ICAS-82-5.5.3

VARIABLE GEOMETRY AEROFOILS AS APPLIED TO THE BEATTY B-5 AND B-6 SAILPLANES

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ABSTRACT

Two variable-geometry aerofoils using flexible surfaces have been developed by Beatty for his B-5 and B-6 sailplanes. The B-5 has a basic Eppler 1001 section with a flexible upper surface which is raised to form a thick, high-camber profile for thermalling. The B-6 uses a Wortmann FX 05-H-126 section with flexible upper and lower surfaces over the rear 40% of the chord. Predictions for the polars of the two sections using the Eppler and NASA viscous, two-dimensional computer programs are presented. Predictions for the overall aircraft polars are compared with flight-test measurements obtained by conventional methods and by using a prototype glide-angle indicator.

1. INTRODUCTION

In the continuing endeavour to achieve higher cross-country speeds, the conventional sailplane wing appears to have been developed to its full potential. In the past twenty years, with the advent of the Eppler and Wortmann series of aerofoil sections, and of glass and carbon re-inforced plastics, a high degree of laminarisation of the boundary layer has been made possible. (However, perhaps, the major contribution of these sections has been in widening the laminar flow bucket, rather than in increasing its depth.) Further progress requires the application of sophisticated techniques, such as boundary layer suction (or blowing) or variable-geometry.

The racing sailplane flies cross-country using a technique of alternate low-speed thermal-climbs and high-speed inter-thermal "dashes", and the problem facing the designer is to achieve a low, relative sink rate in both these phases of flight. The climb phase requires the wing to have a large area and a high usable lift coefficient to reduce the stalling speed; a low profile drag coefficient at this high lift coefficient, and a large span to minimise the induced drag. On the other hand the dash phase requires the wing to have a small area and a very low profile drag coefficient at low lift coefficients to minimise the profile drag which is dominant at high speed.

To summarise these requirements we have:

For the Thermal Climb

(a) Large area (for low wing loading)
(b) Large span (for low induced drag)
(c) Thick, high-camber section (for high $C_{L_{max}}$ with low C_{D_o}).

For the Inter-Thermal Dash

(d) Small area (short span) (for low profile drag)
(e) Thin, low-camber section (for low C_{D_o} at low C_L).

These conflicting requirements are capable of complete solution only by the use of variable geometry. Thus, in order to satisfy (a) and (b) at low speed and (d) at high speed, the preference would be for variable span, with the second choice being variable chord length. To satisfy (c) at low-speed and (e) at high speed, full-span variable thickness and camber are required. Thus, on paper, the ideal solution is a wing with variable span, thickness and camber. However, the construction of such a wing would be formidable.

Partial solutions to the problem have been tried over the years. Thus, Beatty and Johl[1] in South Africa, pioneered the use of chord-increasing flaps on the B.J. series of sailplanes. These aircraft employed Fowler flaps for the thermalling phase in conjunction with a thin laminar-flow aerofoil for the dash. They held five world records in the sixties. In the next decade the Canadian Gemini[2] used double-slotted flaps. The British Sigma[3] and German S.B.11 and Milomei were attempts to carry the chord-increasing concept to the extreme, using an aerofoil specially designed by Wortmann. Meanwhile the F.S.29[4] was developed at Stuttgart University to explore the possibilities of a variable-span, telescopic wing. However, in 1975 Beatty decided to adopt a different approach - that of varying the camber and thickness of the wing.

2. DESIGN PHILOSOPHY

As with the earlier B.J. series the primary consideration in designing the B-5 wing was the use of a "no-compromise", thin, laminar-flow section for the high-speed regime. In 1976 Prof. Eppler designed such a section for the B-5. This E1001 profile, of 12,5% thickness, was to have 70% laminar flow on the upper surface and 80% on the lower surface in the "bucket". It was predicted to have the very low minimum profile drag coefficient of 0,0034, and a bucket lift coefficient width of 0 to 0,6 at a Reynolds' number of 3 million.

At high speed the induced drag is very small and the wing profile drag makes the major contribution to the total drag of the sailplane. Consider, for instance, an open-class sailplane of aspect ratio 30, flying at around 200 km/h (110 kts) at a lift coefficient of 0,2 and a Reynolds' number of 3 million. Assuming a parasite drag coefficient of 0,0022 the glide ratios (L/D) for the aircraft using either the Wortmann FX67-K-150/17[5] (the Nimbus II section) or the E1001 are as shown below:

AEROFOIL	FX67-K-150	E1001
Thickness	15%	12,5%
C_L	0,2	0,2
C_D Parasite	0,0022	0,0022
C_D Induced	0,0004	0,0004
C_D Profile	0,0054	0,0034
C_D Total	0,0080	0,0060
L/D	25	33,3

Thus, there is a gain of over 8 points, or one third, in the glide ratio due to the reduction in the profile drag coefficient for the thinner section.

