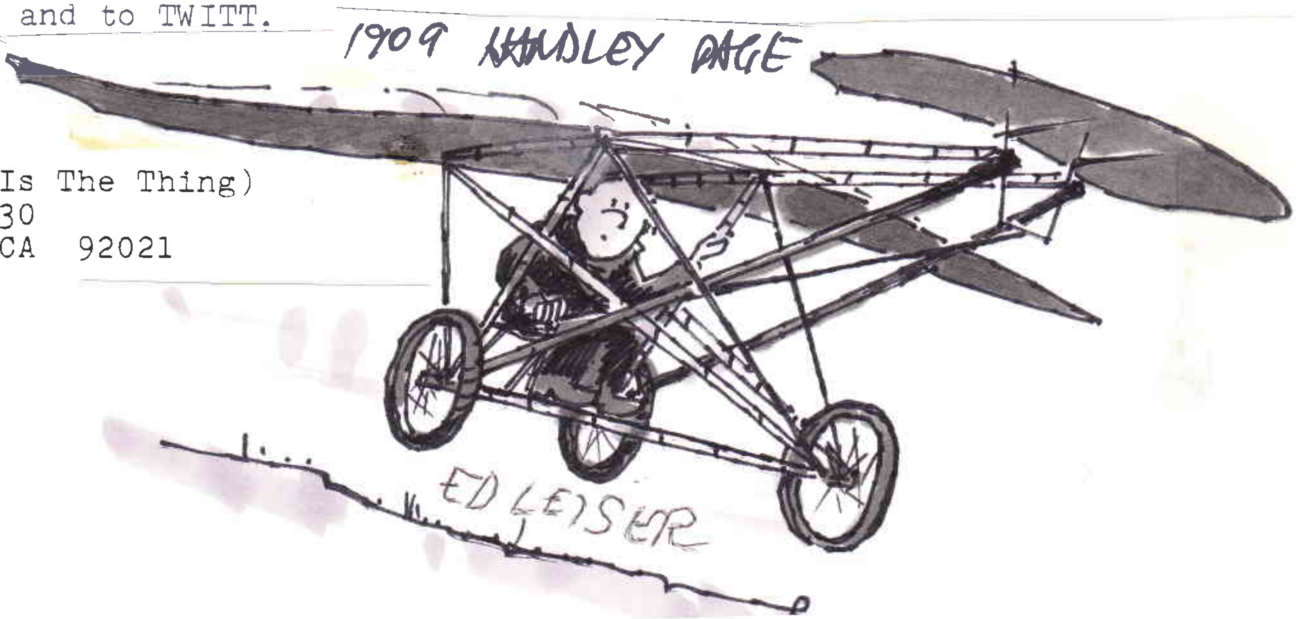
No. 5

TWITT
NEWSLETTER

November 1986

MARC de PIOLENC, EDITOR & SECRETARY



TWITT
(The Wing Is The Thing)
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MINUTES OF MEETING, 18 OCTOBER 1986

The fifth TWITT meeting convened at Gillespie Field on 18 October 1986. Present were Jack Green, Phillip Burgers, Pete Girard, Bruce and Georgie Carmichael, Cathy and Irv Culver, Bill Hannen, Ed Lockhart, Bob Peck, Andy Kecskes, Ray Johnsen, Harold Pio, Jim Neiswonger, Hernan Poznansky, Ralph Wilcox, Klaus Savier, Bob Fronius and Jeff Sawyer. Your Editor, to his disgust, found that his Army Reserve unit had scheduled a drill on the weekend of the 18th and 19th; his place was filled by a tape recorder. Unfortunately compressors, aircraft and other background noise are faithfully reproduced on the tape, making it a bit of a strain to listen to. If these minutes seem a bit sketchy, that's why.

The featured speaker was Irv Culver, who has been connected with aviation for so long that some authorities believe that he persuaded the Wright brothers to get out of the crowded bicycle-repair field and into something more lucrative. Irv (as far as your Editor can determine) covered the material in his flying-wing paper (published in issue no. 4) but went into more detail in describing the derivation of the simplified equations presented in that paper. But he also discussed another topic of considerable interest: flutter. Irv's thoughts on the "Physics of Flutter" appear elsewhere in this issue. A key point made in his talk is that there is no shortage of good aerodynamic data; the problem is that most of it is presented in a way that makes it unintelligible to most mortals. Irv seems to have gone to considerable trouble to put at least some of it within reach. Thank you, Mr. Culver.

NEXT TWITT MEETING: 15 November 1986, same old place! The highlight of the meeting should be a discussion by Hernan Poznansky and Danny Howell on the subject of active controls. Danny's background is in flight testing, airfoil selection, wind tunnel testing and servomechanisms.

