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A Review of Comparative Theoretical and Experimental Aerodynamic Data Relevant to Zero- and Low-Frequency Aeroelastic Problems

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Summary

This Report provides, in tabular form, résumés of the contents of reports which give comparative theoretical and experimental aerodynamic loading data and, in some cases, structural deformation data, relevant to zero- and low-frequency aeroelastic problems. There is a broad classification of reports according to whether data have been obtained from tests on nominally rigid models with built-in warp or from tests on flexible models or full-scale aircraft. Details of reports which contain only experimental data are also tabulated.

From an analysis of the data it is concluded that, when the flow is attached and the flow field is wholly subsonic or wholly supersonic, existing theoretical methods provide the basis for estimates of aeroelastic distortion effects which are adequate for engineering purposes. There is, however, an urgent need to develop theoretical and/or empirical methods of estimating such effects for important design cases which occur outside these flow régimes.

LIST OF CONTENTS

* Replaces RAE Technical Report 70089—A.R.C. 32 451

1. Introduction

In recent years there has been an increasing realisation, on the parts of aircraft designers and research workers alike, of the need to take detailed account of possible aeroelastic effects from the earliest stages of the design of new aircraft, in order to achieve optimisation of performance, stability and handling characteristics in conjunction with economy in structural weight. Thus, for instance, when discussing this need in an unpublished RAE Memorandum, the present author suggested that the degree of structural flexibility of current and future high-speed aircraft was 'such as to make it highly desirable, if not imperative, to take account of aeroelastic effects when assessing the effect of changes in aerodynamic, structural and mass configuration on such design parameters and features as:

1. jig shape required in order that the deformed shape in flight at the design cruise condition shall be the optimum for performance;
2. L/D ratio under design and off-design conditions;
3. $C_{L_{max}}$, particularly under limit load conditions;
4. buffet boundaries;
5. stability and control derivatives and associated trim and manoeuvre discriminants.

Accurate theoretical assessment of such effects is dependent on the provision of reliable aerodynamic and structural input data as well as on the availability of methods of solving the aeroelastic equilibrium equations with good accuracy. The present paper, prepared at the instigation of a Working Party on Aeroelastic Deformation Effects, set up by the Aeronautical Research Council in 1969, is concerned only with the aerodynamic-input aspect of the overall problem. Its object is to compare existing theoretical and experimental aerodynamic data relevant to zero- and low-frequency aeroelastic problems and thereby to establish the range of applicability and accuracy of theoretical methods.

Essentially the aerodynamic problem involved in aeroelastic investigations is that of predicting the load distributions under conditions corresponding to all points of typical flight envelopes, over lifting surfaces having various planforms and incorporating arbitrary distributions of 'warp' (camber + twist), characteristic of those which result from aeroelastic deformation. In this context difficulties can arise when dealing with the cases of flight at high incidence or at transonic speeds, for both of which, conditions of mixed flow exist over the aerofoil surfaces. In these circumstances linearised potential theory is invalidated and satisfactory alternative theories are not yet available. It must be conceded at the outset that the resolution of these difficulties constitutes a major problem of research in the field of general aerodynamics, irrespective of its relevance to aeroelastic problems. In the present report the author has tried to establish whether one may consider as solved the more straightforward problem of predicting loading over aeroelastically warped wings which are operating under conditions for which linearised potential theory might be expected to provide an adequate basis for calculation. To this end, the existing literature has been examined and the available experimental evidence relating to the simpler problem has been assembled herein.

Some attention had previously been given to this problem by Hancock[1], in a general paper on the estimation of aerodynamic loads in steady manoeuvres, published in 1963. In this he commented on the paucity of experimental data available for checking the validity of methods for estimating loads on flexible wings. Thus it was not expected that a new review of the literature would reveal a great abundance of experimental evidence relating to this topic. In the event, the author's search brought to light a rather larger number of relevant papers than had been anticipated. All of the data revealed by this latest literature search have been presented here in a convenient tabular form to provide designers and research workers with a readily accessible state-of-the-art summary of the current situation.

2. Scope of the Literature Review

2.1. Techniques Used in the Acquisition of Aeroelastic Data

If we consider an actual (flexible) aircraft in steady or quasi-steady manoeuvring flight, a complete check on the accuracy of theoretical predictions of its aeroelastic equilibrium state would necessitate measurements of:

(a) parameters defining the overall ('rigid-body') attitude and motion of the aircraft,
(b) the static deflection characteristics of the structure (determined in ground tests),
(c) pressure distribution over the surfaces of the aircraft (with, possibly, supporting strain-gauge measurements of resultant loads at selected stations), and
(d) elastic deformations in flight.

Because of the complexity and expense of such an experimental programme one could expect to find few, if any, examples reported in the literature and, in fact, no record has been found of any such programme which included pressure distribution measurements, although there is at least one example where all the other measurements, including strain-gauge measurements of resultant loads at a number of spanwise wing-stations, have been made. However, it is to model tests, either in wind-tunnels or in free-flight that one must look for the bulk of the experimental data bearing on our problem.

There have been two main lines of approach in the use of models to acquire basic aeroelastic information. One involves the testing of flexible models while the other utilises models which are nominally rigid but which incorporate built-in 'warp' of the wing (twist and/or camber) typical of that acquired as a result of elastic deformation in flight. Provided that the appropriate pressure distribution (or resultant load) and model deflection measurements are made in the tunnel, and supplemented by calibration of the model to determine its static deflection characteristics, the former approach yields information which can be used to check the accuracy of theoretical predictions of both the aerodynamic and the structural characteristics involved in the aeroelastic equilibrium problem. The latter (rigid model) approach yields data appertaining only to the aerodynamic side of the problem. It should, perhaps, be observed, in passing, that data obtained from tests of nominally rigid models are of dubious value unless it has been confirmed that deflections of the models were, in fact, negligibly small throughout the range of test conditions.

2.2. Literature Search

Having regard to the above discussion and to our present preoccupation with the aerodynamic-input-data aspect of aeroelastic work, a search was made, with the assistance of R.A.E. Library staff, for reports purporting to give experiment/theory comparisons of pressure and/or loading distribution data, or of overall forces and moments, as determined from tests in the subsonic, transonic and supersonic régimes, on full-scale aircraft, flexible models and nominally rigid models incorporating built-in camber or twist. Regarding the latter category of model, interest resides mainly in those models for which, with the aeroelastic deformation problem in mind, a considerable degree of camber or twist has been incorporated. However, it may be noted that the camber and twist distributions which are currently being incorporated in wing profiles to achieve optimum cruise performance are just as severe as those resulting from elastic deformation in non-cruise conditions. Thus data obtained in some fairly recent performance-oriented investigations of the use of camber and twist would be considered relevant to the present investigation, while most of the older pressure-plotting data, obtained in routine tests on 'rigid' models of wings with little or no camber and twist, could be regarded as superfluous, inasmuch as it is already generally accepted that lift distributions over such wings at moderate incidences can be estimated with acceptable accuracy by linearised lifting-surface theory.

2.3. Analysis of Relevant Literature and Presentation of Data

As intimated in the Introduction, tabulation of relevant data in a standardised form was a procedure which appeared equally convenient as regards extraction of information from individual reports and its subsequent presentation as the basis of a state-of-the-art review. The main object of the analysis has been the extraction of comparative experimental and theoretical data, and information under this heading may be broadly classified according to whether the experimental data have been obtained from tests on nominally rigid models, with built-in warp (i.e. camber and/or twist), or on flexible models and full-scale aircraft. The basic tabulation of information presented here, in Tables 5 and 6, is therefore in accordance with this subdivision. Within each table items are arranged in approximately chronological order according to the dates of the corresponding reference reports.

It is hoped that, besides providing the basis for a general appraisal of the applicability of current methods of calculating incremental aerodynamic loadings for aeroelastic studies, the data presented will be of use to designer and research worker alike, in their respective tasks of evaluating particular configurations and of closing the gaps in our theoretical knowledge. To facilitate these tasks, an index to Tables 5 and 6 is provided by Tables 1 to 3, each of which summarises the information available for aircraft configurations of one particular type. The types in question are tapered swept-wing configurations (Table 1), delta-wing configurations (Table 2) and trapezoidal- (including rectangular-) wing configurations (Table 3). Where appropriate there is subclassification according to the origin of the experimental data (i.e. from tests either on nominally rigid models or on flexible models and full-scale aircraft). Within the 'Index' tables, items are arranged in ascending order of wing aspect-ratio.

The literature search brought to light a number of reports containing only unanalysed experimental data, some of which have, however, been used in comparisons with calculated data in reports included in Tables 5 and 6. Since they represent a potentially useful, but largely untapped, source of experimental information these 'data-only' reports are listed in Table 4, which indicates the scope of the data which they contain.

A column in each of Tables 5 and 6 gives information about the theoretical methods employed to obtain results for comparison with experimental data. In some instances such methods are developed in the report under consideration; more generally reference is made to other reports and these are listed as items 42 to 72 of the list of references appended to the present Report.

3. Discussion

3.1. General Comments

A fairly thorough, though not necessarily exhaustive, search of the literature has been made to discover reports which provide evidence as to the applicability of the various theoretical methods of predicting aerodynamic loading data relevant to zero- and low-frequency aeroelastic problems. The search has revealed a rather larger accumulation of information than had been anticipated. Thus Tables 5 and 6, which present particulars of all reports which give comparative experimental and theoretical data, contain a total of twenty-seven items, involving twenty-nine reports, while Table 4, in its eight items, lists a further eleven reports which provide experimental data only. Practically all of the reports are of American origin* and most of them are at least ten years old.

As can be seen from Tables 1(A) and 1(B) there are data providing a fairly wide, if not very dense, coverage of tapered swept-wing configurations (aspect ratios from 2·24 to 10·0) operating in the various flow régimes (subsonic, transonic or supersonic) appropriate to the respective configurations. Experimental data have been derived in fairly equal proportions from tests on nominally rigid models on the one hand (Table 1(A)) and on flexible models or full-scale aircraft on the other (Table 1(B)).

The data available for delta-wing configurations, which are indexed in Tables 2(A) and 2(B), cover a range of aspect ratios from 0·424 to 4·0. In the category of trapezoidal-wing configurations (Table 3) the published literature yielded only one example (a rectangular wing of aspect ratio 2·0) for which comparative experimental and theoretical data had been obtained.

Additional experimental data are provided by the reports listed in Table 4. For the most part these do not extend the range of configurations covered by Tables 1 to 3; in fact much of the information duplicates or supplements that already presented for the configurations under consideration, in the reports quoted in those tables. However, there are useful extensions in respect of such matters as the modes of warp built-in to nominally rigid models, the range of Mach number covered, and the form of the data presented (e.g. pressure distributions or section force and moment coefficients).

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* The literature search was not deliberately restricted to reports written in English but of the few foreign-language reports examined none was found to provide relevant information.

With regard to the sub-division of experimental data according to whether they have been obtained from tests on 'nominally rigid' models or from tests on 'flexible' models or full-scale aircraft, it must be admitted that this procedure is somewhat artificial, inasmuch as no model can be completely rigid. Fortunately, as noted in Tables 4 and 5, in many of the investigations for which nominally rigid models have been used, a check has been made on the deformations occurring under load and results have been corrected accordingly. The magnitude of the deformations observed in some of these cases is such that data obtained from tests in which no check was made on model deflections must be regarded as of dubious value.