As it would not be possible to develop high lift coefficients solely by deflecting a flap on the thin section, Beatty had the idea of changing both the camber and the thickness for the low-speed regime, by raising the top surface of the wing to approximate to the shape of the FX67-K-150 profile.

3. THE B-5

Initially it was proposed to use an elasticised upper surface to the wing which would be inflated for low speed flight. However, it would not have been possible to control the shape of this surface in the raised position, so that the system shown in Figure 1 was finally adopted. The upper surface of the wing consisted of an inner, load-carrying skin with an outer, 75% chord, flexible Dural plate fixed at the leading edge, but movable chordwise at its rear edge. With the rear edge of this outer plate pulled backwards, it was held in contact with the inner skin, and took on the E1001 profile. However, with the rear edge pushed forwards, the plate adopted a new shape pre-determined by the positioning of rows of T-shaped strips fixed to the underside of the plate. In conjunction with the upper surface movement a plain 20% chord flap could be deflected downwards. Thus the low-speed thermalling section was 16,25% thick with 4% camber before flap deflection. In order to remove buckling loads, the outer skin was made in sections of one metre width spanwise with silicone rubber sealing strips in between. The skin was vented by a single hole at 60% chord to balance the airloads and to relieve internal pressure during the transition from high to low positions.

Initially Beatty had been worried about the change from low to high camber. Although the movable upper surfaces rigidly conformed to the profile shape in the high and low camber positions, during transition they were fixed only at their leading and trailing edges. In fact, no problems were experienced and the wing was flown to indicated speeds of 315 km/hr (170 kts) with low camber and (unintentionally!) to 240 km/hr (130 kts) with high camber.

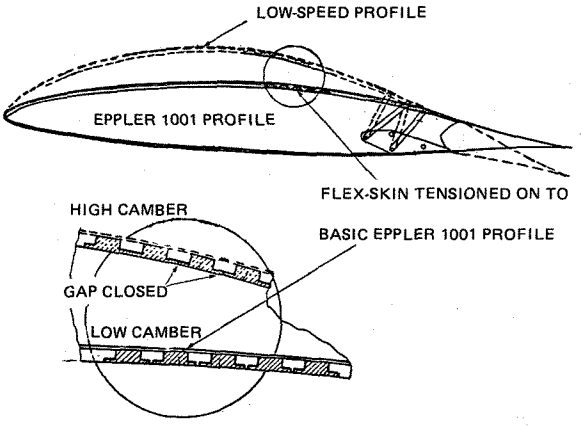

FIGURE 1 B-5 FLEXIBLE WING SECTION

The basic wing was of 15 m span and constant 850 mm chord. In its original configuration 2 m internally-telescopic tips were also fitted, but these were removed after early flight trials and replaced by fixed 2,1 m tips giving a span of 19,2 m.

The reasons for this were as follows:

(1) The severe discontinuity in lift at the junction of the main wing and tip,

(2) Quite severe scratching of the GRP surface of the tips as a result of sand becoming embedded in the nylon bearings through which the tips slid,

(3) Indexing of the tip, so that it retracted absolutely flush, proved very difficult,

(4) It was difficult to prevent air leaks in both the retracted and extended configurations.

The fuselage was made of circular crossection to the "waisted" shape which had been tunnel-tested by Althaus.[6] In order to reduce interference drag the wing was positioned on a pylon above the fuselage. The General Arrangement of the aircraft is shown in Figure 2. As can be imagined, a high level of engineering skill was needed to route the controls through the pylon to operate the movable upper-surface, telescopic tips, flaps, and ailerons. (In a later modification additional 10% split flaps operated by a Freon pressure system were added to the plain flaps). A 12 volt D.C. electrical system was used to operate the upper-surface and the telescopic tips. Transition time was 5 seconds for the upper-surface and 6 seconds for the tips. In the low-speed configuration 6° aileron droop was used in conjunction with 10° flap deflection.

As described in the Appendix, the Eppler single-element aerofoil program was used to analyse the B-5 section in both the low- and high-speed configurations. (The same program was used by Eppler to design the E1001 section). Figure 3 shows the results for the section polars at Reynolds' numbers of 1,5 and 3 million respectively and with up to 12 degrees of flap deflection in the low-speed configuration.