BEAT THE HEAT: The little table below, taken from information in the August 84 Designee Newsletter (author: R. Caler) shows what temperature increase to expect on a colored fiberglass surface facing the sun at two ambient air temperatures.

COLOR	Temp. (80 deg ambient)	Temp (110 deg ambient)
White	128	163
Yellow	134	169
Lt. Blue/Aluminum	143	177
Purple/Silver	148	183
Red/Green	178	219
Brown	191	231
Black	198	237

PHYSICS OF FLUTTER

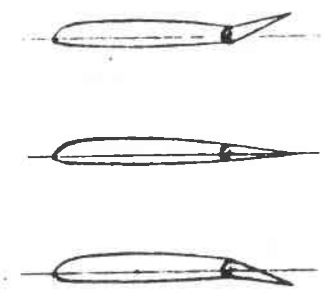
I. H. CULVER

There appears to be a general lack of understanding of flutter among aircraft designers, even many flutter analysts are mathematicians with little physical understanding of the subject. Therefore it would seem appropriate that more papers on the subject of the physics of flutter are in order. This paper is on cantilever configurations, no struts or wires.

We chose a simple first case like first mode symmetrical flapping with aileron symmetrical rotation, both ailerons up and both down. Before we get into the details let's make a simple single degree of freedom oscillatory system. Hang a weight from the rafters in your garage on a long string so the weight is close to the floor. This will make the frequency low enough so that you can excite it by pushing on the weight. Never mind that the stiffness for this system is provided by gravity rather than structural stiffness as in the case of flutter. You will find that pushing on the weight while it is moving will change the amplitude. Pushing with the motion will increase the amplitude of the motion while pushing against the motion will decrease the amplitude. In flutter these are called driving and damping forces. What do we learn from this? First, the excitation must have the same frequency (cycles per second (cps)) as the swinging motion, and pushing in the same direction as the motion increases the amplitude. The relation between the experiment and the flapping motion of a wing is that both are mass stiffness oscillatory systems. The only difference is that for the weight on the string the stiffness is due to gravity while for the wing the stiffness is due to structural stiffness and maybe some aerodynamic stiffness.

From the above it is apparent that if I push up on the wing tip while it is moving up, and down while it is moving down, the amplitude will increase (called excitation or driving). If I push down while the tip is moving up the result is damping. Where could the force come from that can push up while the wing is moving up, and down when the wing is moving down at the same frequency as the wing is flapping? First, we all know that we can raise and lower a wing with the ailerons, so if both ailerons went trailing edge down while the wing was moving up with sufficient amplitude and at the same frequency as the wing flapping we would have flutter.

If the ailerons are trailing edge heavy, the inertia forces of the ailerons will tend to deflect the aileron up when the wing flaps up and comes to a stop. So when the wing is in the full up position, just before it starts down, the aileron will be in the full up angular position, that is trailing edge up. And of course the trailing edge of the aileron will be down when the wing is down.



The non-oscillatory deflected position of the wing in flight is "0" →

At first look it appears that this just stiffens the wing in flapping. That is, the aerodynamic forces on the wing are in phase with the flapping motion so the aero forces only add to the structural flapping stiffness. This only raises the flapping frequency. However there are other effects.

The simplest effects first. Friction in the hinge pins and other parts phase-shifts the aileron motion to later; that is, the aileron is still moving up slightly when the wing is fully flapped up. So that when the wing is passing through "0" on the way down, the TE of the aileron is still up a little, pushing down on the wing while the wing is moving down.

The next effect is aerodynamic damping of the angular motion of the aileron. The wind is blowing by while the TE is moving. This causes a retarding or damping force similar to friction in the hinges and adds to the phase shift.

Next: aerodynamic lag of non-steady aerodynamics. There are two explanations worthy of consideration. Both are correct and non-conflicting. We are all familiar with the fact that as aspect ratio increases the induced drag decreases. This is due to the fact that at ∞ aspect ratio the up-wash ahead of the wing is equal to the down-wash aft of the wing and that the slope of the lift curve is 2π per radian of α. Now if we reason that deflecting the air down aft produces one half the lift and the other half comes from the upwash ahead of the wing, then if we suddenly change the angle of attack from 0 lift angle to some + angle, the air flowing off the TE will come off at the new angle of attack thereby creating one half the final lift. However the up-wash ahead of the wing takes time to develop since, as Newton said, a body in motion tends to stary in motion in a straight line unless acted upon by an external force. The air ahead of the wing is flowing straight at the wing when it is at 0 lift angle. When we suddenly change the angle to some + angle the pressure rises on the lower surface and drops on the upper surface. This causes an up pressure gradiant ahead of the wing, and, as Newton said, this pressure gradiant will cause the flow approaching the wing to start to curve up, and as it curves up it causes the pressure difference on the wing to rise, which strengthens the pressure signal telling the air to curve up more ahead of the wing until the up-wash ahead of the wing is equal to the down-wash aft of the wing.