3.2. The Adequacy of Theoretical Methods

The main purpose of the present investigation was to assess the adequacy of theoretical methods for determining the aerodynamic input data to zero- and low-frequency aeroelastic problems, although some of the investigations using flexible models or full-scale aircraft have provided checks on the accuracy of other aspects of the prediction of aeroelastic effects. In the context of the principal objective it is to be observed, from the relevant columns of Tables 5 and 6 and from the lengthy list of references (42 to 72) called up there that a wide variety of methods of computing aerodynamic loading data for warped lifting surfaces has been employed in the comparison with experimental data. The methods range from pure linearised theory to the empirically modified influence-coefficient (line- or field-matrix) approach pioneered by Zisfein and his associates in America. In some of the reports which provide items for Tables 5 and 6, semi-empirical methods are developed as an integral part of the investigation. In these circumstances it is difficult to draw generalised conclusions about the adequacy of 'theoretical methods'. All that one can comment on with real confidence is the adequacy of a particular method within the range of circumstances for which it has been tested.

The making of an independent appraisal-in-depth of the theory/experiment comparisons in each report would have taken a prohibitively long time. Accordingly a measure of reliance has had to be placed upon the claims of the respective authors as to the degree of agreement achieved in their investigations, and these claims are summarised and commented on, as appropriate, in the last two columns of Tables 5 and 6.

Most authors have felt justified in claiming 'reasonably* good' agreement in their comparisons of aerodynamic loading data, at least for those combinations of Mach number and (small) incidence range that should ensure attached, ummixed flow over the whole of the lifting surface. In particular instances, 'good agreement' has apparently to be interpreted as meaning 'within about 20 per cent', but in general it signifies a rather higher standard than this. There are some reservations and exceptions to the general rule of 'reasonably good' agreement in the attached, unmixed flow régime, particularly at supersonic Mach numbers. Thus, for some supersonic wings, the claim of good agreement between experimental and theoretical pressure distributions is restricted to portions of the wing not affected by the wing trailing edge and tips (*see*, e.g., Item 3, Table 5). Generally, for supersonic wings, agreement on pressure distributions appears less satisfactory for the chordwise distribution than for the spanwise distribution (*see*, e.g., Item 8 of Table 6); moreover the effects of camber are not predicted as accurately as those of twist (*see*, e.g., Item 11 of Table 5). Even when a claim of good agreement is substantiated, either for subsonic or for supersonic flow, the range of positive root incidences for which it is valid seldom extends much beyond 5 degrees; when local incidences at any section of a lifting surface are much in excess of this figure, methods based on linearised theory manifestly fail to predict the measured pressure or load distributions. In several reports attempts are made to relate such failures to anticipated changes in the flow pattern which, in a few instances, have been confirmed experimentally using various flow-visualisation techniques.

As would be expected, empirical methods which utilise such experimental data as sectional life-curve slopes, or the measured pressure distribution over an unwarped wing of the planform under consideration, normally improve the agreement as compared with that achieved when using the 'undoctored' theoretical

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* Fairly, quite, etc. are alternative adjectives used by various authors.

methods from which they are derived. However, as suggested by Gainer[13], such empirical methods need further verification by tests on other wings and at Mach numbers other than those considered when developing the method.

Considered in the context of computer-aided, iterative design techniques which are currently being developed, the preceding two paragraphs suggest that, within the limited range of flight conditions for which unmixed, attached flow conditions prevail, existing lifting-surface methods provide the basis for estimates of aeroelastic distortion effects which are adequate for engineering purposes. Thus, for the earliest stages of design, purely theoretical methods should provide aerodynamic input data of an accuracy commensurate with that of the available structural data, while, as the design develops, it should be possible to improve the accuracy by empirical modifications to these methods based on experimental (model) data.

Apart from the cases of the slender delta configurations of Item 10, Table 6 and some of the configurations of Item 12 of Table 5, there have been few, if any, attempts to check non-linear theories against the experimental data included in the reports of Tables 5 and 6. From the appropriate entry for Item 10 in Table 6, it will be noted that, while one non-linear theory (Küchemann[66]) showed fair agreement with experimentally determined lift characteristics for all three slender delta configurations considered, no analytical method was completely adequate for the prediction of lift and pitching moment characteristics throughout the entire angle-of-attack range.

In view of the above-mentioned inadequacies of existing methods, there is clearly a need to develop new theoretical and/or empirical methods for the prediction of loading characteristics in the important design cases which occur in the régimes of mixed (transonic) and separated flow.

Comparative experimental and theoretical data on structural deformation are very sparse, but such as there are generally suggest that, within the parameter ranges for which the aerodynamic loads can be satisfactorily predicted, the deformations can also be fairly reliably predicted. (*See* Items 3 and 13 of Table 6 for qualifications to this statement.)

4. Conclusions

(1) A thorough search of the literature has revealed a considerable body of comparative theoretical and experimental aerodynamic data (mainly of American origin) which are relevant to the task of determining aerodynamic load distributions for zero- and low-frequency aeroelastic problems in the subsonic, transonic and supersonic flow régimes. The same sources provide a considerably less comprehensive collection of comparative structural deformation data. From an analysis of all these data, which cover a wide range of wing planforms, it is concluded that, in the limited range for which the flow is attached and the flow field is wholly subsonic or wholly supersonic, existing theoretical methods generally provide the basis for estimates of aeroelastic distortion effects which are adequate for engineering purposes.

Accuracy of aerodynamic input data can generally be improved by empirical modification of results obtained by linearised potential theory, on the basis, for example, of experimental data for an unwarped wing of the planform under consideration.

(2) Serious aeroelastic effects can occur in important design cases well outside the restricted flow régime envisaged in (1) above, and the present analysis has shown that, in such circumstances, the application of existing methods may lead to large discrepancies in predicted aerodynamic data. There is thus an urgent need to develop theoretical and/or empirical methods of estimating the effects of wing twist and chamber when the flow is mixed (transonic) or when flow separations occur.

(3) Experimental data contained in the reports reviewed in the present paper (Tables 4 to 6) should provide a useful nucleus of pressure and loading data against which to test the new methods of calculation envisaged in (2) above, but there will be scope for more pressure distribution tests at transonic speeds and at high incidence on wings of current and likely future interest.

(4) It is hoped that the data tabulations presented here will provide designers and their aeroelasticians with a useful quick guide to the loading information available for configurations which approximate to those in which they are interested at any particular time. In this connection it is recommended that, in

the first instance, reference be made to the 'Index' Tables 1 to 3, wherein configurations are arranged in order of ascending aspect ratio. Cross-reference to the appropriate items of Tables 5 and 6 may then be made to ascertain whether the relevant reports are likely to be worthy of detailed study. Finally, reference to Table 4 will indicate whether there are any 'experimental-data-only' reports that could be useful.

TABLE 1

Tapered Swept-Wing Configurations for which Comparative Theoretical and Experimental Data are Available

(A) *Nominally rigid models with built-in warp*

(See Table 5 for full details)

Item No. in Table 5 and (Ref. List No.)	Wing geometry				Details of warp of model(s)	Model configuration	Type of test facility	Mach No. range	Angle-of-attack range
	A.R.	T.R.	Sweep	$\left(\frac{t}{c}\right)_{max}$					
10 ((12) and (15))	2·24	0	70° (L.E.)	3%	1 unwarped; 2 twisted and cambered ($C_{L_{design}}$ = 0·08 and 0·16).	Semi-span wing with reflection plate.	W/T	M = 2·05	$-2° \leqslant \alpha \leqslant 8°$; unwarped wing $-4° \leqslant \alpha \leqslant 6°$, $-6° \leqslant \alpha \leqslant 4°$ warped wings
2 (3)	3·5	0·25	60·8° (c/4)	Not specified	Twisted: 18·5° washout at tip.	—	W/T	Subsonic	—
3 (4)	3·5	0·25	63° (L.E.)	5%	Twisted: 3·5° washout at tip. Cambered: zero at root, 1·1% at tip.	Full-span wing-body combination.	W/T	$1·15 \leqslant M \leqslant 1·70$	Variable with M Max. range (0°, 10°)
7 (9)	3·5	0·20	50° (c/4)	5%	1 unwarped; 2 twisted* (6° tip washout) * linear and quadratic	Half-wing.	W/T	M = 1·60, 2·0	α = 0°, 4°, 8°, 12°
11 (13)	3·5	0·20	50° (c/4)	5%	1 unwarped; 1 uncambered + linear twist; 1 untwisted + 4% camber.	Half-wing.	W/T	M = 1·61, 2·01	α = −10°, 6°, 10°, 12°
13 (16)	3·5	0·20	50° (c/4)	5%	1 unwarped, 1 uncambered + linear twist (6° tip washout) 1 untwisted + 4% camber.	Half-wing.	W/T	M = 1·61, 2·01	$-20° \leqslant \alpha \leqslant 20°$
7 (9)	4·0	0·15	45° (c/4)	6% root 3% from s/2 to tip	1 untwisted; 2 twisted* (6° tip washout); small camber * linear and quadratic	Full-span wing-body (mid-wing).	W/T	M = 0·90, 1·20	α = 0°, 4°, 8°, 12°
8 (10)	4·0	0·60	45° (c/4)	6%	1 unwarped 1 twisted ($4\frac{1}{2}$° tip washout).	Full-span wing-body.	W/T	M = 0·6, 0·95, 1·0	α = 4°
8 and 9 ((10) and (11))	4·0	0·15	45° (c/4)	6% root 3% from s/2 to tip	Cambered, but no built-in twist.	Full-span wing-body.	W/T	M = 0·8, 0·94, 0·98, 1·0, 1·03	α = 4°
6 (8)	4·3	0·326	44° (L.E.)	10%	1 unwarped model. 7 models with various kinds of warp.	Half-wing.	W/T	Low subsonic	$-3° \leqslant \alpha \leqslant 18°$

TABLE 1 (Contd.)

(A) *Nominally rigid models with built-in warp* (Contd.)

Item No. in Table 5 and (Ref. List No.)	Wing geometry				Details of warp of model(s)	Model configuration	Type of test facility	Mach No. range	Angle-of-attack range
	A.R.	T.R.	Sweep	$\left(\frac{t}{c}\right)_{max}$					
1 (2)	6·10	0·327	−12° (c/4)	15% root 12% tip	4% camber, no built-in twist.	Half-wing.	W/T	Low subsonic	$-4° \leqslant \alpha \leqslant 17°$
1 (2)	7·51	0·243	23° (c/4)	18%	4% camber + uniform built-in twist (4° washout at tip).	Half-wing.	W/T	Low subsonic	$-3° \leqslant \alpha \leqslant 21°$
4 (5)	8·02	0·45	45° (c/4)	12%	1 unwarped; 1 twisted and cambered (11° washout at tip; 4% camber).	Full-span wing alone.	W/T	Low subsonic	$-3{\cdot}5° \leqslant \alpha \leqslant 31°$
5 ((6) and (7))	10·0	0·40	40° (c/4)	14% root 11% tip	Cambered and twisted (5° tip washout).	Semi-span wing alone and in combination with half-body.	W/T	$0{\cdot}25 \leqslant M \leqslant 0{\cdot}90$	$-4° \leqslant \alpha \leqslant 20°$

TABLE 1 (Contd.)