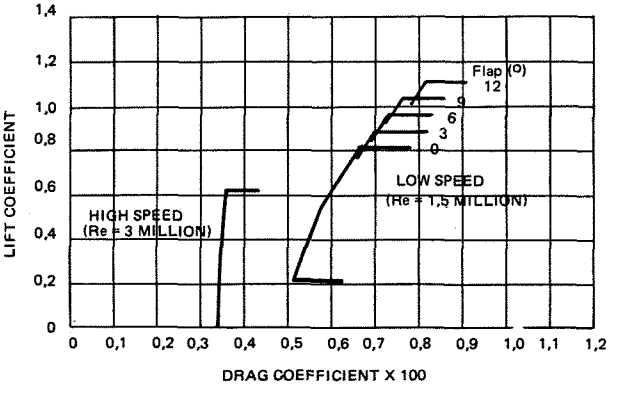

FIGURE 3 B-5 THEORETICAL SECTION POLARS

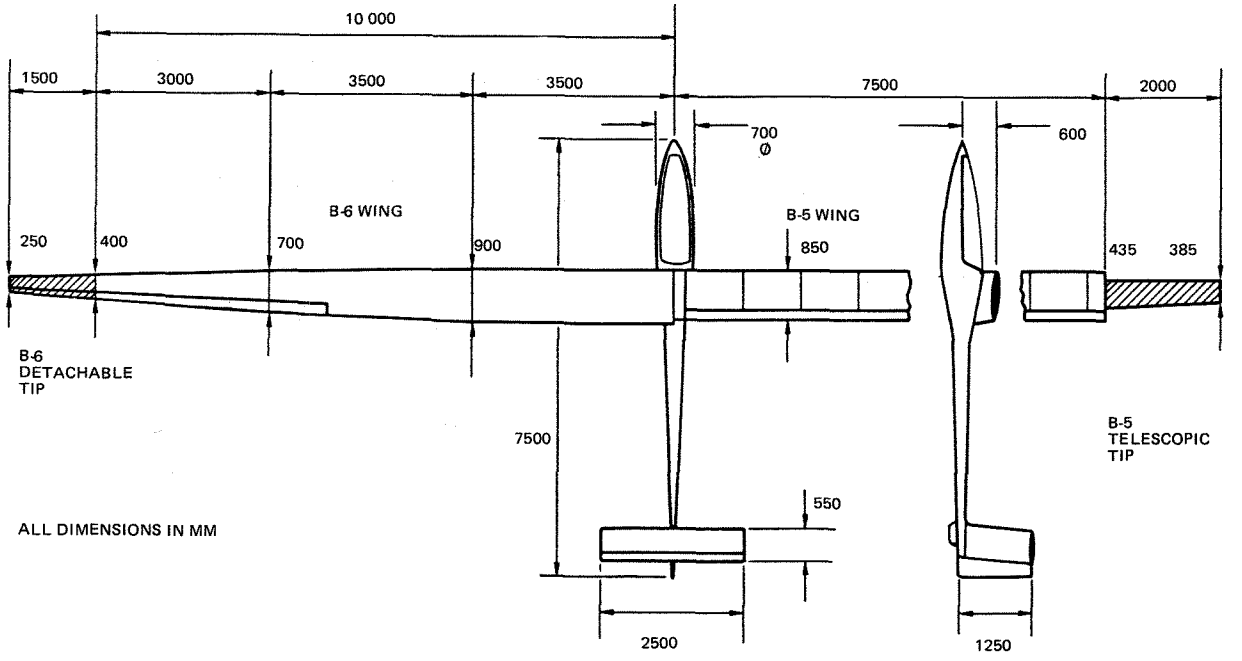

FIGURE 2 GENERAL ARRANGEMENT OF B-5 AND B-6

4. B-5 TEST RESULTS

Measured polars for the B-5 aircraft in the high-speed configuration are shown in Figure 4 in comparison with predictions. The mass of the aircraft including pilot was 651 kg (1435 lb) giving a wing loading of 420 N/m^2 (8,70 lb/ft^2) on the fixed-span configuration of 19,2 m (63 ft). The aspect ratio was 24,3.

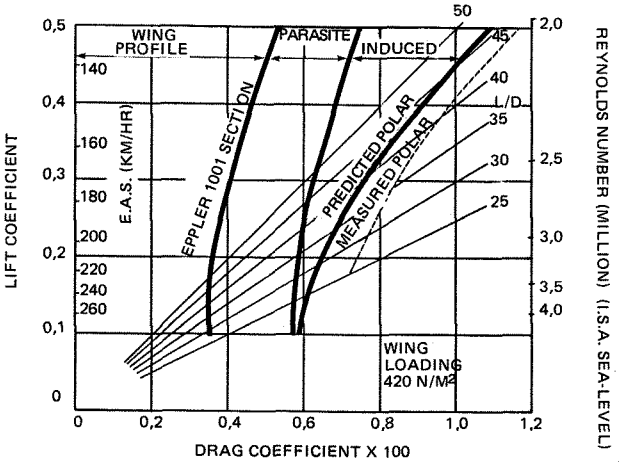

FIGURE 4 B-5 PREDICTED AND MEASURED AIRCRAFT POLAR

The performance of the B-5 was disappointing in both low and high-speed configurations. The reasons for the loss of high-speed performance are thought to be as follows:

