the effect of main tank and tip-tank fuel contents on the flutter characteristics of the wing-aileron. The flight test investigations to date have been less successful. A satisfactory method of excitation in flight using control surface jerks etc. has not been found and the records obtained show a poor signal-to noise ratio.

3.4 Flutter of Aircraft Aerials

Many failures of aircraft aerials have occurred on various types of aircraft during flight which have been attributable to stalling flutter brought about by the change of cross-section of the aerials through ice accretion at their leading edges. All these aerials had low internal damping and considerable bending flexibility - and flexure-torsion flutter was considered most unlikely. For blade aerials with a streamlined profile the usual flutter cure has been to mount the aerials with their thick leading edges aft. This artifice alters the shape of the ice then formed on the aerial and is found to prevent stalling flutter.

Blade aerial flutter has occurred on the 'Paris' jet aircraft during icing conditions even though the blade was mounted in the reversed sense

described above. The pilot was able to observe large amplitude oscillations before failure occurred.

Subsequent wind tunnel tests were made at Cranfield, on a similar aerial to that which fluttered, for various types of simulated ice patterns (Fig. 11). Plasticine was in fact used to simulate the ice. The blade aerial (type 386) had a length of 15.57", a root chord of 1.16" and a tip chord of 0.52". Its fundamental bending frequency was 45 c.p.s.

For small, symmetric, 'ice' accretions, small amplitude flutter occurred between 160-180 ft./sec. - being maximum at about $2\frac{1}{2}^\circ$ incidence to the airstream. No flutter was found at 200 ft/sec. but at 215 ft/sec. large amplitude flutter occurred which was greatest at 2° but zero at 3° and above.

Tests with larger amounts of symmetrical or asymmetrical icing (hence a cambered profile) showed similar results viz. maximum amplitudes at about 2° to 3° incidence and definitely no flutter above 6° for a very wide range of speeds.

The conclusion reached was that flutter can occur on reversed blade aerals but this can probably be prevented by mounting the aerals at a moderate angle of incidence.

4. Investigations on Design Project Studies

Aeroelastic investigations have been made on many Design Project Studies (see list of references) some of which will now be described.

4.1 Supersonic Delta Wing Airliner (1960)

Ref. T.16 contains an aeroelastic assessment of a supersonic airliner project (A.60) designed to carry 108 passengers on the transatlantic route, cruising at a Mach number of 2.2 between altitudes of 57,000 and 65,000 ft. The shape of the aircraft in plan is that of a slender delta wing with streamwise tips and having no horizontal tail (Fig. 12). A basic structure in aluminium alloys was chosen and the wings were of multispar, multi-rib construction. The fuselage comprised two shells - an inner pressure shell and an outer shell carrying the external loads. The aircraft all-up weight was 325,000 lb.

The aeroelastic effects analysed were elevator and aileron effectiveness, defined in terms of the initial pitch and roll acceleration per change in control angle, where the controls were mounted on the wing trailing edge. The aerodynamic forces were calculated using slender body theory and linear piston theory and the analyses were made for flight at $M = 2.2$.

In the calculation of elevator effectiveness the planform was idealised as a chordwise beam with rigid spanwise sections, and at 57,000 ft. the effectiveness was determined as 89% of the value for the same rigid aircraft.

For aileron effectiveness a semi-rigid approach was used involving nine elastic modes representing asymmetric plate action of the structure about the fuselage longitudinal centre line. A value of aileron effectiveness of 54% was obtained for flight at 65,000 ft.

4.2 V.T.O.L. Freighter (1961)

This project (F.61) shown in Fig. 13 was a V.T.O.L. version of the conventional freighter design studied in 1959 (F.59). The effect of the large V.T.O.L. engine pods mounted at 65% semispan of the wing, on the flutter of the wing, was studied and is reported in Ref. T.17. In the absence of more realistic data at the time the stiffness distributions of the F.59 wing were assumed for the F.61 wing. The influence on the corresponding natural frequencies is shown in the following table:

Modal Frequencies (c.p.s.)		
Mode	F.59	F.61
Fundamental Flexure	3.74	2.89
Fundamental Torsion	22.20	6.05

Flutter calculations for the conventional (F.59) wing using the AV.P.970 criterion gave a flutter speed of 658 knots which was satisfactory since $1.25 V_D$ was 438 knots. A similar calculation for the V.T.O.L. (F.61) wing using the appropriate frequencies gave a flutter speed of only 164 knots.

More detailed calculations were made on the F.61 wing using the theoretical vibration modes and flutter was found for all speeds considered above 200 knots. Subsequent investigations suggested that the term having the greatest influence on the flutter has the form

$$\bar{A}_{\alpha\alpha} = 1 + \rho A_{\alpha\alpha} / \bar{I}_{\alpha}$$

and represents the combined aerodynamic ($A_{\alpha\alpha}$) and inertia (\bar{I}_{α}) forces in the torsion mode. To obtain a reasonably high flutter speed $\bar{A}_{\alpha\alpha}$ should be negative and for the F.59 wing the appropriate values were $\rho A_{\alpha\alpha} = -38,000$ slugs ft² and $\bar{I}_{\alpha} \approx 12,000$ slugs ft². For the F.61 wing however, $\bar{I}_{\alpha} \approx 140,000$ slugs ft² so that the sign of $\bar{A}_{\alpha\alpha}$ becomes positive. These results confirm that wings with large concentrated inertias and masses need to be examined closely for flutter.

4.3 Supersonic Canard Airliner

Ref. T.20 describes a number of theoretical aeroelastic investigations which have been made on the Mach 3 Design Project Study (A.62) shown in Fig. 14. It should be noted that pitch control is achieved by trailing edge

flap movement on the canard lifting surface (the foreplane).

The effect of fuselage deformation on the longitudinal bending moment distribution was determined for trimmed flight at $M = 3.0$ and 68,000 ft. altitude at the beginning of cruise. As in the other investigations made it was assumed that spanwise sections were rigid. A direct method of analysis for the trim condition showed a maximum increase in the bending moment, in the region of the centre fuselage, of 26% compared with rigid aircraft values.