Now for the classical explanation: A sudden change in angle of attack creates a circulation (or vortex) around the wing and a vortex of opposite sign and equal strength at the TE of the wing. The net result is that the vortex at the TE cancels one half the lift; however the TE vortex floats off downstream allowing the lift to grow to its full static value.

Now back to flutter. We have the wing flapping up and down. When the wing is flapped up the aileron TE is up, creating downlift. Now if the downlift due to the aileron TE being up lags in time, then when the wing passes through 0 on the way down, there will still be some downlift left. Result: some driving force. Remember, pushing in the direction of motion adds energy to the oscillatory system.

The above leaves several questions. First, how does the aileron natural frequency get close to the flapping frequency? The answer is that if the aileron natural frequency on the ground is below the wing flapping frequency, then as you increase speed the aerodynamic stiffening raises the natural frequency up to the flapping frequency or slightly above; the result is flutter.

A more detailed explanaation is, the aerodynamic center of the control surface is aft of the hinge line so as the dynamic pressure $q = \frac{\rho V^2}{2}$ rises, the stiffness around the hinge line increases, resulting in an increase in the natural frequency of the control surface.

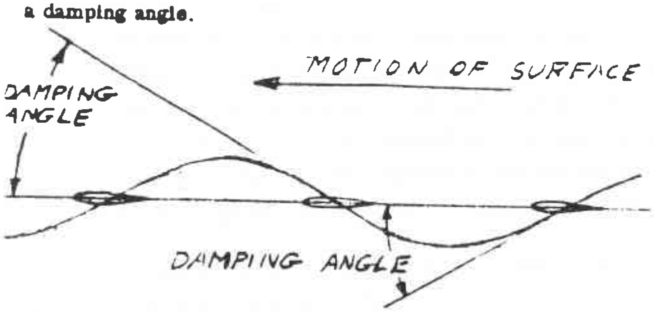
ω = FREQUENCY
I = MOMENT OF INERTIA AROUND H.L.
$K_{\alpha o}$ = 0 SPEED ANGULAR STIFFNESS
$K_{\alpha a}$ = AERODYNAMIC ANGULAR STIFFNESS
$\Sigma K_{\alpha} = K_{\alpha o} + K_{\alpha a}$

$$\omega_{A} = \left(\frac{\Sigma K_{\alpha}}{I}\right)^{\frac{1}{2}} \quad \text{RADIANS PER SEC. AVIS}$$

$$\omega_{H} = \frac{\left(\frac{\Sigma K_{\alpha}}{I}\right)^{\frac{1}{2}}}{2\pi} \quad \text{CYCLES PER SEC. HERTZ}$$

It should be noted that control cables can be loose due to thermal expansion differences between the airframe and the cables, or neglecting to keep the cables tight. Also, push-pull systems have some lost motion so that the frequency of the control surface is a function of amplitude so that up to some amplitude you can have flutter (limited amplitude flutter). Even though the amplitude is limited the damage causes to the system by limite cycle flutter can result in increased lost motion. It is not smart to rely on control systems to prevent flutter.

You should note that we had two degrees of motion available, one that moved normal to the airstream and one that changed incidence or angle relative to the airstream. If we had flapped the wing up and down only, while travelling through the air, we would have generated a damping angle.

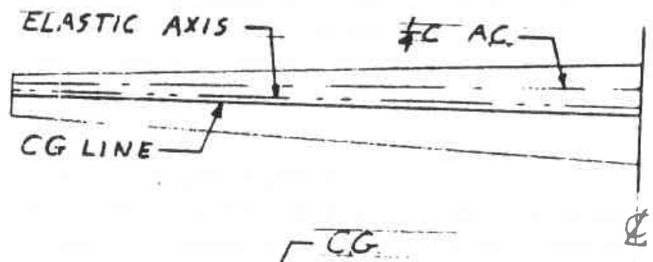


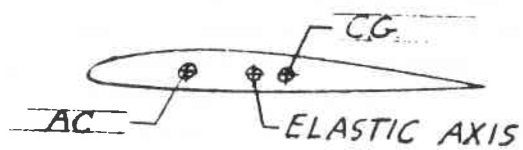
Suppose that for the same dynamic pressure $P = \frac{\rho V^2}{2}$ we go to altitude where the density ρ is lower so that we must go faster to get the same airspeed reading. Then for the same flapping amplitude and frequency we have less damping angle.

Now for the same flapping amplitude and the same control surface response the flutter problem is worse. That is, flutter will occur at a lower indicated airspeed at altitude. If one likes geometry, when the change in the angle of 0 lift line due to moving of the control surface is equal to the damping angle, then we have reached the 0 stability (neutral) flutter point.