(i) The high-speed E1001 section appears to be too "refined" in an attempt to get extreme lengths of laminar flow on both surfaces. Thus even at modest lift coefficients the upper surface boundary layer becomes critical at about 20% chord, although transition is not until 70%. Moreover, the transition region is only 5% chord which is insufficient to prevent bubble formation on one or other surface at almost all incidences. Figure 4 shows an increase of total drag coefficient of approximately 0,001 over the whole high-speed range. However, this cannot justifiably be attributed to the E1001 section because profile drag measurements were not made. Furthermore a number of minor effects must have contributed to a drag increase on the wing, viz:

(ii) the presence of the chordwise breaks in the upper wing skin must have given rise to wedges of turbulent flow arising from the leading edge, and causing the early onset of transition.

(iii) The centre-section and tips were fixed in the raised position, leaving a discontinuity in the high-speed configuration.

Despite these effects, the glide ratio was still better above 170 km/hr (90 kts) than an ASW17 or Nimbus II fully loaded.

More serious was the poor performance in the climb configuration. Early tests indicated a maximum lift coefficient of only 1,23 with the upper surface raised plus 10 degrees of aileron droop. To improve this value a 10% chord split flap was fitted to the full-span 20% plain flap and aileron, and was deflected 10 degrees. By this means the maximum lift coefficient was raised to 1,31 giving a stalling speed of 82 km/hr (44 kts) at the above wing loading. A contributory factor to the low overall maximum lift coefficient was that a progressive reduction of flap deflection towards the tips had been used to improve the spanwise loading on the constant chord wing. The available lift is then only about three quarters ($\pi/4$) of the lift given by the uniformly loaded wing. However, despite this reduction, the spanwise loading was still not fully elliptic causing some increase of induced drag and further detracting from the climb performance.

In 1980 it was concluded that further development of the B-5 wing was not worthwhile. However, although the aircraft had failed to live up to expectations, the construction of a variable thickness wing had been shown to be physically possible.

5. THE B-6

In 1980, despite the high-level of investment in the B-5 wing, it was decided to abandon it in favour of a new wing. With this new wing the aircraft is designated the B-6.

Although the B-6, like the B-5, uses the concept of a fixed chord with flexible surfaces, the principle is different. Having failed to achieve a satisfactory maximum lift coefficient on the B-5 wing, Beatty decided to use as the basic section for the B-6 a profile with a better high lift capability and a wider laminar-flow bucket. He chose the Wortmann reflexed, helicopter section FX05-H-126 of 12,6% thickness, for which wind tunnel

results were given in Ref. 7. This section has sufficient forward camber to enable a high maximum lift coefficient to be obtained. At the same time a minimum profile drag coefficient of 0,0050 is obtained in the "bucket" at a Reynolds' number of 1,8 million. The low-drag lift coefficient range of the basic section is from 0,4 to 1,1. However, on the B-6, the rear 40% of both upper and lower surfaces are flexible, although the section thickness remains constant (Figure 5).

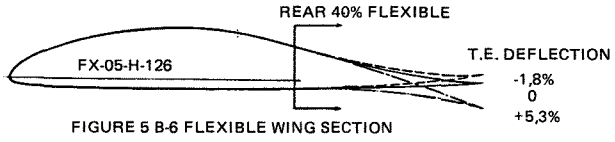

FIGURE 5 B-6 FLEXIBLE WING SECTION

By bending the trailing-edge of the section up by 1,8% chord and down by 5,3% chord theoretical predictions using the NASA program show that the bucket may be shifted as shown in Figure 6 to cover the range of lift coefficient from 0,25 to 1,45. The section in its basic form has the advantage that the wind-tunnel test results are available from reference 6, so that it is not an untried section like that used on the B-5.