An iterative method was also used in which the separate deformation modes were first calculated for the cases $\alpha \neq 0, \beta = 0$ and $\alpha = 0, \beta \neq 0$ ($\alpha =$ body incidence, $\beta =$ foreplane flap incidence). For trim the condition of zero pitching acceleration gave a relationship between α and β from which the longitudinal bending moment distribution could be found. The results agreed with those above but the method was more lengthy.

Using the results for the case $\alpha = 0, \beta \neq 0$ the initial pitching acceleration per change in foreplane flap angle was determined as a measure of pitch control effectiveness. An increase of only 3%, compared with the rigid aircraft, was obtained.

An iterative method was used to obtain the fundamental, longitudinal body bending mode (frequency 2.4 c.p.s.) and this was used, with two rigid body modes and a foreplane torsion mode, in a flutter investigation. This study has been extended in a current investigation (Ref. T.24) which is making use of a dynamic model, described in Ref. T.21, to obtain more realistic modal shapes. The theoretical studies in Ref. T.24 are using three aircraft elastic modes in a six degree of freedom study.

The effects of the fuselage flexibility upon the dynamic stability of a supersonic canard aircraft at $M = 2$ have been examined in Ref. T.34. The fuselage was presumed to bend in the manner of a uniform cantilever beam, 'built in' at the centre of gravity of the aircraft.

Using Lagrange's equations certain expressions were obtained which had to be satisfied for the aircraft to be dynamically stable. The main parameters introduced into the stability contours were fuselage mass, length and moment of inertia, fuselage flexibility C and control surface area.

It was found that the damping of the aircraft is increased as C is increased but, consequently, the frequency is reduced, leading to a decrease in static stability.

Further, as C is increased the response to pitch control movement is also increased.

4.4 General Purpose Freighter (1963)

On this aircraft a flutter investigation is being made of the spring tab-aileron-wing system. The analysis is conventional and uses well-established techniques but it is hoped to make an appraisal of various methods of calculating the aerodynamic derivatives for such configurations.

This study is to be reported in Ref. T.25.

5 Helicopter Problems

5.1 Analyses Including Aerodynamic Forces

In Ref. T.36 a method of helicopter blade vibration analysis is presented in which special account is taken of the forces due to unsteady axial and in-plane flows, about the rotor blades in the vertical and forward flight conditions.

Equations of motion are derived, in terms of the flapping, lagging and torsional motions, which contain the effects of elastic, inertial, aerodynamic and control forces. The periodic aerodynamic damping associated with forward flight rendered these equations non-linear. Difficulty was found in linearising these equations and in assessing the effect of the skewed, helical wake system on the unsteady aerodynamic forces in forward flight.

It was concluded that accurate vibration analyses are possible in the vertical flight condition but for forward flight much more research is required on wake effects on the unsteady aerodynamic forces, and on effective linearisation of the aerodynamic derivatives.

With a view to improving the flapping stability of an articulated rotor an analysis has been made (Ref. T.37) of the flapping stability of a blade with two flapping hinges. The introduction of the second hinge allows an additional degree of freedom in the flapping system, the motion of which is described by a pair of simultaneous linear differential equations with periodic coefficients. The second hinge is located outboard of the hub.

When there is no damping or spring restraint at the outboard hinge a maximum gain in stability is obtained when the motions in the two degrees of freedom have the same frequency but are anti-phased; this permits the elimination of one of the variables from the equations of motion.

A more general analysis has been made incorporating hinge constraints which, to satisfy the above criterion, yields expressions defining these constraints and the blade parameters as functions of the location of the outer hinge. It is thus found possible to eliminate from the equations of motion the second harmonic terms which are predominantly destabilising when

the advance ratio is greater than unity. The resultant system shows a gain in the flapping stability.

5.2 Helicopter Ground Resonance

Although this problem is not truly an aeroelastic one, the nature of the self-excited oscillations which can occur in ground resonance shows considerable similarity to flutter.

A useful summary of this work already exists (Ref. 9) and will not be described further.

6. Acknowledgements

The author is indebted to Messrs. D. Howe and E.A. Boyd, Lecturers at The College of Aeronautics, for valuable assistance in providing illustrations and data for use in this paper.

List of references

T - Unpublished theses*

Department of Aircraft Design

- T.1 Radok, J.R.M. Dynamic aeroelasticity of aircraft with swept wings, 1952 (see Ref. R.7).
- T.2 Sevier, H.G.: An experimental investigation of helicopter ground resonance, 1952.
- T.3 Smith, G.C.C.: Aeroelastic calculations on the 1951/52 Design Fighter Project, 1952.
- T.4 Cooke, A.: An investigation of the theory of self-excited mechanical oscillations of a helicopter, 1953.
- T.5 Davies, G.L. Flight resonance testing - a technique of investigation, 1954.
- T.6 Kemp, T.D.: Theoretical determination of the optimum distribution of wing skin thickness for various planforms to meet the condition of absence of aileron reversal at supersonic speeds, 1955.

*These are submitted in partial fulfillment of the requirements for The Diploma of The College of Aeronautics. They are retained in the College Library where they may be consulted by appointment.

- T.7 Reed, P.R.: Ground resonance testing of the Skeeter helicopter dynamic model, 1957.
- T.8 Coates, T.: Calculation of flutter speeds for the 1957-58 Design Project wing using influence coefficient methods, 1958.
- T.9 McLaren, W.S.C.: Design of a segmented flutter model and development of a construction technique using plastic foam and glass reinforced resin, 1958.
- T.10 Pickwick, P.J.: Aeroelastic problems on modern aircraft, 1959.
- T.11 Davies, R.J.: An aeroelastic investigation of thin panels, 1960 (See Ref. 4 and 5).
- T.12 Roe, D.: Design for a segmented low-speed flutter model of the 1957-58 Design Project tail unit, 1960.
- T.13 Mitchell, C.G.B.: A theoretical and experimental investigation of the flutter of the tail unit of the M.S. 760 'Paris', 1961. (See Ref. R.15).
- T.14 Nayler, G.H.F.: Flutter of circular panels in supersonic flow, 1961. (See Ref. 4 and 5).
- T.15 Thomas, M.G.: Some investigations into panel and membrane aeroelastic instabilities, 1961 (See Ref. 5).
- T.16 Turner, M.R.: An overall aeroelastic assessment of a supersonic aircraft, 1961.
- T.17 Momirski, M.: Flutter characteristics of the V.T.O.L. F.61 strategic freighter wing, 1962.
- T.18 Rhodes, A.J.: The chordwise divergence of rectangular flat plate wings at supersonic speeds, 1962.
- T.19 Sykes, D.R.: Deflection of two dimensional panels under subsonic aerodynamic loading, 1962 (See Ref. 5).
- T.20 Gresswell, D.J.: Some aeroelastic investigations on the A.62 Mach 3 airliner project, 1963.
- T.21 Suiter, B.E.: Wing flutter analysis of the A.62 Mach 3 Design Project, 1963.
- T.22 Wilkie, D.G.M.: Some aeroelastic investigations on the M.S.760 'Paris', 1963.