(B) *Flexible models and full-scale aircraft*

(See Table 6 for full details)

Item No. in Table 6 and (Ref. List No.)	Wing geometry				Model or aircraft details	Test configuration and type of test	Mach No. and dynamic pressure ranges	Angle-of-attack range
	A.R.	T.R.	Sweep	$\left(\frac{t}{c}\right)_{max}$				
8 (25)	3·1	0·43	23° (L.E.)	4·5%	X-3 aeroplane.	Full-scale flight tests: wing-panel normal force measurements.	$0·7 < M < 1·2$	
8 (25)	3·6	0·60	38° (L.E.)	10% root 12% tip	D-558-II aeroplane.	Full-scale flight tests: wing-panel loading measurements.	$M = 0·8$ $M = 1·0$ $M = 1·8$	$\alpha = 4°, 7°$ $\alpha = 3°, 11°$ $\alpha = 5°, 9°$
8 (25)	3·9	0·30	49° (L.E.)	4%	F-100A aeroplane.	Full-scale flight tests: wing-panel normal force measurements.	$0·65 < M < 1·45$	—
8 (25)	4·0	0·50	8° (L.E.)	4%	X-1E aeroplane.	Full-scale flight tests: wing-panel loading measurements.	$M = 0·8$ $M = 1·0$ $M = 1·9$	$\alpha = 4°, 8°$ $\alpha = 4°, 7°$ $\alpha = 4°, 14°$
2 (19)	4·0	0·60	45° (c/4)	6%	Complete aircraft models, one with steel wing and one with dural wing. Fixed vertical tail and all-movable horizontal tail.	Free-flight rocket-model tests; measurements made in decelerating flight.	$0·92 \leqslant M \leqslant 1·33$ $1100 \leqslant q \leqslant 2500$ (lb/ft^2)	$-2° \leqslant \alpha \leqslant 18°$
7 (24)	4·0	0·15	45° (c/4)	6% root 3% from s/2 to tip	Cambered flexible wing with no built-in twist.	Full-span wing-body combinations tested in W/T.	$0·8 \leqslant M \leqslant 1·43$	$0° \leqslant \alpha \leqslant 20°$, subsonic $0° \leqslant \alpha \leqslant 12°$, supersonic
11 (28)	4·0	0·15	45° (c/4)	6% root 3% from s/2 to tip	Cambered flexible wing with no built-in twist.	Full-span wing-body combination tested in W/T.	$0·8 \leqslant M \leqslant 0·98$	$-4° \leqslant \alpha \leqslant 8°$
1 (18)	4·79	0·513	35° (c/4)	12% root 11% tip	YF-86A aeroplane.	Flight tests of aircraft performing progressively tightening turns, diving turns or pull-ups.	$0·5 \leqslant M \leqslant 1·11$ (at 35 000 ft approximately)	$0° \leqslant \alpha \leqslant 20°$
3 (20)	6·0	0·60	45° (c/4)	9%	Rigid and flexible wing models. (Structurally similar M- and W-wings also tested.)	Semi-span models with reflection plane tested in W/T.	Subsonic M $4·7 \leqslant q \leqslant 46$ (lb/ft^2)	Variable with wing and value of q. Of order $-4° \leqslant \alpha \leqslant 4°$ at highest q.
4 (21)	6·0	0·60	45° (c/4)	9%	Three similar models with various degrees of wing flexibility.	Rocket-powered wing + fuselage models; measurements made in coasting flight.	$0·8 \leqslant M \leqslant 1·3$ $600 < q < 2900$ (lb/ft^2)	Not quoted in report.

TABLE 1 (Contd.)

(B) *Flexible models and full-scale aircraft* (Contd.)

Item No. in Table 6 and (Ref. List No.)	Wing geometry				Model or aircraft details	Test configuration and type of test	Mach No. and dynamic pressure ranges	Angle-of-attack range
	A.R.	T.R.	Sweep	$\left(\frac{t}{c}\right)_{max}$				
9 (26)	8·55	0·398	35° (c/4)		Boeing B-52 aeroplane.	Flight tests of aircraft performing slow-rate 'roller-coaster' manoeuvres.	$0.5 \leqslant M \leqslant 0.82$ $200 < q < 460$ (lb/ft^2) at 20 000 ft $0.7 \leqslant M \leqslant 0.9$ $200 < q < 360$ (lb/ft^2) at 30 000 ft	Not quoted (but corresponds to 'lower lift' range).
5 (22)	9·42	0·42	35° (c/4)	12%	Six-engined jet bomber (B.47A).	Flight tests of aircraft in 'clean' condition, under various loading conditions.	$0.427 \leqslant M \leqslant 0.812$ $127 \leqslant q \leqslant 364$ (lb/ft^2)	$0° < \alpha < 8°$
6 (23)	9·42	0·42	35° (c/4)	12%	B.47A aeroplane.	Full-scale flight tests. Measurements (flight parameters, strain gauges, structural deflections) made during push-pull manoeuvres.	$0.47 \leqslant M \leqslant 0.81$ $145 \leqslant q \leqslant 445$ (lb/ft^2)	Not quoted in report.

TABLE 2

Delta-Wing Configurations for which Comparative Theoretical and Experimental Data are Available

(A) *Nominally rigid models with built-in warp*

(See Table 5 for full details)

Item No. in Table 5 and (Ref. List No.)	Wing geometry A.R.	T.R.	L.E. sweep	$\left(\frac{t}{c}\right)_{max}$	Details of warp of model(s)	Model configuration	Type of test facility	Mach No. range	Angle-of-attack range
13 (17)	1·0	0	76°	2·09%	(i) Unwarped (ii) 3·57% camber (iii) 9·46% camber	Full-span wings.	W/T	Low subsonic	(i) $0° \leqslant \alpha \leqslant 22{\cdot}5°$ (ii) $-20° \leqslant \alpha \leqslant 30°$ (iii) $-10° \leqslant \alpha \leqslant 30°$

(B) *Flexible models and full-scale aircraft*

(See Table 6 for full details)

Item No. in Table 6 and (Ref. List No.)	Wing geometry A.R.	T.R.	L.E. sweep	$\left(\frac{t}{c}\right)_{max}$	Model or aircraft details	Test configuration and type of test	Mach No. and dynamic pressure ranges	Angle-of-attack range
10 (27)	0·424	0	*Series B* 84°		1 rigid model 1 flexible model	Full-span wings, in combination with a single conical-cylindrical body, tested in W/T.	$0{\cdot}7 \leqslant M \leqslant 1{\cdot}1$ $150 \leqslant q \leqslant 250$ (lb/ft^2)	$-4° \leqslant \alpha \leqslant 30°$
	0·848	0	*Series A* 78°		1 rigid model 2 flexible models			
	0·848	0	*Series C* 78°		1 rigid model (area one-half that of Series A models).			
8 (25)	2·2	0	60°	4%	JF-102A aeroplane.	Full-scale flight tests; wing-panel loading measurements.	$M = 0{\cdot}8$ $M = 1{\cdot}0$ $M = 1{\cdot}2$	$\alpha = 5°, 20°$ $\alpha = 5°, 10°$ $\alpha = 3°, 9°$
8 (25)	2·3	0	60°	6·5%	XF-92A aeroplane.	Full-scale flight tests; wing-panel normal force measurements.	$0{\cdot}7 < M < 0{\cdot}85$	
12 (29)	4·0	0	45°	2%	Flexible wing with no built-in warp.	Half-wing tested in W/T.	$1{\cdot}30 \leqslant M \leqslant 4{\cdot}0$ $250 \leqslant q \leqslant 2000$ (lb/ft^2)	$2° \leqslant \alpha \leqslant 10°$

TABLE 3

Trapezoidal- (including Rectangular-) Wing Configurations for which Comparative Theoretical and Experimental Data are Available

(See Table 6 for full details)

Item No. in Table 6 and (Ref. List No.)	Wing geometry				Details of wing models	Test configuration and type of test	Mach No. and total pressure (P_0) ranges	Angle-of-attack range
	A.R.	T.R.	L.E. sweep	$\left(\frac{t}{c}\right)_{max}$				
13 (29)	2·0	1·0	0°	Hexagonal section; $t/c = 4\%$ from $c/4$ to $3c/4$.	Flexible model with no built-in warp.	Half-wing tested in W/T.	M = 2·0 P_0 = 5, 10 psia M = 3·0 P_0 = 10, 30 psia	$\alpha = 2°, 4°$ $\alpha = 3°, 9°$

TABLE 4

Particulars of Reports Giving Experimental Data Only

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel	Details of tests: Test configurations	Details of tests: Flow particulars: Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Comments
1	NACA RM L54 H18 (TIB 4456) (1954) C. V. Williams (31)	4·0 0·6 45° (c/4)	NACA 65A006 1 untwisted wing 1 twisted wing (4·5° washout between wing-body junction and wing-tip).	Langley 8 ft transonic tunnel.	Full-span wings mounted in mid-wing position on a body of revolution.	$0{\cdot}60 \leqslant M \leqslant 1{\cdot}13$ $1{\cdot}7 \times 10^6 \leqslant Re \leqslant 2{\cdot}1 \times 10^6$	Pressure coefficients at 5 spanwise stations on wing and 6 meridian stations around body. $0 \leqslant \alpha \leqslant 20°$	Loading data obtained by graphical-mechanical integration procedures. (Section normal force and $\frac{1}{4}c$-pitching moment coefficients; wing spanwise load distribution; lateral centres of pressure and wing bending characteristics; overall pitching moment characteristics.)	Nominally rigid models. No reference made to any check on deformations occurring under load.
2	NACA RM L58 D23 (TIL 6034) (1958) F. C. Grant (32)	3·5 0·2 50° (c/4)	NACA 65A005 thickness distribution. 5 wings: (i) unwarped (ii)–(iv) twisted wings (linear, quadratic and cubic variations along span; 6° washout at tip) (v) cambered: NACA $a = 0$ mean line, 4% high at every spanwise station.	Langley 4 ft × 4 ft supersonic pressure tunnel.	Semi-span wings mounted on boundary layer by-pass plate.	$M = 1{\cdot}61,\ 2{\cdot}01$ $1{\cdot}7 \times 10^6 \leqslant Re \leqslant 3{\cdot}6 \times 10^6$ $P_0 = 8,\ 15$ psia	Pressure coefficients at 6 semi-span stations for wings (i) and (v) and at 7 semi-span stations for wings (ii)–(iv). $-20° \leqslant \alpha \leqslant 20°$	Some measurements of tip deflection during tests indicated maximum aeroelastic tip twist of about 1·5° (washout) at $\alpha = 10°$ for $P_0 = 15$ psia.	See Item 6 regarding integration of pressure data to obtain section normal-force and pitching-moment data.

TABLE 4 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio, Taper ratio, Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel	Details of tests: Test configurations	Details of tests: Flow particulars: Mach No. (M), Reynolds No. (Re), Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Comments
3	NASA Memos 10-20-58L (TIL 6405) (1958) 12-28-58L (TIL 6182) (1959) 2-24-59L (TIL 6326) (1959) 5-12-59L (TIL 6453) (1959) J. P. Mugler ((33)–(36))	4·0 0·15 45° ($c/4$)	NACA 65A206 $a = 0$ at root (0·1 s) varying linearly to NACA 65A203 $a = 0{\cdot}8$ (modified) at 0·5 s with thickness then remaining constant to tip. 4 wings: (i) untwisted (ii) with linear twist (iii) with quadratic twist (iv) with cubic twist. (6° washout between 0·1 s and tip.)	Langley 8 ft transonic pressure tunnel.	Full-span wings mounted in mid-wing position on a body designed to have minimum wave drag for a given length and volume.	$0{\cdot}80 \leqslant M \leqslant 1{\cdot}20$ at $P_0 = 1{\cdot}0$ atm and $P_0 = 0{\cdot}5$ atm $2{\cdot}6 \times 10^6 \leqslant Re \leqslant 2{\cdot}9 \times 10^6$ for $P_0 = 1{\cdot}0$ $1{\cdot}3 \times 10^6 \leqslant Re \leqslant 1{\cdot}5 \times 10^6$ for $P_0 = 0{\cdot}5$	Pressure coefficients at 6 semi-span stations on wing, and along 5 longitudinal rows on body. *Wing (i)*: $-4° \leqslant \alpha \leqslant 12°$ at all M for $P_0 = 0{\cdot}5$. $-4° \leqslant \alpha \leqslant 4°$ at all M for $P_0 = 1{\cdot}0$ with additional data for $\alpha = 8°$ and 12° at M = 1·2. *Wing (ii)*: $-4° \leqslant \alpha \leqslant 12°$ *Wing (iii)*: $-4° \leqslant \alpha \leqslant 20°$ at all M for $P_0 = 0{\cdot}5$. $-4° \leqslant \alpha \leqslant 20°$ for M = 0·8, 0·9, 0·94 and $-4° \leqslant \alpha \leqslant 12°$ for other Ms, at $P_0 = 1{\cdot}0$. *Wing (iv)*: $-4° \leqslant \alpha \leqslant 20°$ at all M for $P_0 = 0{\cdot}5$. $-4° \leqslant \alpha \leqslant 20°$ for M = 0·8, 0·9, 0·94 and $-4° \leqslant \alpha \leqslant 12°$ for other Ms, at $P_0 = 1{\cdot}0$.	Wing pressure distributions integrated numerically to obtain section normal force coefficients and section pitching moment coefficients about $\frac{1}{4}c$. Flexibility influence coefficients for the wing were obtained from static calibration tests, and used in conjunction with the experimental wing section data to estimate aero-elastic wing-twist angles.	The largest twist angle tabulated was for the initially untwisted wing: $\Delta\alpha = -6{\cdot}3°$ at tip when M = 1·20, $P_0 = 1{\cdot}0$ and $\alpha = 12°$. Thus the aero-elastic twist could be of the same order as the built-in twist. See Item 5 for details of tests on these wings at M = 1·43.