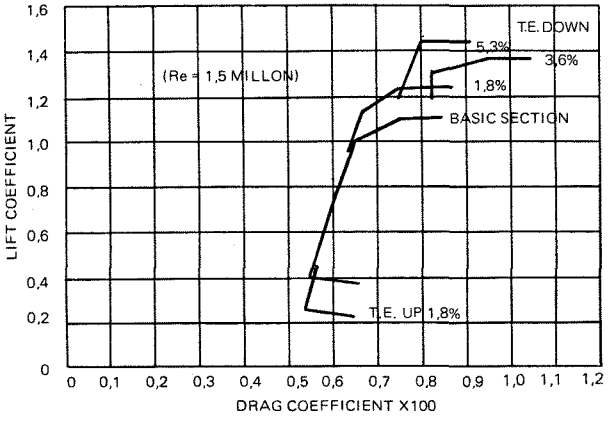

FIGURE 6. B-6 THEORETICAL SECTION POLARS

The B-6 wing has a parallel inboard panel with two outboard panels of different taper to approximate to the elliptic planform (Figure 2). On the inboard two panels the rear 40% of the chord is flexible, whereas on the outboard panel a 30% chord plain aileron is fitted. The ailerons are drooped up to 7 degrees in conjunction with downwards deflection of the inboard panel trailing-edges. To avoid tip stall an extreme aileron differential of 4:1 is used, (although this may be adjusted on the ground). Thus, the aileron travel is -28° to +7° undrooped, and -21° to +14° with 7° droop.

1,5 m detachable tips with aileron extensions, may be fitted, increasing the span from 20 m to 23 m. At 20 m span the wing area is 15,2 m^2 and the aspect ratio 26,3; at 23 m span the area is 16,2 m^2 and the aspect ratio 32,7.

136 litres of water ballast may be carried in the inboard wing leading edges.

6. B-6 TEST RESULTS

The B-6 first flew in March, 1981 and during 1981 was grounded several times for modifications. In December 1981 it was found that water from the ballast tanks had penetrated the sealer and had caused expansion of the wooden leading-edge riblets, so that a major rebuild was necessary. As a result a comprehensive series of flight tests on the current configuration has not been carried out. Furthermore the proposed wake-traverse drag measurements on the wing have not been made. Full flight tests will be carried out later in 1982.

The climb performance of the B-6 has proved entirely satisfactory. Downward trailing-edge deflection of 5,3% chord is normally used and a maximum lift coefficient of 1,5 has been obtained giving a usable lift coefficient of 1,36. Although the low-speed polar has not been investigated, climbs alongside a Nimbus II have shown a superior performance.

Figure 7 shows the predicted drag polar for the B-6 with flight test results in the high-speed regime. For these flight tests the 23 m (75 ft) configuration was used at a mass of 620 kg (1370 lb) giving a wing loading of 380 N/m^2 (7,9 lb/ft^2).

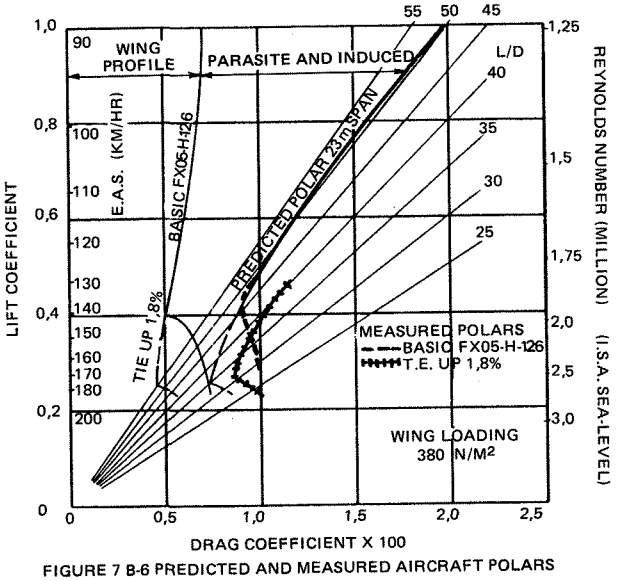

FIGURE 7 B-6 PREDICTED AND MEASURED AIRCRAFT POLARS

Comparison of the measured and predicted high-speed polars shows that the basic FX05-H-126 section does have a bucket range down to a lift coefficient of 0,4 as predicted by Wortmann. However, upward movement of the flexible trailing-edge appears to cause an immediate increment in profile drag coefficient of about 0,0015 at lift coefficients above 0,4 reducing to 0,001 at a lift coefficient of 0,27, where the lower bucket edge occurs. This means that whereas an L/D of 33,5 was predicted at a lift coefficient of 0,25 with the trailing edge up (at 185 km/hr or 100 kts) the achieved L/D at this lift coefficient is about 28. At a lift coefficient of 0,2 the drag is the same with or without the trailing-edge up. In actual flying competitions Beatty has found it not worthwhile to raise the trailing-edge at all because of the penalty above a lift coefficient of 0,36 shown in Figure 7.