- T.23 Jennings, P.: Wing-aileron flutter of the M.S. 760 'Paris', 1964.
- T.24 Riley, S.: The flutter analysis of the A.62 Mach 3 airliner project, 1964.
- T.25 Williams, D.: Spring tab-aileron-wing flutter investigation of the G.P. 63 General Purpose Freighter Design project, 1964.

Department of Aerodynamics

- T.26 Earle, D.H.: The influence of distortion on stability and control, 1948.
- T.27 Ross, R.J.: Aeroelastic effects and flutter, 1948.
- T.28 Harris, K.D.: Flutter derivatives of swept wings, 1950.
- T.29 Beverley, R.J.: Flutter derivatives for wings of small aspect ratio at supersonic speeds, 1951.
- T.30 Campion, B.S.: Estimation of the effects of distortion on the longitudinal stability of swept wing aircraft at high speeds (sub-critical Mach numbers), 1953 (See Ref. R.9)
- T.31 Milne, R.D.: Flutter of wings of low aspect ratio, 1954.
- T.32 Caldwell, J.: Flutter derivatives of swept-back wings of finite aspect ratio at supersonic speeds, 1955.
- T.33 Martin, E.C.: Wing divergence at supersonic speeds, 1955.
- T.34 Thomson, F.: Effect of forward fuselage flexibility on the longitudinal stability of a tail first supersonic aircraft, 1959.
- T.35 Chester, D.H.: The static aeroelastic behaviour of doubly-swept wings (M-wings), 1960.
- T.36 Eden, M.S.: An analysis of rotor blade vibration with particular reference to the influence of the wake and unsteady flow conditions, 1961.
- T.37 Sawney, A.K.: 'A two flapping hinge arrangement for improved flapping stability of helicopter blades, 1963.

R - College Reports

- R.1* Duncan, W.J.: Ignorance of distortional co-ordinates in the theory of stability and control, College of Aeronautics Report 1, December, 1946.
- R.2* Robinson, A.: On some problems of unsteady supersonic aerofoil theory, College of Aeronautics Report 16, May 1948 (Reprinted in 7th Int. Cong. App. Mech. Vol. 2 p.p. 500-514).
- R.3* Duncan, W.J.: Flutter of systems with many freedoms, College of Aeronautics Report 19, August, 1948.
- R.4* Babister, A.: Flutter and divergence of swept back and swept forward wings, College of Aeronautics Report 39, June 1950 (Reprinted as ARC. R and M 2761).
- R.5 Buchan, A.L., Harris, J., Somervail, P.M.: Measurement of the derivative Z_w for an oscillating aerofoil, College of Aeronautics Report 40, June 1950 (Reprinted as ARC. CP. 52).
- R.6* Radok, J.R.M.: Aileron reversal and divergence of swept wings with special considerations of the relevant aerodynamic and elastic characteristics, College of Aeronautics Report 55, March 1952.
- R.7* Radok, J.R.M.: Dynamic aero-elasticity of aircraft with swept wings, College of Aeronautics Report 58, April, 1952.
- R.8* Lilley, G.M.: An investigation of the flexure-torsion flutter characteristics of aerofoils in cascade, College of Aeronautics Report 60, May 1952.
- R.9* Campion, B.S.: Estimation of the effects of distortion on longitudinal stability of swept wing aircraft at high speeds (sub-critical Mach numbers) College of Aeronautics Report 77, January, 1954 (Reprinted in Aero. Quart. Vol. 5, pp. 280-322, 1954).
- R.10 Whitmarsh, G.E.: Measurement of the derivative Z_w for oscillating sweptback wings, College of Aeronautics Report 92, July 1955.
- R.11 Milne, R.D., Willcox, F.G.: Measurement of the derivative Z_w for oscillating wings in cascade, College of Aeronautics Report 93, July 1955.
- R.12* Milne, R.D.: The unsteady aerodynamic forces on deforming low aspect ratio wings and slender body combinations oscillating harmonically in compressible flow, College of Aeronautics Report 94, July, 1955.

* Items marked thus are out of print.

- R.13* Craven, A.H., Davidson, I.: The flexure-torsion flutter of cambered aerofoils in cascade, College of Aeronautics Report 95, December, 1955.
- R.14 Craven, A.H.: The aerodynamic derivatives of an aerofoil oscillating in an infinite staggered cascade, College of Aeronautics Report 125, January, 1960.
- R.15 Mitchell, C.G.B., Johns, D.J.: A theoretical and experimental investigation of tail unit flutter on the M.S. 760 'Paris', College of Aeronautics Report 154, December 1961.

Other References

1. Johns, D.J.: Some panel flutter studies using piston theory. J. Aero/space Sci. 25 (11): 679 November, 1958.
2. Johns, D.J.: Parks, P.C.: Effect of structural damping on panel flutter, Aircr. Engng. 32, 304, October 1960.
3. Johns, D.J.: The influence of panel deformations on wing flutter, J. Aero/space Sci. 27(2): 137, February, 1960.
4. Johns, D.J., Davies, R.J., Naylor, G.H.F.: The flutter of thin, plane, elliptic panels, College of Aeronautics Note 136, January 1963.
5. Johns, D.J.: Some panel aeroelastic instabilities, AGARD Rep. 474, September, 1963.
6. Johns, D.J.: Supersonic flutter of cylindrical shells, College of Aeronautics Note 104, July 1960.
7. Johns, D.J.: Supersonic flutter of a cylindrical panel in an axisymmetric mode, J. Roy. Aero. Soc. 64: 362, December 1960.
8. Johns, D.J.: Some design problems of thin shell structures, College of Aeronautics Note 118, September 1961.
9. Howe, D.: The use of models in helicopter vibration investigations, Aeroplane, January 1959.