TABLE 4 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel	Test configurations	Flow particulars: Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic parameters measured, with angle-of-attack range	Other parameters measured or deduced	
4	NASA TN D-421 (1960) D. D. Arabian (37)	3·0 0·2 30° (c/4)	NACA 65A004 No built-in twist or camber.	Langley 16 ft transonic tunnel.	Two wings of identical planform, one solid steel and one plastic with steel core, tested at mid-wing position on a body of revolution.	$0{\cdot}80 \leqslant M \leqslant 1{\cdot}03$ $7 \times 10^6 \leqslant Re \leqslant 8 \times 10^6$	Pressure coefficients measured at 6 semi-span stations on wing $-2° \leqslant \alpha \leqslant 26°$ (2° increments).	Structural flexibility influence coefficients were measured in static tests on both wings and used in conjunction with measured aero-dynamic loads to determine aero-elastic twist.	Maximum calculated twist angles at $\alpha = 20°$ and $M = 1{\cdot}0$ were $-0{\cdot}4°$ for steel wing and $-0{\cdot}9°$ for plastic wing.
5	NASA TN D-528 (1960) J. P. Mugler E. R. Woodall (38)	4·0 0·15 45° (c/4)	NACA 65A206, $a = 0$ at root (0·1 s) varying linearly to NACA 65A203, $a = 0{\cdot}8$ (modified) at 0·5 s, with thickness then remaining constant to tip. 4 wings as in Item 3.	Langley 8 ft transonic pressure tunnel.	As for Item 3.	$M = 1{\cdot}43$ (nominal) at $P_0 = 0{\cdot}5$ and 1·0 atm. (Actual M for $P_0 = 0{\cdot}5$ was 1·42.) For $P_0 = 0{\cdot}5$, $Re \approx 1{\cdot}5 \times 10^6$; for $P_0 = 1{\cdot}0$, $Re \approx 2{\cdot}9 \times 10^6$.	Pressure coefficients at 6 semi-span stations on wing, and along 5 longitudinal rows on body. $-4° \leqslant \alpha \leqslant 12°$ for $P_0 = 0{\cdot}5$, and $-4° \leqslant \alpha \leqslant 12°$ for $P_0 = 1{\cdot}0$.	Wing pressure distributions integrated numerically to obtain section normal force coefficients and section pitching moment coefficients about $\frac{1}{4}c$. Flexibility influence coefficients for the wing were obtained from static calibration tests, and used in conjunction with experimental wing section data to estimate aero-elastic wing-twist angles.	Largest twist angle tabulated was for the initially untwisted wing $\Delta\alpha = -3{\cdot}57°$ at tip when $M = 1{\cdot}43$, $P_0 = 0{\cdot}5$ and $\alpha = 20°$. Twist was larger in relation to initial tip incidence in the case $P_0 = 1{\cdot}0$, $\alpha = 4°$, when $\Delta\alpha = -2{\cdot}48°$. See Item 3 for details of tests on these wings in the range $0{\cdot}8 \leqslant M \leqslant 1{\cdot}2$.

TABLE 4 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel	Details of tests: Test configurations	Details of tests: Flow particulars: Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Comments
6	NASA TN D-1244 (1962) E. J. Landrum (39)	3·5 0·2 50° (c/4)	NACA 65A005 thickness distribution. 4 wings: 1 unwarped, 3 twisted (linear, quadratic and cubic variations along span; 6° washout at tip).	Langley 4 ft by 4 ft supersonic pressure tunnel.	Semi-span wings mounted on boundary layer by-pass plate.	M = 1·61, 2·01 $1 \cdot 7 \times 10^6 \leqslant \mathrm{Re} \leqslant 3 \cdot 6 \times 10^6$ P_0 = 8, 15 psia	Pressure distribution data were obtained in the investigation described in Ref. 32 (*see* Item 2 of this table). $-20° \leqslant \alpha \leqslant 20°$.	Section normal-force and pitching-moment coefficients obtained by streamwise integration of the pressure distributions. Measurements of tip deflection during tests indicated maximum aero-elastic tip twist of about 1·5° (washout) at α = 10° for P_0 = 15 psia.	See Item 2 of this table regarding pressure-distribution tests.
7	NASA TN D-1393 (1962) E. J. Landrum (40)	3·5 0·2 50° (c/4)	NACA 65A005 thickness distribution. (i) Cambered and twisted wing; section: a = 0 mean line, modified to have maximum height of 4% c; linear spanwise twist giving 6° washout at tip. (ii) Reflex cambered wing: 1 wavelength sinusoidal mean line with leading-edge angle of attack of −6°.	Langley 4 ft by 4 ft supersonic pressure tunnel.	Semi-span wings mounted on boundary layer by-pass plate.	M = 1·61, 2·01 P_0 = 15 psia giving Re = $3 \cdot 6 \times 10^6$ at M = 1·61 and Re = $3 \cdot 1 \times 10^6$ at M = 2·01	Pressure coefficients measured at 7 semi-span stations. $-20° \leqslant \alpha \leqslant 20°$	Section normal-force and pitching-moment coefficients are tabulated.	These wings have same planform as those of Item 2 (Ref. 32). Pressure coefficients for the unwarped, cambered, and linearly twisted wings of Ref. 32 are tabulated in the present report (Ref. 40). No reference made to any check on aeroelastic deformations; but see Items 2 and 6 which concern similar wings.

TABLE 4 (Concluded)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel	Details of tests: Test configurations	Details of tests: Flow particulars: Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Comments
8	NASA TN D-1394 (1962) E. J. Landrum (41)	(a) 1·342 0·721 unswept T.E. Two semi-span trapezoidal wings, one unwarped and one warped* (maximum angle of local slope relative to chord of unwarped wing is 6°) (b) 1·456 0 70° (L.E.) Three semi-span delta wings, one unwarped, one cambered† and one cambered and twisted‡. For cambered wing and cambered and twisted wing the maximum angles of local slope relative to chord of uncambered wing are −6° and −8° respectively.	3% circular-arc thickness distribution. NACA 65A003 thickness distribution.	Langley 4 ft by 4 ft supersonic pressure tunnel.	Semi-span wings mounted horizontally from turntable in a boundary layer by-pass plate.	M = 1·61, 2·01 Re = $3{\cdot}6 \times 10^6$, $3{\cdot}1 \times 10^6$ P_0 = 15 psia	Pressure coefficients measured at 7 semi-span stations. $-20° \leqslant \alpha \leqslant 20°$	Section normal-force and pitching-moment coefficients are tabulated.	Nominally rigid models. No reference made to any check on deformations occurring under load.

* Equation of mean thickness plane is $\left(\frac{z}{c}\right)_{i,j} = 0{\cdot}033454 \sin\left(\frac{3\pi}{2}\frac{y_i}{b/2}\right) \cos\left[\pi\left(\frac{x}{c_i}\right)_j\right]$ where i, j locate spanwise and chordwise positions respectively,

$0 \leqslant \frac{y_i}{b/2} \leqslant 1;\ 0 \leqslant \left(\frac{x}{c_i}\right)_j \leqslant 1.$

† Equation of mean thickness plane is $\left(\frac{z}{c}\right)_{i,j} = -\frac{2{\cdot}18958 \times 10^{-3}}{c_i}\left[\frac{y_i}{m} + c_i\left(\frac{x}{c_i}\right)_j\right]^2$ where m is cotangent of L.E. sweep.

‡ Equation of mean thickness plane is $\left(\frac{z}{c}\right)_{i,j} = -3{\cdot}35183 \times 10^{-4}\frac{y_i}{c_i}\left[\frac{y_i}{m} + c_i\left(\frac{x}{c_i}\right)_j\right]^2.$

TABLE 5

Particulars of Reports Giving Comparative Experimental and Theoretical Data for Nominally Rigid Models with Built-in Warp

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
1	NACA TN 1351 (1947) R. A. Mendelsohn J. D. Brewer (2)	(a) 6·10 0·327 $-12°$ ($c/4$) (b) 7·51 0·243 23° ($c/4$)	Root: NACA 4415 Tip: NACA 4412 untwisted. NACA 4418 Uniform washout: 4° at tip.	Langley 6 ft by 6 ft stability tunnel.	Semi-span wings alone. (Note: A full-span model of the sweptback wing was previously tested in the 20 ft Langley propellor-research tunnel at Re $\approx$ $1{\cdot}30 \times 10^6$)	$q = 98{\cdot}3$ lb/ft^2 for $\alpha \leqslant 9°$ (corresponds to speed of 196 mile/h at sea level) Re $= 3{\cdot}31 \times 10^6$ for wing (a) Re $= 2{\cdot}10 \times 10^6$ for wing (b). $q = 39{\cdot}7$ lb/ft^2 for $\alpha > 9°$ (124·6 mile/h) Re $= 2{\cdot}10 \times 10^6$ for wing (a) Re $= 1{\cdot}76 \times 10^6$ for wing (b).	Chordwise pressure distributions at 9 stations on semi-span. Overall forces and moments.	Spanwise distributions of basic and additional loading. For wing (a), aeroelastic twist was determined from measured span loading and wing rigidity (determined from static tests). For wing (b) spanwise variation of twist was determined by optical method*	Additional span loadings calculated by lifting line theories. ignoring sweep. (Wings (a) and (b).) Basic and additional span loadings for wing (b) also computed by a lifting surface theory which takes sweep into account (Ref. 43)	'Although the differences between span loadings determined in two wind tunnels were small, they were as great as the differences between span loadings determined from a lifting-line and a lifting-surface theory. The theoretical curves approximated the experimental ones within the accuracy required for engineering calculations.'	* Corrections were applied to angle of attack for model deflections (max. twist corrections near tip, for $q = 98{\cdot}3$ lb/ft^2 and $\alpha = 9°$, were 0·54° for wing (a) and 0·77° for wing (b)).
2	NACA Report 921 (1948) J. de Young C. W. Harper (3)	3·5 0·25 60·8° ($c/4$)	Sections not specified Wing twisted spanwise variation of washout angle $\varepsilon°$: η ε 0·3 10° 0·6 15° 1·0 18·5°	Wind tunnel.	No details given.	Subsonic speed; no details given.	Pressure distribution (4 stations on semi-span).	Spanwise distribution of basic loading ($C_L = 0$).	The simplified lifting-surface method of the referenced report is a development of the 7-point Weissinger method.	Agreement between theory and experiment for basic loading is good. On the basis of this comparison and of force test/theory comparisons for two similar 30°-sweptback wings (untwisted and	Four stations on semi-span are hardly adequate to define the shape of the experimental basic loading curve. Agreement over outboard half of semi-span appears to be only fair.