Various explanations for the drag increment with trailing edge up have been suggested. Perhaps the major contribution may be due to the fact that the lower surface does not bend smoothly from the 60% chord point as shown in Figure 5. (A parabolic flexure was assumed in the computer analysis). In practice there is a noticeable change in shape at this point with the trailing edge up, which may cause a premature transition in the very slack pressure gradients experienced on the lower surface at low lift coefficients. However, this would not in itself be sufficient to cause the drag coefficient increment of 0,001 to 0,0015 experienced. An allowance must also be made for the aileron hinge-line at 70% chord on the outer wing panel. During the current overhaul it is proposed to form the lower surface of the wing to the correct profile with the trailing-edge in the up position to avoid this problem.

As with the B-5 it could be misleading to base estimates of wing profile drag on measurements of aircraft glide ratios, and it is proposed to measure this drag directly by the wake-traverse method when the aircraft is flying again. However, as the predicted glide ratios are achieved above a lift coefficient of 0,4, the aircraft parasite drag estimate cannot be far out. Therefore the deductions made above regarding the behaviour of the wing profile drag at low lift coefficient should be valid.

Too much emphasis must not be placed on the loss of top speed performance. The B-6 has already proved itself a competitive aircraft. In 1981 Beatty won a race in it over a 300 km triangle on a weak day during the South African National Championships (beating the World Champion, George Lee, on the day). Had it not been for the problem of the water ballast soaking in to the leading-edge structure, the aircraft might well have won overall. As it was, Beatty was forced to fly the latter half of the Championships, on strong days, without water and with a distorted leading-edge profile.

7. CONCLUSIONS

The B-5 and B-6 sailplanes have been pioneers in a new form of variable geometry that of flexible wing surfaces. There is a truism which states that the product of a highly developed technology will always beat an undeveloped application of a new technology. This has certainly been true of the B-5 but it is believed that it will not be true of the B-6 in its final configuration.

In retrospect several things could have been done better on both the B-5 and B-6. It would, no doubt, be possible to make a flexible upper surface without chordwise breaks, and with a tapered planform to improve on the B-5. It would have been better to have constructed the basic B-6 wing in the high-speed (trailing-edge up) configuration to ensure profile accuracy for this very important flight régime. Or, alternatively, a reflexed section (such as NACA 8H12) with a laminar-flow bucket down to a lift coefficient of 0,25 could have been used instead of the Wortmann FX05-H-126. However, hindsight is always 20:20!

In conclusion it can be said that the B-5 and B-6 have proved the aerodynamic and engineering principles involved without, so far, achieving the anticipated performance improvements. However, development of the B-6 continues.

APPENDIX 1 GLIDE-ANGLE INDICATOR

For the later flight trials on the B-6 an experimental glide-angle indicator was tested. The principle of such an indicator is that the inclination to the horizon of the glide-path through the surrounding air is the same, irrespective of the vertical motion of the surrounding air. Therefore, if it is possible to measure this angle accurately, a direct reading of L/D may be obtained whether or not thermals or downdraughts are present.

However, to detect the true motion of the aircraft through the surrounding air, the indicator must be situated in air which is undisturbed by the presence of the aircraft. Thus, the indicator must be far enough away from the aircraft for any disturbance to be negligible, or in a position closer to the aircraft where it is known that the disturbance is zero. The former position could be obtained using a trailing bomb of the type used extensively for static pressure calculation. However, there are several disadvantages to this system, not the least being the difficulties of deploying and retracting the bomb. Therefore, it was decided to try and position the indicator on a fixed boom at a point where the disturbance velocity was zero.

It is clear that the dominant perturbation field around the aircraft is that arising from the wing vortex system. The trailing vortex system gives rise to downwash at any point on the aircraft plane of symmetry, whereas the bound vortex system give rise to upwash ahead of the wing and downwash behind it. There is, therefore, a locus of points ahead of the wing along which there is a net zero induced vertical velocity. Assuming elliptic spanwise and "flat-plate" chordwise loading in an extended lifting-line analysis, Figure 8, was developed showing the downwash field in the plane of symmetry expressed as a fraction of the downwash at the bound vortex ($C_L/\pi A$). Figure 8 shows that zero downwash exists at points which are generally further above (or below) the wing than ahead of it. A good null position exists at 0,1 semi-span ahead, and 0,3 semi-span above, the wing centre-of-pressure.

Here the downwash velocity gradients are small and positioning of the indicator is not critical. However, for a 20 m span wing, this point is 3 metres above the wing, which makes the boom rather long. A compromise position was therefore chosen at 0,05 span forward, and 0,2 span above, the wing (0,5 m and 2 m on a 20 m span wing).

It must be borne in mind that the glide-angle is very small, and even a modest tolerance on L/D measurement requires very small angular resolution from the indicator. For instance, if L/D is 40, the glide angle is 1,43 degrees. If this L/D occurs at a lift coefficient of 0,5 on an aspect ratio 30 wing the downwash angle at the bound vortex is 0,30 degrees and the contours of W = +/- 0,2 in Figure 8 therefore represent increments of +/- 0,06 degrees in downwash angle which would be equivalent to +/- 1,5 on L/D.