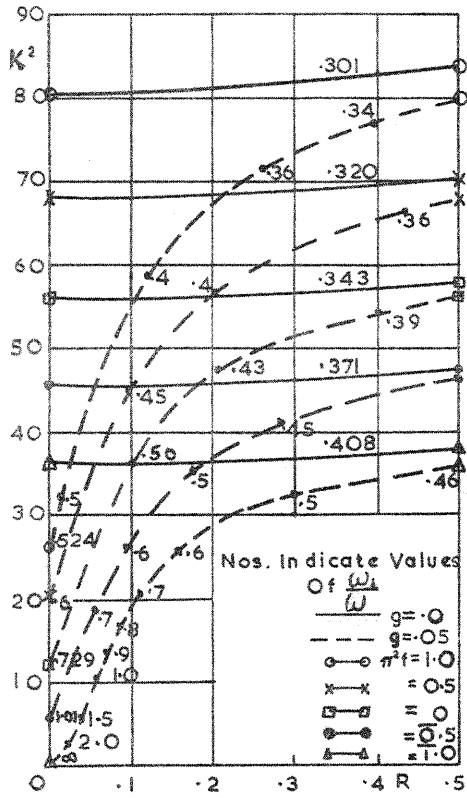


FIG. 1

VARIATION OF FLUTTER PARAMETER, K^2 FOR R AND f FOR $g = 0$ AND $g = 0.05$ (FROM REF. 2).