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	*Details of tests*: Wind tunnel or flight	*Details of tests*: Test configurations	*Details of tests*: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	*Details of tests*: Aerodynamic or flight parameters measured, with angle-of-attack range	*Details of tests*: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory *and experiment* claimed by author	Comments
										with 8·5° twist) it is concluded that 'the subject method can adequately predict the effects of twist and/or camber on the characteristics of wings of arbitrary planform'.	
3	NACA RM A9C16 (TIB 2205) (1949) V. I. Stevens J. W. Boyd (4)	3·5 0·25 63° (L.E.)	NACA 64A005 profile on $a = 1$ camber line. Built-in camber, varying from zero at root to about 1·1% at tip. Built-in twist giving 3·5° washout at tip.	Ames 6 ft by 6 ft supersonic wind tunnel.	Full-span wing-body combination.	$1{\cdot}15 \leqslant M \leqslant 1{\cdot}70$ $4{\cdot}6 \times 10^6 \geqslant Re \geqslant 4{\cdot}0 \times 10^6$	Pressure distributions α-ranges: (0°, 2°) for M = 1·15 (0°, 8°) for M = 1·3 (0°, 10°) for M = 1·4, 1·5, 1·6 (0°, 4°) for M = 1·7	Chordwise distributions of pressure coefficient. Measurements of wing-root and tip deflections under load (significant distortion-twist was indicated).	Rigid wing loading calculated by method of Ref. 46 (Cohen). Correction for the effect of aeroelastic distortion made using method of Ref. 49 (Frick and Chubb) and the measured twist. Effect of fuselage shown to be negligible.	'... over the portions of the wing not affected by the wing trailing edge and tip the agreement ... is generally good, the best agreement existing near zero lift. Over the regions influenced by the wing tip and trailing edge the effects of viscosity apparently are responsible for the poorer agreement between theory and experiment.'	In the discussion of results, theory/experiment comparisons are based on predicted loadings for the elastic wing which differ slightly from those for the rigid wing. Allowance for the aeroelastic distortion does not seem invariably to have improved agreement.

TABLE 5 (Contd.)

<table>
<tr><th rowspan="2">Item No.</th><th rowspan="2">Report No.
Date
Authors
(Ref. List No.)</th><th colspan="2">Lifting surface particulars</th><th colspan="5">Details of tests</th><th rowspan="2">Theoretical methods employed</th><th rowspan="2">Degree of agreement between theory and experiment claimed by author</th><th rowspan="2">Comments</th></tr>
<tr><th>Planform:
Aspect ratio
Taper ratio
Sweep</th><th>Basic section(s) camber and twist of unloaded surface</th><th>Wind tunnel or flight</th><th>Test configurations</th><th>Flow particulars
Mach No. (M)
Reynolds No. (Re)
Dynamic pressure (q)
or
Total pressure (P_0)</th><th>Aerodynamic or flight parameters measured, with angle-of-attack range</th><th>Other parameters measured or deduced</th></tr>
<tr><td>4</td><td>NACA RM L52J03a
(TIB 3541)
(1952)

G. L. Pratt
(5)</td><td>8·02
0·45
45° (c/4)</td><td>Wing A:
NACA 63_1A012 (uncambered) no built-in twist.

Wing B:
Sections of same thickness distribution as A but incorporating twist ($\approx 11°$ washout at tip) and camber (approx. 4%).</td><td>Langley 19 ft pressure wind tunnel.</td><td>Full-span wing alone.</td><td>Force tests:
Re × 10^{-6} M
1·5   0·07
2·2   0·11
3·0   0·14
4·0   0·19
4·8   0·25

Pressure distribution data obtained at Re = $1·5 \times 10^6$ and $4·0 \times 10^6$</td><td>Lift, drag and pitching-moment data obtained from force tests and pressure distributions for $-3·5° \leqslant \alpha \leqslant 31°$</td><td>Spanwise distributions of lift, pitching moment and drag loading parameters.</td><td>For span loading methods of Weissinger (7 × 1 point solution) and Multhopp (15 × 1, 15 × 2 and 23 × 1 point solutions).</td><td>(i) Span loading: on unwarped wing—'excellent agreement between experiment and calculations by the Multhopp solutions' which gave practically identical loadings at $C_L = 0·31$.'

For warped wing 'The loadings calculated by the Multhopp solutions having 15 or 23 spanwise control points are in good agreement with the experimental results where no separation exists on the wing.'

(ii) Wing coefficients ($dC_L/d\alpha$, α_0, dC_m/dC_L, C_{m_0}): Multhopp 15 × 2 solution predicts these much better than solutions having one chordwise control point.</td><td>No corrections applied for aeroelastic twist of models.</td></tr>
</table>

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
										Multhopp 15 × 2 gives 'very good agreement' for $dC_L/d\alpha$, α_0 and C_{m_0}, and 'good agreement' for section centres of pressure.	
5	NACA RM A52 F18 (TIB 3318) (1952) G. G. Edwards B. E. Tinling A. C. Ackerman (6) and NACA RM A52 K20 (TIB 3663) (1953) F. W. Boltz H. H. Shibata (7)	10 0·4 40° (Reference sweep line joining $c/4$ points of sections inclined 40° to plane of symmetry.)	NACA 00XX thickness distributions (sections perpendicular to reference sweep line). $a = 0{\cdot}8$ modified mean line, ideal lift coefficient = 0·4 $t/c = 14\%$ at root; 11% at tip. Tip washout = 5°.	Ames 12 ft pressure wind tunnel.	Semi-span wing, alone, and in combination with fuselage (half of a body of revolution of fineness ratio 12·6). High wing.	*Wing fuselage* $0{\cdot}25 \leqslant M \leqslant 0{\cdot}9$ $Re = 2 \times 10^6$ and $M = 0{\cdot}25$ $Re = 8 \times 10^6$ *Wing alone* $M = 0{\cdot}165$ $Re = 8 \times 10^6$	Overall force measurements (Ref. 6). Chordwise pressure distributions at 9 semi-span stations (Ref. 7) α-range dependent on values of M and Re. Overall range of tests: $-4° \leqslant \alpha \leqslant 20°$ (Note: Tests were made without fences and with 3 or 4 fences.)	Section normal force and pitching-moment characteristics at 9 stations. Spanwise distributions of loading coefficient $c_n\,(c/c_{av})$ for various α and of additional loading coefficient $dc_1/dC_L\,(c/c_{av})$.	Span loading calculated by (1) modified versions of Falkner 19 × 1 method, with and without allowance for fuselage effect. (2) Weissinger 7 × 1 method as adapted by De Young and Harper (Ref. 3).	'The modified Falkner 19 × 1 method was found to predict the spanwise load distribution to a good degree of accuracy provided little flow separation existed on the wing.'* '... the (additional) loading calculated by the Weissinger 7 × 1 method ... is shown to be too high over the outer portions of the wing span.'†	* This statement is apparently applicable to distributions of both $c_n\,(c/c_{av})$ and $dc_1/dC_L(c/c_{av})$. † Error of order 7%. Static load tests showed that aero-elastic washout at tip could be as much as 2·2° per unit C_L, but experimental data were not corrected for this effect.

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
6	ARC R. & M. 2938 (1953) J. Black (8)	4·3 0·326 44° (L.E.)	$(t/c)_{max} = 10\%$ at 0·4 c. Surface unwarped in the undeformed condition.	Open-jet wind tunnel	Semi-span Perspex model, undeformed, and with 7 deformation modes 'frozen' in.	Low subsonic M; Re = 0.575×10^6	Chordwise pressure distribution at 8 spanwise stations and for a range of root-chord incidences $-3° \leqslant \alpha \leqslant 18°$.	Spanwise distribution of sectional normal force coefficient c_n; overall normal force and pitching-moment coefficients $\bar{C}_n$ and $\bar{C}_{m\bar{c}/4}$. Spanwise loadings at zero lift ($\bar{C}_n = 0$) and at sub-stalling incidences ($\bar{C}_n = 0.3$).	(1) Weissinger 4 spanwise stations (as applied by Stevens in Ref. 47) (2) Diederich (Ref. 48).	*Span loading:* *Weissinger:* agreement reasonable for modes exhibiting smooth variation of washout across semi-span; poor for modes with abrupt changes. *Diedrich:* Generally good as regards shape; absolute accuracy dependent on accuracy of assumed $dC_L/d\alpha$. *Zero-lift pitching moment* $\bar{C}_{m_0}$ predicted 'quite accurately' by Weissinger method. *Root-incidence for no lift* Predictions by Weissinger and Diedrich 'reasonably good' for most modes.	As suggested by Black, shapes of loading distributions predicted by Weissinger would probably be improved by increasing number of spanwise locations.

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
7	NACA RM L57 D24a (TIL 5595) (1957) F. C. Grant J. P. Mugler (9)	(a) 4·0 0·15 45° (c/4) (b) 3·5 0·2 50° (c/4)	NACA 65A206 at root NACA 65A203 from 0·5 s to tip. Small built-in camber: (i) untwisted; (ii) linearly twisted; (iii) quadratically twisted (6° washout at tip). NACA 65A005 uncambered: (i) untwisted; (ii) linearly twisted; (iii) quadratically twisted (6° washout at tip).	Wind tunnel tests at Langley Aero- nautical Laboratory.	(a) Full-span wing-body combination. (b) Half-wing	M = 0·90, 1·20 M = 1·6, 2·0	Chordwise lifting-pressure distributions and spanwise load distribu- tions. $0 \leqslant \alpha \leqslant 20°$ (Data presented for α = 0°, 4°, 8°, 12° only.)	Incremental lifting pressures and span loadings due to twist.	For M = 0·9: Lifting surface theory with provision for presence of body (as used by Crigler— see Ref. 10). For M = 1·2 and 1·6: Linearised supersonic theory for subsonic leading edges, Refs. 54, 63 (Heaslet and Lomax). No allowance for body in configuration (a). For M = 2·0: Method for supersonic leading and trailing edges due to Kainer (Ref. 61)	For incremental span loadings: '... in general rather good agreement—for zero angle of attack. The measured loadings due to twist generally diminished with increasing angle of attack through the M-range. At M = 0·9 incre- mental loadings progressively vanished from the tip inboard with increasing α. ... at about 20° there was no difference in loadings of the flat and twisted wings.' At the higher supersonic speeds similar behaviour was starting near α = 20° but for α < 20°, no important change in shape of incremental loadings occurred, although the strength of the loading diminished with increasing α.	It is noted that for the transonic wings (a), at positive angle of attack, incremental aeroelastic twists occurred (about 10% of the 6° of built-in twist at α = 12°).

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
8	NACA TN 3941 (1957) J. L. Crigler (10)	(a) 4·0 0·6 45° (c/4)	NACA 65A006 (i) Untwisted; (ii) twisted (approx. 4½° washout from root to tip).	Langley 8 ft transonic tunnel.	Full-span wing-body.	M = 0·6, 0·95, 1·0	Spanwise load distributions for α = 4° (Wings (a) (i) and (ii).)	Chordwise centre-of-pressure locations for several spanwise sections for wings (a) and (b)	Method is described as 'similar to that of Falkner (Ref. 44) for treating wings in incompressible flow and that of Runyan and Woolston (Ref. 70) for treating oscillating finite wings in subsonic compressible flow.' For wing-body combination use is made of Ref. 64 (Zlotnick and Robinson).	'. . . the magnitude and the distribution of spanwise loading calculated . . . are in good agreement with experiment up to M = 0·95, and for the highly tapered wing (wing (b)) the agreement is still good up to M = 0·98.' For local chordwise cp positions on wings (a) and (b), agreement between theory and experiment is good over inboard section up to M = 1·0. At outboard stations experimental data show sharp rearward shift between M = 0·95 and M = 1·0, not indicated by calculations.	It is stated that for test wings, deflections under load were very small at α = 4°, so that any discrepancies introduced in the calculations due to deflection or bending were small. (But see Item 9.) Experiment/theory comparisons for overall longitudinal cp positions indicate trends consistent with the comparisons for local cp behaviour.
		(b) 4·0 0·15 45° (c/4)	NACA 64A206 a = 0 at mid-span, fairing into NACA 64A203, a = 0·8 (modified) at 0·5 semi-span; then constant section to tip.	Langley 8 ft transonic tunnel.	Full-span wing-body.	M = 0·8, 0·94, 0·98, 1·0	Spanwise load distributions for α = 4°. (Wing (b).)	Longitudinal centre-of-pressure positions for wings (a) at C_L = 0·2, 0·4 and for wing (b) at C_L = 0·4.			