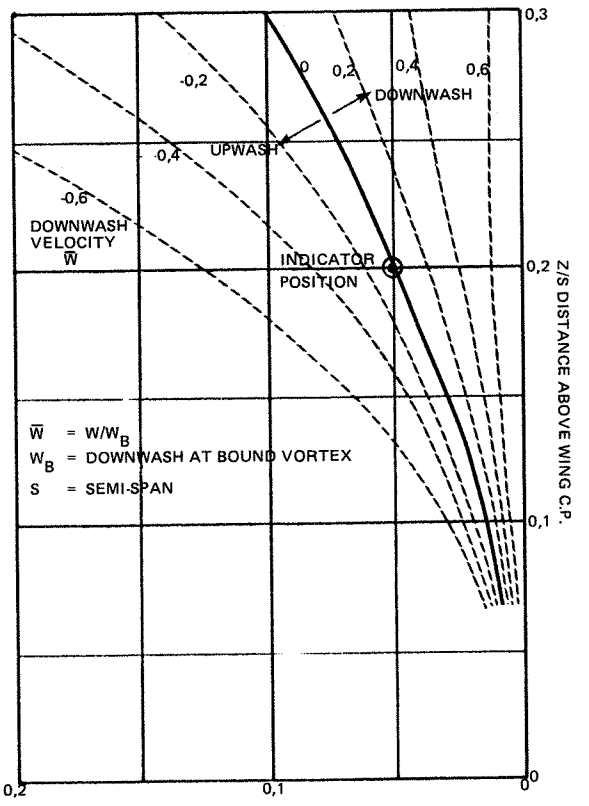

FIGURE 8. DOWNWASH FIELD ON AIRCRAFT CENTRE-LINE

For the prototype indicator a Penny and Giles inclinometer, model number IT24/4, was used. This instrument has a range of +/- 2 degrees and a sensitivity of 650 mV per degree when powered by a stabilised 10 volt D.C. supply. Its output was displayed on a digital voltmeter on the B-6 control panel. The inclinometer was supported in a streamlined (Althaus) body of revolution with cruciform fins, mounted via a spherical bearing on a 'boom' extending from the centre-section of the B-6 wing as shown in Figure 9. The fins were positioned in the null position described above.

The drag coefficient of the body and the boom was estimated to be 0,0004 based on the B-6 wing area of 16,2 m^2. A correction based on this value was made to measured glide ratios.

One of the problems with such an indicator is that of zero calibration. Adjustable mounting screws in the body enabled the inclinometer to be set exactly on the body longitudinal axis whilst on a level surface-table. However, no attempt was made to check that the body "aerodynamic axis" was the same as the geometric axis - a very difficult task when it is necessary to know the direction of a calibrating

airflow to an accuracy of about 0,02 degrees. (Available wind-tunnels were known to have flows inclined at around half a degree to their axes).

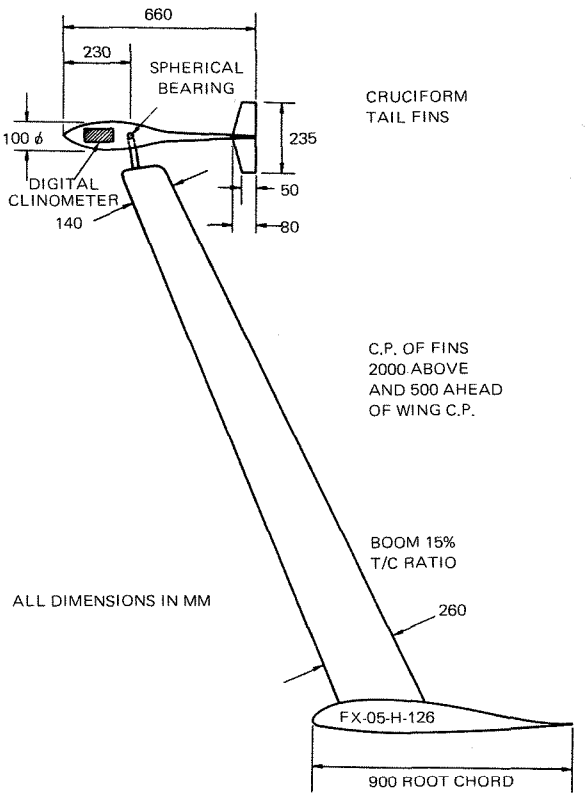

FIGURE 9. B–6 GLIDE ANGLE INDICATOR

One major problem encountered so far in flight trials has been that of inadequate damping in the system. Passive electronic damping in the form of a simple RC network was initially provided with a time constant of 4 seconds. This was found to be inadequate, so that the time constant was raised to 12 seconds which enabled satisfactory results to be obtained in smooth air. However, in the presence of any degree of air turbulence, time constants of the order of 30 seconds may be necessary.