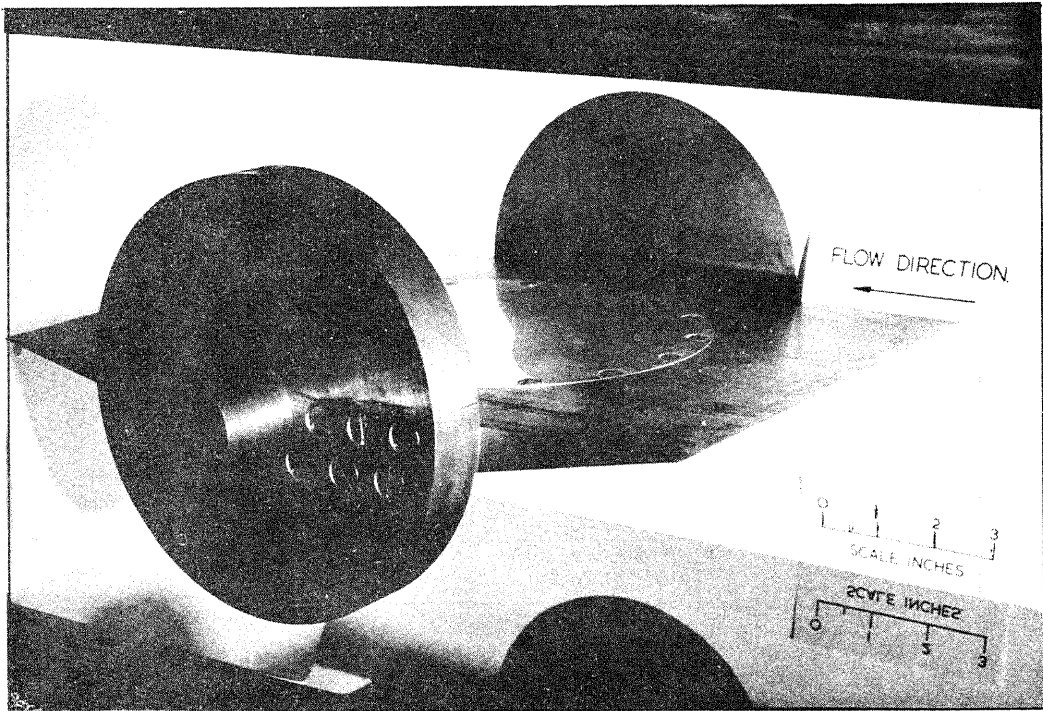


FIG. 2 FLUTTER MODEL WEDGE IN WIND TUNNEL



FIG. 3 SECOND TYPE OF MEMBRANE FLUTTER

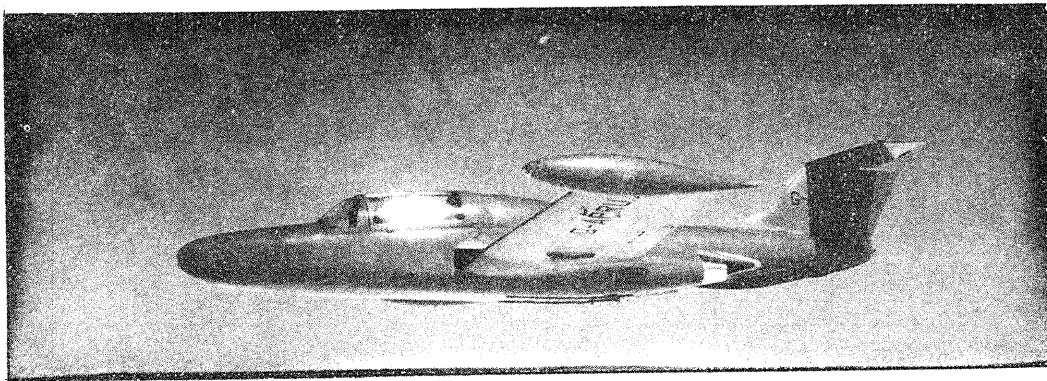


FIG. 4 MORANE-SAULNIER M. S. 760 PARIS

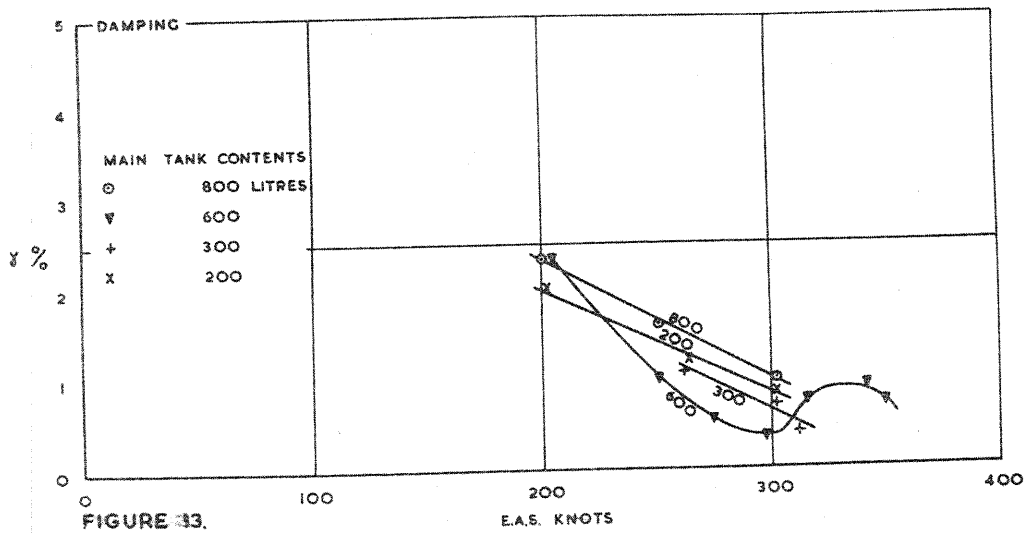
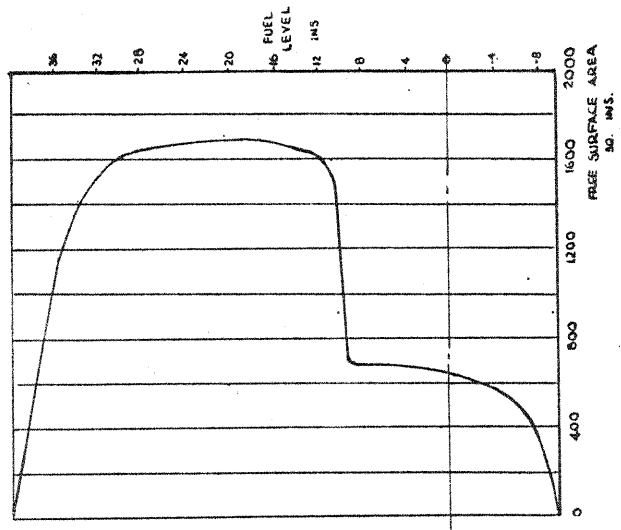
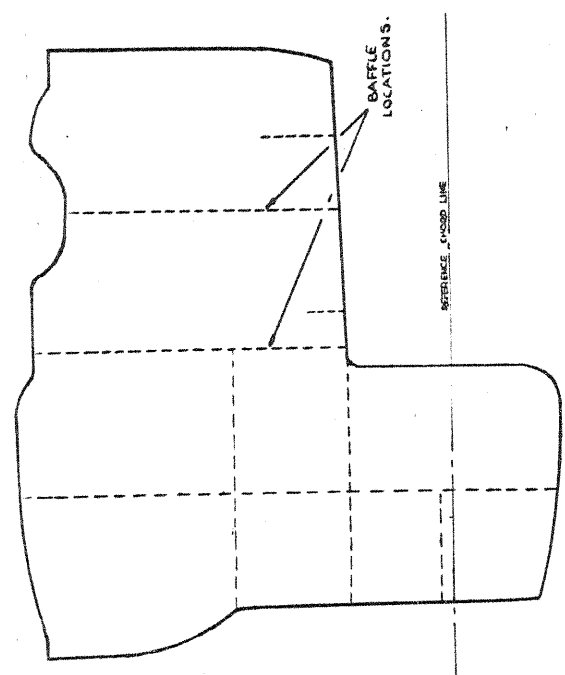