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
9	NASA TN D-96 (1959) J. L. Crigler (11)	4·0 0·15 45° ($c/4$) (Same planform as Wing B of Item 8.)	NACA 64A206, $a = 0$ at mid-span, fairing into NACA 64A203, $a = 0{\cdot}8$ (modified) at 0·5 semi-span; then constant section to tip.	Langley 8 ft transonic tunnel.	Full-span wing-body.	M = 0·98, 1·03	Spanwise load distributions for $\alpha = 4°$		A lifting-surface procedure similar to that used by Crigler in Item 8 but with chordwise integrations performed analytically rather than numerically. Wing twist due to aeroelastic effects was evaluated at M = 0·98 and 1·03, using measured loads and influence coefficients obtained by static-deflection calibrations of wing. Calculated loadings made on assumption that wing at M = 1·0 was pretwisted by average of values for M = 0·98 and 1·03†.	In the Concluding Remarks the author claims that 'the magnitude and distribution of the calculated spanwise loading are in good agreement with experiment at M = 1·0'. More detailed statement from main text: 'the calculated shape of the loading at M = 1·0 is about the same as the experimental shapes for M = 0·98 and 1·03. The magnitude of the lift for the calculated data for M = 1·0 is about 10% higher than the experimental data at M = 0·98 which is greater than the value for M = 1·03*'.	*No experimental data were obtained at M = 1·0. Results calculated by method of Ref. 11 are relatively insensitive to choice of control points, whereas results obtained by Crigler's other method (Item 8) are shown to vary enormously with choice of control points. †Calculated washout at tip for M ≈ 1·0, $\alpha = 0°$, is about $3\frac{1}{2}°$. The wing of the same planform in Item 8 was assumed to suffer negligible deformation. It is not clear whether wings were the same structurally.

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
10	NASA TM X-332 (1960) and NASA TN D-1264 (1962) H. W. Carlson (12) and (15)	2·24 0 70° (L.E.)	3 wings with thickness distribution corresponding to 3% circular arc sections: (i) unwarped; (ii) twisted and cambered for $C_{L_{design}} = 0{\cdot}08$, (iii) twisted and cambered for $C_{L_{design}} = 0{\cdot}16$	Langley 4 ft by 4 ft supersonic pressure tunnel.	Semi-span wings attached to reflection plate; sting-mounted.	$M = 2{\cdot}05$ $Re = 4{\cdot}4 \times 10^6$	Overall force measurements (Ref. 12). Chordwise pressure distributions at 5 semi-span stations (Ref. 15). $-20 \leqslant \alpha \leqslant 8°$, Wing (i) $-4° \leqslant \alpha \leqslant 6°$, Wing (ii) $-6° \leqslant \alpha \leqslant 4°$, Wing (iii)	Spanwise distributions of axial force and normal force. Overall forces obtained from integration of pressure data.	Pressure distribution estimated by linearised theory; formula quoted includes terms giving contributions due to thickness, camber and 'flat-plate' incidence.	'Measured pressure distributions for all wings agreed fairly well with the linearised theory estimates.'* 'Agreement between integrated pressure data and force data is reasonably good except for axial-force coefficients for the wings with twist and camber.'†	*Carlson suggests discrepancies may be due to aeroelastic deflections (not measured, apparently) and the presence of a vortex flow originating at wing apex. †Discrepancies may be due to fairing of pressure data, which becomes more critical with increased twist and camber.
11	NASA TN D-801 (1961) P. A. Gainer (13)	3·5 0·2 54° (L.E.)	NACA 65A005 thickness distribution: (i) untwisted and uncambered (ii) built-in linear twist (6° washout from root to tip); uncambered; (iii) untwisted; cambered—NACA $a = 0$ mean camber line, 4% high at each spanwise station.	Wind tunnel		$M = 1{\cdot}61, 2{\cdot}01$	Pressure distributions $\alpha_{root} = -10°$ 6°, 12° at $M = 1{\cdot}61$ $\alpha_{root} = -10°$ 6°, 10° at $M = 2{\cdot}01$	Perturbation velocity potential $\phi(x, y)$ calculated from the measured distribution of lifting-pressure coefficient by linearised potential-flow theory.	Method developed in reference report combines linearised theory with empirical adjustments, based on experimental pressure-distribution data measured on a flat wing of the desired planform; method leads to semi-empirical influence coefficients and	'The end result (of the method) is a single matrix equation which relates aerodynamic load distribution to angle-of-attack distribution over a wide range of angles of attack with good accuracy.' This statement from the Introduction is qualified in the Discussion—'The greatest discrepancies at	As author suggests, errors in cambered wing loadings would probably be reduced by replacing each leading-edge influence panel by a number of smaller panels which would more closely approximate the large curvature of the cambered wing. One endorses his

TABLE 5 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
									includes an approximate representation of non-linear effects exhibited by experimental data.	positive angles of attack were obtained in predicting the cambered wing loadings, expecially at M = 2·01.'	view that any empirical method such as this should be further verified by tests on other wings and at other Mach numbers.
12	ASD TR 61-645 pp. 369–398 (1961) E. E. Covert (14)	Table 1 of this paper gives details of 12 wing or wing-body configurations incorporating distorted (cambered and/or twisted) surfaces which had been, or were due to be, tested by end of 1961. Experiment-theory comparisons for a few other configurations are given.		Wind-tunnel	Various.	Supersonic (Various M, Re, P_0).	Not detailed.		Linear theory; second order theories (e.g. Beane—Ref. 72) shock-expansion theory; Zisfein-type empirical approach.	For computing pressures on an aerodynamic surface subject to a smooth distortion: 'linear theory is valid only for very slight distortions.* The empirical procedure is generally accurate to within 5% although it is hard to determine the limits of application. Second-order procedures are generally more accurate and appear valid over a wide range of Mach numbers.'	*The criterion $M\theta < 0{\cdot}1$ for 5% accuracy, is suggested. (M, θ are local Mach number and slope respectively).
13	NASA TN D-929 (1961) E. J. Landrum K. R. Czarnecki (16)	3·5 0·2 50° (c/4)	NACA 65A005 thickness distribution 3 wings: (i) unwarped; (ii) cambered NACA $a = 0$	Langley 4 ft by 4 ft supersonic pressure tunnel.	Semi-span wings mounted on boundary layer by-pass plate.	M = 1·61, 2·01 $1{\cdot}7 \times 10^6 \leqslant \mathrm{Re} \leqslant 3{\cdot}6 \times 10^6$ P_0 = 8, 15 psia	Forces and moments on wings measured by 4-component strain-gauge balance	Tip deflections measured. Max. tip twist ≈ 1·5° (washout) for α = 10° at P_0 = 15 psia*.	(1) Linear theory used to calculate lift-curve slopes for unwarped wing. (2) Theoretical	'Comparison of experimental data with theory for the flat and twisted wings shows for the subsonic leading	*Angles of attack have not been corrected to allow for aeroelastic effects.

TABLE 5 (Concluded)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio, Taper ratio, Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
			mean line 4% high at every spanwise station; (iii) linearly twisted (6° washout at tip).				$-20° \leqslant \alpha \leqslant 20°$ (overall range)		span loadings, from Refs. 63 and 54 for M = 1·61 and from Ref. 61 for M = 2·01 used to determine lift increment due to wing twist. (1) and (2) combined to predict lift for twisted wing.	edge (M = 1·61) that the experimental lift-curve slope at $\alpha = 0°$ is higher than the theoretical curve slope†. For the supersonic leading-edge (M = 2·01), the experimental lift-curve slope is about the same or slightly lower than the theoretical.'	†By about 6%–8%.
14	A.R.C. 31588 (1969) R. K. Nangia G. J. Hancock (17)	1·0 0 76° (L.E.) Original uncambered wing (A) was subsequently moulded into a 3·57% cambered wing (B) and then into a 9·46% cambered wing (C).	$t/c_{root} = 0{\cdot}0209$ symmetric L.E. shape featuring bevelled edges at right angles.	Blow-down wind tunnel (39 in. × 30 in.)	Sting-mounted full-span wing	Airspeed 90 ft/sec	Pressure distribution (13 points spanwise at 16 chordwise stations on each surface). (A) $0° \leqslant \alpha \leqslant 22{\cdot}5°$ (B) $-20° \leqslant \alpha \leqslant 30°$ (C) $-10° \leqslant \alpha \leqslant 30°$	'Day-glo' technique employed for visualisation of surface stream-line flow. Longitudinal lift ditributions C_{L_L}/C_L for various values of overall C_L.	For estimation of longitudinal lift distribution C. R. Taylor's lifting-surface theory (Ref. 69).	'... the linear $C_L \sim \alpha$ curve, according to Taylor's theory, is far below the experimental non-linear curves' but 'Comparison of the experimental streamwise lift distribution with Taylor's linear theory seems to be extremely good'.	The authors relate these results to recent ideas of Polhamus, according to which C_L for delta wings with L.E. separation can be estimated by assuming that, in addition to the linear lift there is incremental nonlinear lift due to rotation of the linearised L.E. thrust through 90° as a result of the L.E. separation.

TABLE 6

Particulars of Reports Giving Comparative Experimental and Theoretical Data for Flexible Models and Full-scale Aircraft

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack	Other parameters measured or deduced			
1	NACA RM A52 A31 (TIB 4635) (1952) L. S. Rolls F. H. Matteson (18)	4·79 0·513 35° (c/4)	NACA sections (modified) normal to $\frac{1}{4}c$-line: Root: 0012-64 Tip: 0011-64 Washout at tip: 2° (streamwise direction).	Flight tests of YF-86A aeroplane. At lower Mach numbers data were obtained in progressively tightening turns at constant speed. At higher M data were obtained in diving turns or pull-ups with, inevitably, some variations in M. Nominal altitude: 35 000 ft.		$0{\cdot}5 \leqslant M \leqslant 1{\cdot}11$ corresponding to $9{\cdot}7 \times 10^6 \leqslant Re \leqslant 21{\cdot}6 \times 10^6$ at 35 000 ft.	Chordwise pressure distributions at 5 spanwise stations on half-wing. $0 \leqslant \alpha \leqslant 20°$ (uncorrected measurements). Normal accelerations recorded.	Span load distributions. Wing-panel normal force coefficients and lateral centres of pressure. Wing bending measured in flight and deflections found to be small. Effects of bending and torsion on load distributions found to be negligible.	Additional spanwise loading computed by De Young and Harper version[3] of Weissinger's method for M < 1·0, and by D. Cohen's linearised method (Ref. 53) for M > 1·0. The latter also provides the theoretical chordwise loading.	'At subsonic speeds the spanwise distribution of loading was adequately predicted by the Weissinger method up to the buffet boundary. At supersonic speeds the centre of loading was inboard from that predicted from either supersonic or subsonic theory .... For normal-force coefficients above the buffet boundary the measured load distribution departed from the theoretical, the amount depending upon the Mach number.'	Comparative data for basic spanwise loading are not presented.