The indicator is still in an experimental form, and further testing is necessary before it can be shown to be satisfactory. A number of criticisms may be levelled at the system, viz:

(i) The zero downwash position is a function of the aircraft type - i.e. wingspan and spanwise lift distribution (This could be overcome by using an adjustable boom, or the trailing bomb system could be adopted).

(ii) An overall measurement accuracy of +/- 1 on L/D is very difficult to achieve. (This appears at this stage to be a valid criticism. However, conventional "timed glides" are notoriously erratic, and require statistical reduction to produce marginally satisfactory polars.)

Perhaps the most useful application of the indicator is seen to be in giving relative, rather than absolute, glide ratios, before and after making some modification to a prototype aircraft like the B-6.

APPENDIX 2
AEROFOIL ANALYSIS PROGRAMS

In order to predict the characteristics of the B-5 and B-6 wing sections the author used two existing computer programs for viscous flow; the Lockheed-NASA[8] and Eppler[10] programs.

The first program used was, in fact, a version of the NASA multi-component aerofoil analysis program produced by the North Caroline State University[9] (NCSU). The NCSU program was developed for single-element aerofoils using the basic sub-routines of the NASA program, but was intended mainly for the estimation of the profile drag of light aircraft wings. For this reason the NASA sub-routine devoted to the evaluation of the turbulent boundary-layer separation and conventional trailing-edge stall was omitted, although the sub-routines for laminar bubble formation and bursting were retained. As the main interest in this work was in the laminar flow "bucket", the omission of the turbulent stall was not important.

In the NASA program the aerofoil contour is split into a series of straight-line segments on each of which constant vorticity is assumed. Then the program uses the well-known iterative procedure of alternately calculating (a) the pressure distribution and (b) the boundary-layer growth, adding the displacement thickness to the aerofoil contour. After convergence the skin friction is integrated and added to the pressure drag to obtain the profile drag coefficient. However, the alternative drag value calculated by the Squire-Young method was found by the author to give better agreement with experiment. This method relates the drag coefficient to the momentum deficit in the wake at infinity downstream. In practice this momentum

deficit is expressed in terms of the boundary-layer momentum thickness at the trailing-edge and the velocity outside the boundary-layer at the same place.

The Eppler program uses a higher-order "panel" method with linear vortex distributions on curved segments. However, it does not apply the above iterative procedure, the boundary-layer characteristics being calculated assuming the inviscid pressure distribution only. (This has been remedied in a later up-date of the program.[11] The drag coefficient is evaluated using the Squire-Young method only.

Because of the omission of the boundary-layer iterations the Eppler program has the advantage of much shorter running time. It was therefore used extensively to evaluate the characteristics of the B-5 aerofoil sections. However, comparison of the pressure distribution, no-lift angle and pitching moment of the FX05-H-126 section used on the B-6, with the experimental results of Ref. 6, showed that the negative lift loop due to the reflexed trailing-edge was over-estimated by the Eppler program. It was clear that, in practice, the reflexing was masked by the boundary-layer, giving rise to less down-load on the trailing edge. After four iterations the NCSU program gave satisfactory agreement so that this program was used exclusively to evaluate the B-6 section (Figure 10). However, a slight adjustment to the Squire-Young drag coefficient was necessary to match the experimental data of Reference 6.

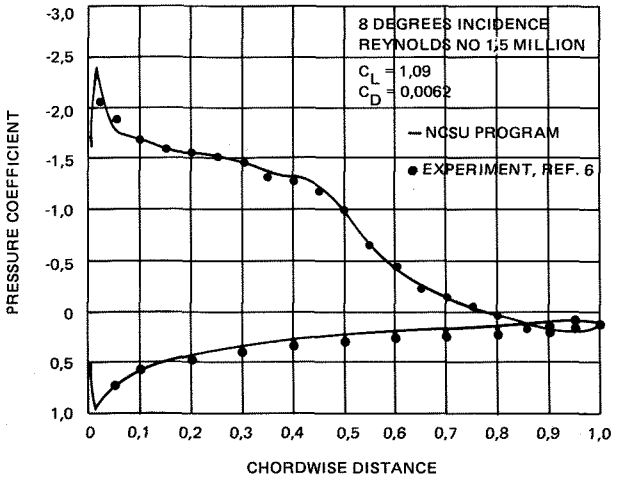

FIG. 10 PRESSURE DISTRIBUTION ON FX05-H-126

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