FIG. 5 DAMPING OF SYMMETRIC VIBRATIONS ZERO NOSE BALLAST; STICK FORCE TRANSDUCER FITTED



6b



6a

FIG. 6

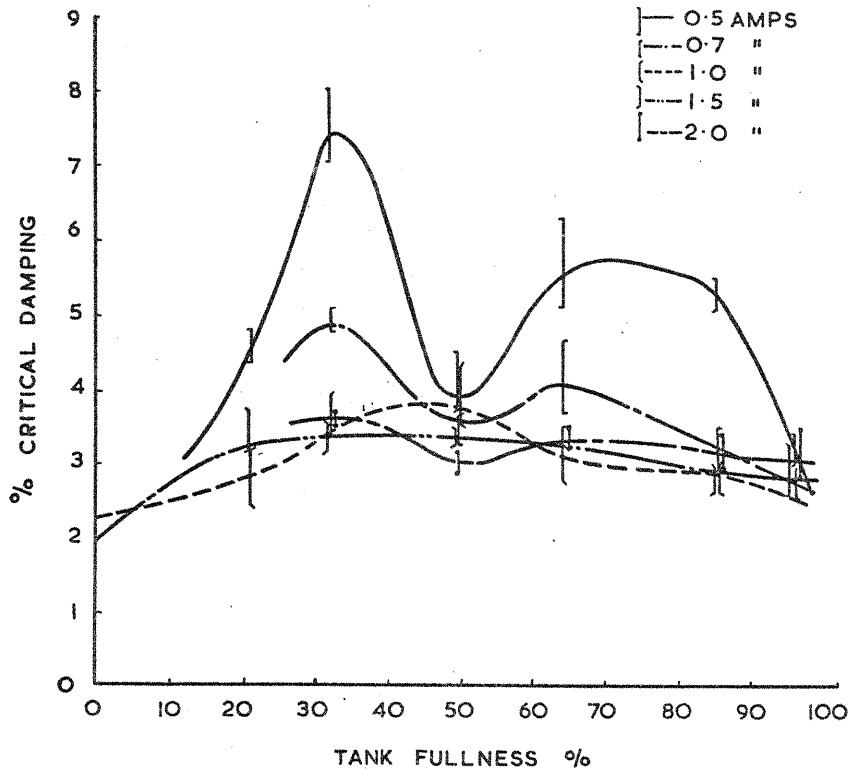


FIG. 7 DAMPING OF ELEVATOR ROTATION MODE VERSUS MAIN TANK FUEL CONTENT FOR A RANGE OF EXCITER CURRENTS