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
2	NACA RM L52 L30 (TIB 3606) (1953) A. J. Vitale (19)	4·0 0·6 45° ($c/4$) Two wings, one in steel and one in duralumin, of identical geometry, were tested.	NACA 65A006 (streamwise) Unwarped.	Free-flight rocket-model tests made at Langley Pilotless Aircraft Research Station. Measurements made in decelerating flight after separation from booster.	Complete aircraft model with fixed vertical tail and all-movable horizontal tail used for longitudinal control.	*Steel wing:* $0{\cdot}97 \leqslant M \leqslant 1{\cdot}27$ $4{\cdot}8 \times 10^6 \leqslant Re \leqslant 7 \times 10^6$ *Dural wing:* $0{\cdot}92 \leqslant M \leqslant 1{\cdot}33$ $4{\cdot}8 \times 10^6 \leqslant Re \leqslant 7{\cdot}4 \times 10^6$ $1100 \leqslant q \leqslant 2500$ lb/ft^2	Time-histories of model short-period oscillations in angle of attack, normal acceleration, longitudinal acceleration, and wing normal force α-range variable with model and M Overall range: $-2° \leqslant \alpha \leqslant 18°$	Mach numbers and dynamic pressures calculated from telemetered total pressure and freestream static pressure obtained from combination of radiosonde and tracking radar data. Structural influence coefficients were obtained for dural wing prior to flight.	For estimation of aeroelastic effects on lift: Modified-strip-theory method given in Appendix of Reference Report, together with methods of De Young and Harper (Ref. 3) and Diederich and Foss (Ref. 57)	'The loss in lift-curve slope due to aeroelastic distortion found experimentally agrees very* well with that predicted by a modified-strip-theory method of calculation.'	*Quotation from Conclusions of Report. According to the Summary, the data agreed 'fairly well'.
3	NACA RM L53 JO2a (TIB 4046) (1953) J. W. McKee D. R. Croom R. L. Naeseth (20)	6·0 0·6 ±45° ($c/4$) Structurally similar (flexible) sweptback, M- and W-wings and also a nominally rigid swept-back wing were tested.	NACA 65A009 surfaces unwarped when unloaded.	Langley 300 mile/h 7 × 10 ft. tunnel.	Semi-span models with reflection plate.	$0{\cdot}4 \times 10^6 \leqslant Re \leqslant 1{\cdot}25 \times 10^6$ $4{\cdot}7 \leqslant q \leqslant 46$ lb/ft^2	Lift, drag and pitching moment. α-range varied according to wing and value of q. It was of order $-4° < \alpha < 4°$ at highest value of q.	Bending and torsional deflections of wings $dC_L/d\alpha$, dC_m/dC_L y_L (lateral centre of lift).	Wing twists and aeroelastic effects on aerodynamic characteristics calculated by theory of Ref. 51 using rigid-wing additional loading distributions calculated by Ref. 50 and basic loading distributions calculated by Ref. 3 (swept-back wing) and Ref. 50 for M- and W-wing.	'Some large discrepancies are shown between the experimentally and theoretically determined twist angles and aerodynamic characteristics but, in general, there is fair agreement in their variation with dynamic pressure.'	The method of Ref. 51 is based on a relaxation approach, utilising aerodynamic loadings based on Weissinger's simplified lifting-surface theory, together with simple beam theory.

TABLE 6 (Contd.)

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4	NACA RM L54 B16 (TIB 4189) (1954) R. E. Walters (21)	6·0 0·6 45° (c/4) Three similarly constructed models of varying degrees of wing flexibility were tested.	NACA 65A009 surfaces unwarped when unloaded.	Rocket powered models flown at Langley Pilotless Aircraft Research Station. Disturbed in pitch by pulse rockets during coasting flight.	Wings mounted on fuselage (curved body of revolution of fineness ratio 10).	$0{\cdot}8 \leqslant M \leqslant 1{\cdot}3$ $3 \times 10^6 \leqslant Re \leqslant 8 \times 10^6$ $600 < q < 2900$ lb/ft^2	4-channel telemeters provided measurements of normal and longitudinal acceleration, total pressure and angle of attack. Trajectory and flight velocity measured by radar. α-range not quoted.	Variation of $dC_L/d\alpha$ and damping-moment coefficient $(C_{m_{\dot{\alpha}}} + C_{m_q})$ with M.	Simple influence-coefficient method, developed in the reference report, using matrix notation and having provision for incorporating various assumptions as to spanwise load distribution (e.g. strip theory, simplified subsonic lifting surface theory) and centre of pressure axis position.	'Values of effective lift slope ratio as predicted by an assumed strip-theory load distribution, coupled with experimentally determined structural influence coefficients show good agreement with experimental results.' Effects of changing spanwise load distribution or cp axis position were small.	Walters believes his results show that the approximate approaches are sufficient to predict flexible-wing lift-slope of wings having A.R. $\geqslant$ 6. Ref. 19 (Item 2) is quoted as appearing to extend this conclusion down to 45° swept wings of A.R. = 4.

TABLE 6 (Contd.)

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5	NACA RM L56 E21a (TIL 5551) (1956) W. S. Aiken R. A. Fisher (22)	9·42 0·42 35° (c/4)	BAC 145 $t/c = 12\%$	Flight tests of 6-engine swept-wing jet bomber (B-47A).	Aircraft in clean condition; forward and normal CG positions, and range of weights.	$0{\cdot}427 \leqslant M \leqslant 0{\cdot}812$ $127 \leqslant q \leqslant 364$ lb/ft^2	Normal accelerations; pitching velocity and acceleration; angle of attack; dynamic and static pressures, in abrupt push-pull manoeuvres. $0° \leqslant \alpha \leqslant 8°$	Corrected values of lift-curve slope and angles of zero lift for flexible aircraft; and corresponding values for tail-off rigid-wing aircraft, for comparison with wind tunnel data.		No ab initio theoretical predictions of lift slope, etc., are given. For the conversion of flexible aircraft values to rigid aircraft values, the superposition method of Brown, Holtby and Martin (Ref. 52) was used.	Excellent agreement between tail-off rigid-wing values of $dC_L/d\alpha$ deduced from flight tests, and rigid wind tunnel data, up to M = 0·7. For 0·7 < M < 0·81 flight-deduced values increased more rapidly with M than wind tunnel values. '. . . in the M-range tested, standard design calculation methods would accurately predict flexible lift-curve slopes if the basic two-dimensional lift-curve-slope data and wing-stiffness data are accurate.'	This method for obtaining rigid lift-curve slopes from flexible flight-test values is essentially the reverse of standard procedures used in design for estimating effects of flexibility on aircraft lift-curve slope.

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
6	NACA RM L57 E28 (TIL 5616) (1957) P. A. Gainer P. W. Harper (23)	9·42 0·42 35° ($\frac{1}{4}c$)	BAC 145 (t/c = 12%) no twist.	Flight tests of B-47A swept-wing medium bomber. Measurements made during gradual push-pull or roller-coaster manoeuvres at various Mach numbers and altitudes. Pertinent quantities recorded continuously during each test run.	Standard configuration with minor external modifications to house instrumentation, which were assumed to cause no appreciable load-distribution changes. Gross weight: 105 500–114 500 lb. Altitude: 15 000–30 000 ft.	$0{\cdot}47 \leqslant M_{av} \leqslant 0{\cdot}81$ $145 \leqslant q_{av} \leqslant 445$ (lb/sq ft) at 20 000 ft	Airspeed, altitude, fuselage angle-of-attack, rotational velocities and accelerations, linear accelerations and control-surface displacements. Normal load distribution along wing-span. Normal and transverse load factors at nacelles. Strain-gauge measurements at various spanwise stations on wings, nacelle struts and horizontal-tail root.	Structural shear, bending moment and torque deduced from strain-gauge readings. Inertia components obtained from weight distribution and measured load-factor distribution. Aerodynamic shear, B.M., and torque then determined by addition of the above. Wing deflections calculated from measured loading and experimentally determined influence-coefficient matrix. Compared with deflections measured in flight.	The method of calculating additional and basic wing-load distributions is described in Appendix B of the reference report. It is based on the matrix method of Gray and Schenk (Ref. 60) used in conjunction with the super-position method of Brown, Holtby and Martin (Ref. 52) and makes use of the results of low-speed wind-tunnel pressure-distribution tests for the B-47 wing.	'(1) Additional-load quantities, including centres of pressure, can be adequately predicted. (2) Basic shears, bending moments, and torques can be adequately predicted especially near the root. (3) A comparison of deflections calculated from measured loads with deflections measured in flight shows good correlation between the two different types of measurements. Both sets of experimental deflections agree well with theoretical deflections.'	To minimise effects of pitching acceleration manoeuvres were performed 'gradually' rather than 'abruptly' as in the companion investigation of Ref. 22 (Item 5). This resulted in some changes in M and q during each run. α-range not quoted but probably of same order ($0 < \alpha < 8°$) as for Ref. 22.

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
7	NACA RM L57 D29a (TIL 5583) (1957) T. L. Fischetti (24)	4·0 0·15 45° (c/4)	NACA 65A206 a = 0 at root NACA 65A203, a = 0·8 (modified) at and outboard of 0·5 s. Untwisted.	Langley 8 ft transonic tunnels.	Full-span wing-body (basic Sears-Haack body and a body indented symmetrically for M = 1·2).	0·8 $\leqslant$ M $\leqslant$ 1·43	Chordwise pressure distributions at 6 semi-span stations on wing. Pressure distributions along body (5 stations spaced at 45° intervals round circumference). 0° $\leqslant \alpha \leqslant$ 20° at subsonic speeds 0° $\leqslant \alpha \leqslant$ 12° at transonic and supersonic speeds.	Flexibility influence coefficients obtained by static deflection calibrations of the wing and used to estimate aeroelastic wing twist. (Max. twist $\approx$ −6·7° for α = 12° at M = 1·125). Various loading characteristics deduced.	Crigler's method, (Ref. 10) based on linearised subsonic theory, was used to calculate spanwise load distributions on the cambered wing, with and without allowance for the aeroelastic twist, deduced from the experimental loading in conjunction with the measured flexibility influence coefficients.	'Calculations of theoretical span loadings at low angles of attack' (α = 0°, 4°), 'considering wing flexibility, were in excellent agreement with experimental loadings at low subsonic Mach numbers' (M = 0·8, 0·9), 'but failed to account for an outboard shift in loading at high subsonic Mach numbers.' 'For an angle of attack of 8° and M = 0·9, separation has occurred over the outboard wing sections and the agreement . . . is poor.'	This investigation was not (apparently) conceived as an aeroelastic investigation; thus the observed occurrence of appreciable aeroelastic twist emphasises the importance, when testing nominally rigid models, of checking that aeroelastic deformations are in fact negligible (or, alternatively of allowing for them).

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
8	NACA RM H57 E01 (TIL 6615) (1957) F. S. Malvestuto T. V. Cooney E. R. Keener (25)	(a) 4·0 0·5 (app.) 8° (L.E.) (b) 3·1 0·43 (app.) 23° (L.E.) (c) 3·6 0·6 (app.) 38° (L.E.) (d) 3·9 0·3 (app.) 49° (L.E.) (e) 2·2 (Delta with slight forward sweep of T.E.) 60° (L.E.) (f) 2·3 0 60° (L.E.)	NACA 64A004 modified Modified hexagon $t/c = 0{\cdot}045$ NACA Sections Root 63-010 Tip 63-012 NACA 64A007 NACA 0004-65 modified NACA 65-006·5	Full-scale flight tests of following aircraft: (a) X-1E (b) X-3 (c) D-558-II (d) F-100A (e) JF-102A (f) XF-92A		(a) M = 0·8, 1·0, 1·9 $Re_{max} \approx 1{\cdot}8 \times 10^6$ per ft (b) 0·7 < M < 1·2 $Re_{max} \approx 4 \times 10^6$ per ft (c) M = 0·8, 1·0, 1·8 $Re_{max} \approx 1{\cdot}2 \times 10^6$ per ft (d) 0·65 < M < 1·45 $Re_{max} \approx 3{\cdot}5 \times 10^6$ per ft (e) M = 0·8, 1·0, 1·2 $Re_{max} \approx 2{\cdot}4 \times 10^6$ per ft (f) 0·7 < M < 0·85 $Re_{max} \approx 2{\cdot}5 \times 10^6$ per ft	Chordwise and spanwise loadings on wing panel: *Aircraft (a)* $\alpha = 4°, 8°$ (M = 0·8); $\alpha = 4°, 7°$ (M = 1·0); $\alpha = 4°, 14°$ (M = 1·9). *Aircraft (c)* $\alpha = 4°, 7°$ (M = 0·8); $\alpha = 3°, 11°$ (M = 1·0); $\alpha = 5°, 9°$ (M = 1·8). *Aircraft (e)* $\alpha = 5°, 20°$ (M = 0·8); $\alpha = 5°, 10°$ (M = 1·0); $\alpha = 3°, 9°$ (M = 1·2) (N.B. Eleven deflections δe appropriate to flight condition.) For all aircraft: Normal-force coefficient C_N v α and the derivative C_{N_α}.		For calculation of wing loads: *Subsonic* (0·5 < M < 0·85) All wings: linear lifting surface theory (De Young and Harper, Allen, etc. Refs. 3, 58 and 42). *Transonic* (M = 1·0) Swept wing: linear lifting surface (Mangler, Ref. 55; Crigler, Ref. 10). Unswept wing: two-dimensional flat-plate and double-wedge theories, Guderley and Yoshihara (Refs. 65 and 62). *Supersonic* (M ⩾ 1·2) All wings: linear lifting surface.	'. . . a reasonable approximation of the span loadings can be determined for the low and moderate angle-of-attack range. The estimation of the chord loadings is less satisfactory, particularly in the neighbourhood of M = 1·0. In general, the calculated normal-force curve slopes compare favourably with those obtained from the flight data.'	In calculations the wings were assumed to be rigid flat plates of negligible thickness. The effect of fuselage interference was approximated by assuming the fuselage to act as a perfect reflection plane located at the wing-fuselage juncture. Since no check appears to have been made on the validity of the rigidity assumption, these results are of dubious significance in the present context.

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars: Planform: Aspect ratio Taper ratio Sweep	Lifting surface particulars: Basic section(s) camber and twist of unloaded surface	Details of tests: Wind tunnel or flight	Details of tests: Test configurations	Details of tests: Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Details of tests: Aerodynamic or flight parameters measured, with angle-of-attack range	Details of tests: Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
9	NACA RM H57 C25 (TIL 6819) (1958) A. E. Kuhl J. T. Rogers M. V. Little (26)	8·55 0·398 35° (c/4)	Root: BAC 233 Tip: BAC 236	Flight tests of Boeing B-52 airplane (slow-rate roller-coaster manoeuvres)	W ≈ 290 000 lb CG maintained at 26 ± 1 per cent mean chord by fuel transfer within body tanks; wing fuel held constant.	$0.5 \leqslant M \leqslant 0.82$ at 20 000 ft $0.7 \leqslant M \leqslant 0.90$ at 30 000 ft $46 \times 10^6 \leqslant Re \leqslant 75 \times 10^6$ $200 < q < 460$ (lb/ft^2) at 20 000 ft $200 < q < 360$ (lb/ft^2) at 30 000 ft	Airspeed and altitude. Normal accelerations at CG, tail, and 3 wing locations; elevator position; gross weight and CG position; pitch velocity at CG. Wing loads, wing deflections.	Variation with M and q of (i) basic air load on wing (i.e. at zero aeroplane acceleration); (ii) aerodynamic centre and spanwise cp of additional air load for lower (linear) lift region.	Method of Gray and Schenk (Ref. 60) used in conjunction with calculated structural properties and wind-tunnel aerodynamic data to determine air loads on flexible wing.	'The comparisons of the measured and calculated loads indicated that this method of predicting the loads appears reasonable for airplane configurations of this general type and speed range.' It is indicated that '. . . nacelle air loads can have a strong influence on the total wing loads and that an accurate estimation of the nacelle air loads may be important in predicting the wing loads.'	Note that the method of Ref. 60, which is, in essence, based on lifting-line theory depends for its accuracy on experimentally (W/T) determined values of section lift coefficient.

TABLE 6 (Contd.)

Item No.	Report No. Date Authors (Ref. List No.)	Lifting surface particulars		Details of tests					Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced			
10	NASA TM X-343 (1960) R. V. Doggett A. G. Rainey (27)	*Series A* 0·848 0 78° (L.E.) (Rigid and 2 flexible delta wings.) *Series B* 0·424 0 84° (L.E.) (Rigid and flexible delta wings.) *Series C* 0·848 0 78° (L.E.) (Rigid delta wing, area $\frac{1}{2}$ of Series A wing area; same body.)		Langley 2 ft transonic aero-elasticity tunnel.	Full-span wings in combination with single conical-cylindrical body.	$0·7 \leqslant M \leqslant 1·10$ $Re = 3·2 \times 10^6$ per ft (all models) and also $Re = 1·8 \times 10^6$ per ft (Series A models). $150 \leqslant q \leqslant 250$ (lb/ft^2) for high Re $80 \leqslant q \leqslant 130$ (lb/ft^2) for low Re	Lift, drag and pitching moment. $-4° \leqslant \alpha \leqslant 30°$ (measured at trailing edge).	Elastic deformations (bending in camber direction) of one of the Series A flexible models were determined by double-exposure photographic technique.	Various linear and non-linear theories for rigid low-aspect-ratio delta wings (Jones,[45] Küchemann,[66] Brown and Michael,[67] etc.). Linear theories for wings deformable in camber direction (Bisplinghoff *et al.*[68] and Garrick[56]). Subsonic lifting surface theory (Watkins, Woolston and Cunningham[71]).	'Good agreement between linear theory and experiment for $(dC_L/d\alpha)_{C_L=0}$. One non-linear theory (Küchemann) showed fair agreement with experimentally determined lift characteristics for all three configurations. No analytical method was completely adequate for prediction of C_L and C_m characteristics throughout entire α-range. Shapes of experimental and theoretical deflection curves compare favourably but theory predicts higher total deflection than found experimentally.	N.B. For theoretical calculations, models were assumed to consist only of Δ-wing (fuselage effects ignored). Experimental results indicated no appreciable effect of flexibility on lift characteristics, but a destabilising effect on pitching moment. (N.B. α was measured at trailing edge.)

TABLE 6 (Contd.)

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11	NASA TR R-58 (1960) J. P. Mugler (28)	4·0 0·15 45° ($c/4$)	NACA 65A206 $a = 0$ at root. Linear variation in t/c to NACA 65A203 $a = 0{\cdot}8$ at 0·5 s, then t/c constant to tip.	Langley 8 ft transonic pressure tunnel	Full-span wing-body combination (mid-wing).	$0{\cdot}8 \leqslant M \leqslant 0{\cdot}98$ $2{\cdot}6 \times 10^6 \leqslant Re \leqslant 2{\cdot}85 \times 10^6$ $P_0 = 1$ atmos	Pressure coefficients at 6 semi-span stations on wing. $\alpha = -4°, -2°, 0°, 2°, 4°, 8°$	Span loading distributions; chordwise cp positions; twist distributions ($\Delta\alpha$) along $c/4$, $c/2$ and $3c/4$ lines.	Method presented in report is an iterative method, based on lifting-surface concepts, the boundary conditions being satisfied at 3 control points on each of 4 spanwise loading stations, and the effects of chordwise deformation being accounted for in the calculation. No allowance for viscous effects.	At M = 0·8 agreement is good for load and twist distributions for $0° \leqslant \alpha \leqslant 4°$. Outside this angle-of-attack range agreement is poor. With increase in M agreement at $\alpha = 2°$ and 4° becomes increasingly poorer. At $\alpha = 0°$ agreement still good at M = 0·98. Deteriorations in agreement are ascribed to development of large regions of mixed flow over wing.	Experimental data taken from Refs. 24 and 33 (see Item 7 of Table 6 and Item 3 of Table 4).

TABLE 6 (Contd.)

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		Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack	Other parameters measured or deduced			
12	NASA TN D-974 (1961) F. V. Bennett (29)	4·0 0 45° (L.E.)	Double wedge $t/c = 2\%$. surface unwarped when unloaded.	Langley 9 by 18 inch supersonic aero-elasticity tunnel.	Wing cantilever-mounted in tunnel with faired semi-circular body attached to tunnel wall and wing extending through it for attachment to strain-gauge balance.	$1{\cdot}30 \leqslant M \leqslant 4{\cdot}0$ $250 \leqslant q \leqslant 2000$ (lb/ft^2)	Total normal force, pitching moment and bending moment. $2° \leqslant \alpha \leqslant 10°$	Static aero-elastic deflections deduced from double-exposure photographs of wing in deformed and undeformed positions.	An influence-coefficient formulation of the equation of aeroelastic equilibrium is used. The method of solution of this matrix equation is dependent on the type of aerodynamic theory used; solutions using linearised potential and piston theories are discussed*.	Generally favourable agreement between potential theory calcula-tions and experimental results for normal force, pitching moment and deflections. Satisfactory agreement of piton theory calculations for the above at M = 4·0; less satisfactory at M = 3·0. Agreement poor for bending moment coefficients†.	*Potential theory for subsonic L.E. at M = 1·30, (Ref. 59) and for supersonic L.E. at M = 1·64, 3·0 and 4·0. Piston theory for M = 3·0 and 4·0. †Experimental results considered inaccurate due to balance insensitivity.

TABLE 6 (Concluded)

		Lifting surface particulars		Details of tests							
Item No.	Report No. Date Authors (Ref. List No.)	Planform: Aspect ratio Taper ratio Sweep	Basic section(s) camber and twist of unloaded surface	Wind tunnel or flight	Test configurations	Flow particulars Mach No. (M) Reynolds No. (Re) Dynamic pressure (q) or Total pressure (P_0)	Aerodynamic or flight parameters measured, with angle-of-attack range	Other parameters measured or deduced	Theoretical methods employed	Degree of agreement between theory and experiment claimed by author	Comments
13	ASD-TDR 63-366 (1963) F. H. Durgin C. J. Bartlett (30)	2·0 1·0 0°	Hexagonal section t/c = 4% from $c/4$ to $3c/4$. Surface un-warped when unloaded.	Closed-return, continuous-flow wind-tunnel. 18 by 24 inch test section at M = 2·0 18 by 18 inch at M = 3·0.	Half-wing mounted vertically, supported on pylon of heavy steel construction which prevented any wing-root twist.	M = 2·0 P_0 = 5, 10 psia M = 3·0 P_0 = 10, 30 psia	Chordwise pressure distributions at 5 semi-span stations. $\alpha = 2°, 4°$ $\alpha = 3°, 9°$	Changes in local angles of attack measured by optical system consisting of point light source, mirrors, screen and camera. Structural influence coefficients measured.	'Field' matrix approach used to analyse data: (i) to determine accuracy of aerodynamic matrix in predicting pressure distribution; (ii) to check how well the structural matrix, used with measured loads, predicts equilibrium slopes over the wing; (iii) to test accuracy of complete aero-elastic equations.	Agreement between theoretical and experimental pressure distribution varies from 'quite good' to 'quite bad' but, except near the leading edge is considered better than when predictions are made by linear theory*. Comparison between theory and experiment for angular deflections was 'somewhat disappointing'.	Considerable difficulties were encountered in the experimental work and results are probably not very reliable. *There is confusion between text and Figs. 5 and 6 as to the symbols used to distinguish the two sets of theoretical results. An interchange of symbols in the figures appears necessary to validate the author's claim.

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