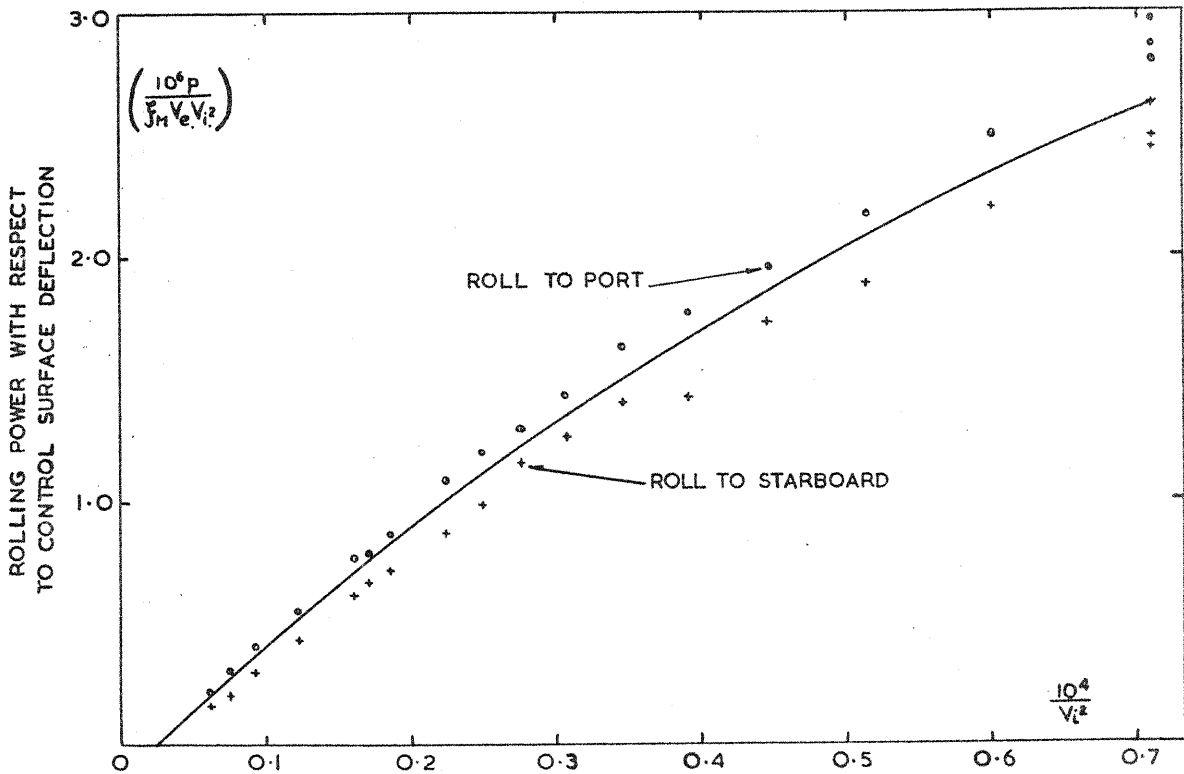


FIG. 8

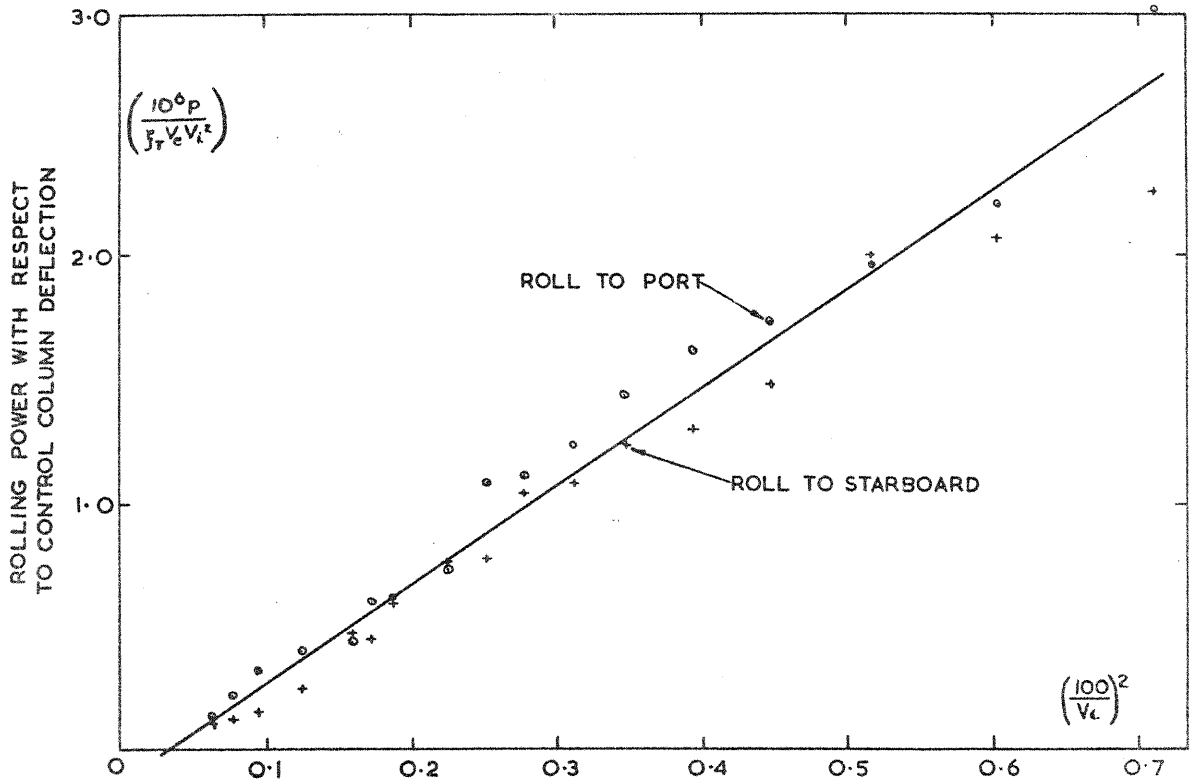


FIG. 9

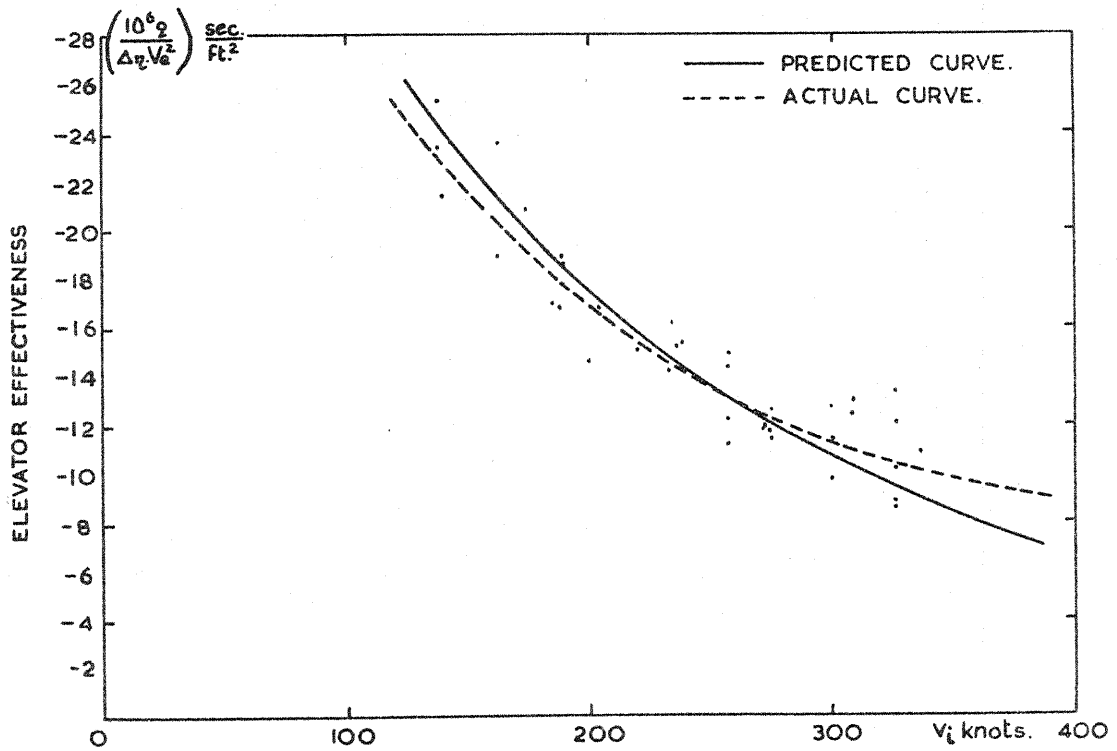


FIG. 10

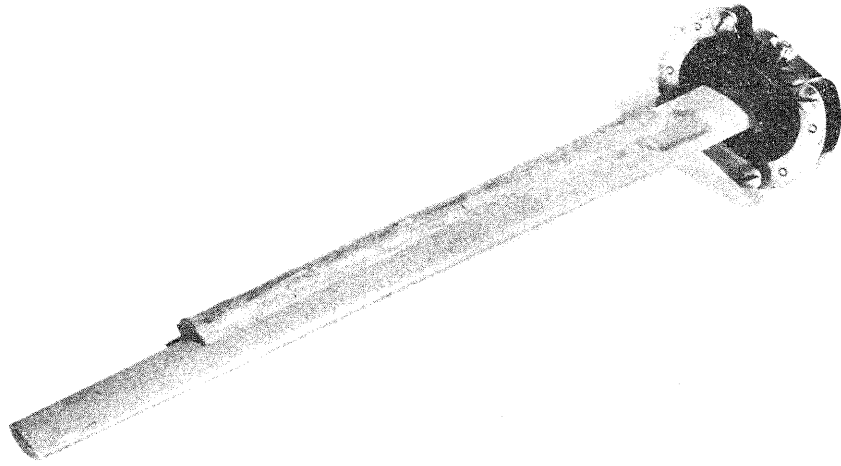


FIG. 11 AERIAL SHOWING SIMULATED ICE ACCRETION FOR WIND TUNNEL FLUTTER TEST

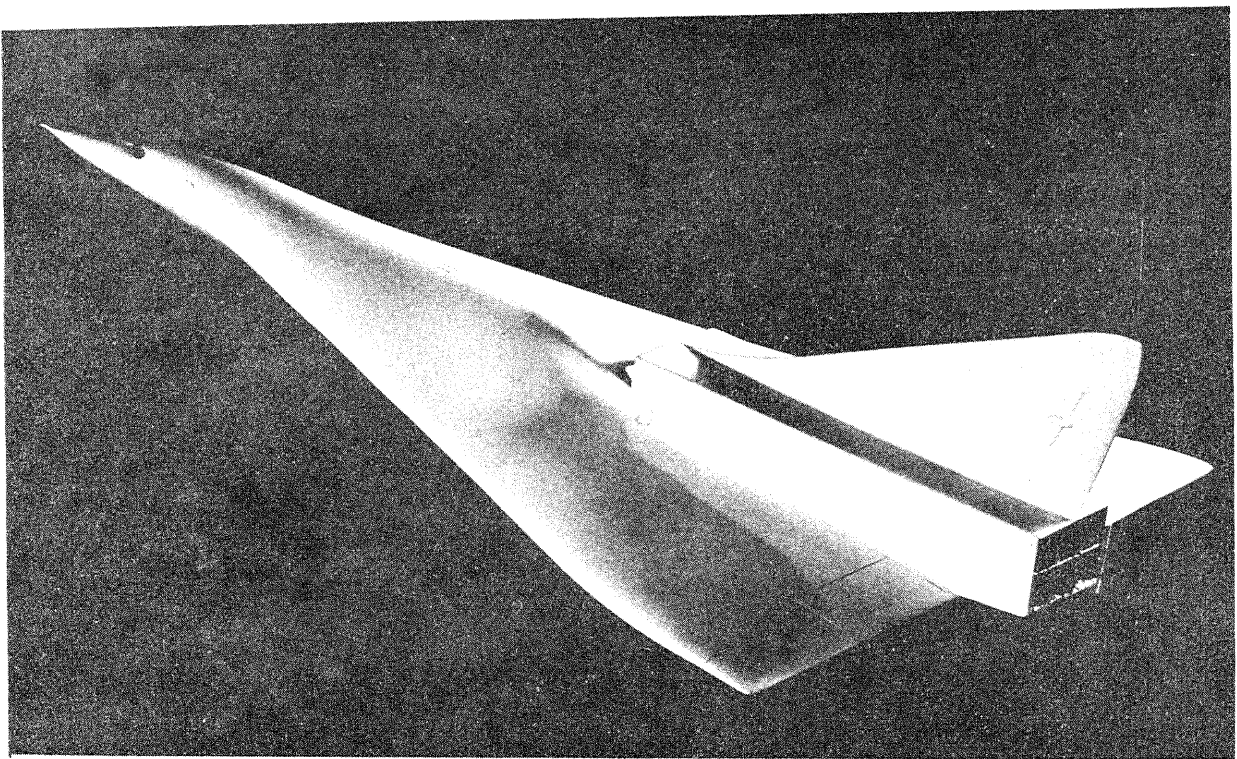


FIG. 12 MACH 2.2 SUPERSONIC AIRLINER PROJECT

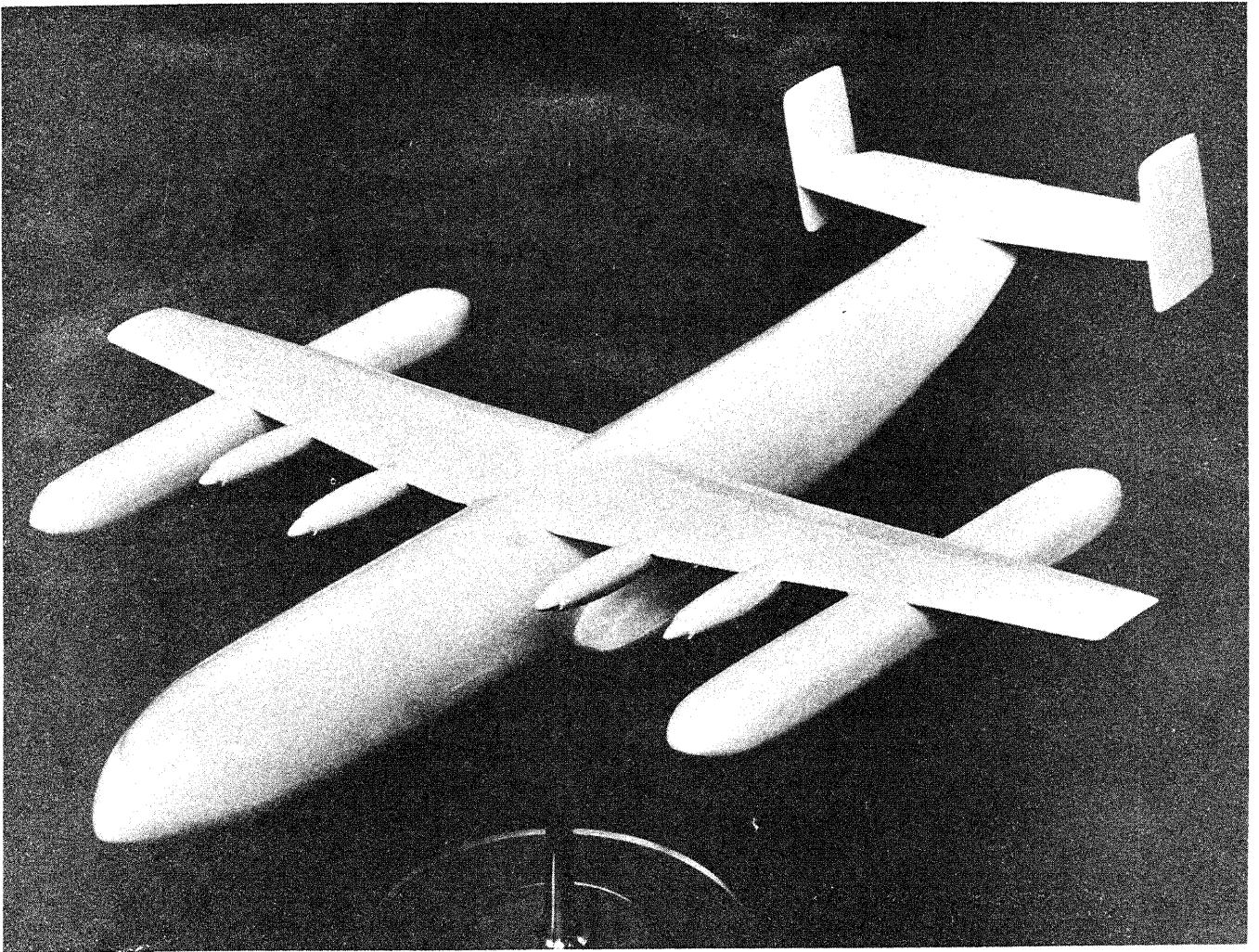


FIG. 13 V. T. O. L. FREIGHTER PROJECT

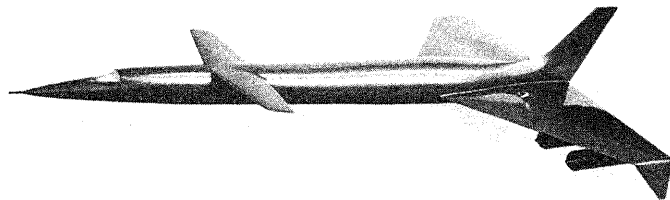


FIG. 14 MACH 3.0 SUPERSONIC AIRLINER PROJECT