We should next look at another 2-degree-of-motion flutter problem that is not likely to occur with today's designs. This is wing flapping and wing torsion. Assuming that we had mass-balanced the ailerons so that they would not contribute to flutter, it is still possible to have flutter. As we know, in the early days of aviation some machines used wing warping for aileron control. This would indicate that wing twisting gives similar effects to aileron motion.

Most wings have the C.G. of the airfoil aft of the elastic axis and both of these are aft of the aerodynamic center, which is approximately at the 1/4 chord. The elastic axis (or shear center) is the spanwise line that is defined by: if you push up or down on this line the wing will not twist.



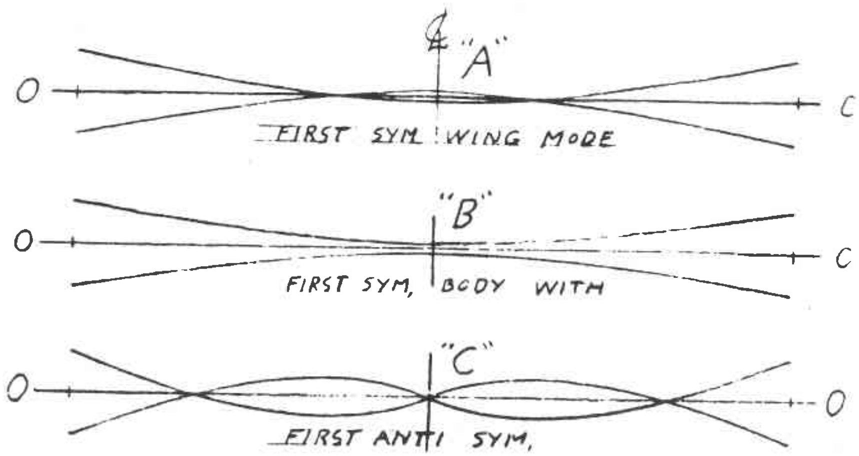


It is easy to see that this is similar to the aileron case with the C.G. aft of the hinge line. The elastic axis acts like a hinge line and the C.G. is aft, so when the wing is flapped up and comes to a stop it tends to deflect the TE up the same as for the aileron case. The difference is that for the wing torsion flapping case the wing torsion starts out at 0 speed at a considerably higher frequency than the flapping frequency. However the aerodynamic stiffness is negative in torsion instead of positive like the aileron case. That is, the aerodynamic center is ahead of the elastic axis so if we increase the dynamic pressure the torsional moment caused by deflectin the wing in torsion gives a negative stiffness that is opposite to the case of the aileron. So although we start with the torsior frequency way above the flapping frequency, increasing dynamic pressure can reduce the torsional frequency until it is just above the flapping frequency (flutter).

It should be noted that at flutter the wing is being stiffened in flapping by aero lift forces at the a.c. due to twisting. Also note that if the elastic axis is on or aft of the C.G. and aft of the a.c. then the wing will not flutter but it can diverge staticly (non-oscillatory divergence). Of course if the a.c., C G. and elastic axis are all together then nothing happens. Most modern surfaces are so stiff in torsion that there is low probability that wing torsion flapping flutter or static divergence is a problem within the useful speed range of sailplanes or light aircraft today.

TWITT has a letter from the co-author of NURFLÜGEL, Peter F. Selinger. Jan Scott sent to Peter, TWITT No. 4, which TWITT had mailed to VINTAGE SAILPLANES. TWITT will now have a contact in Stuttgart, Germany.

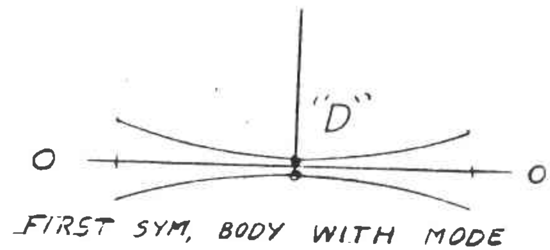
Some flapping mode shapes:

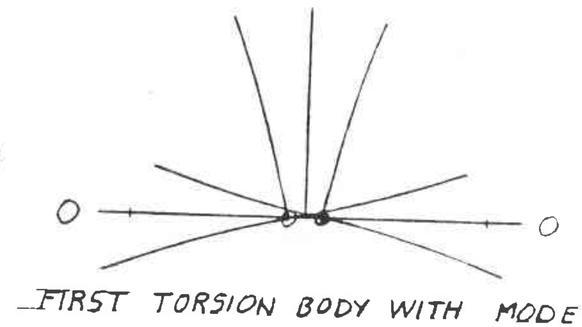


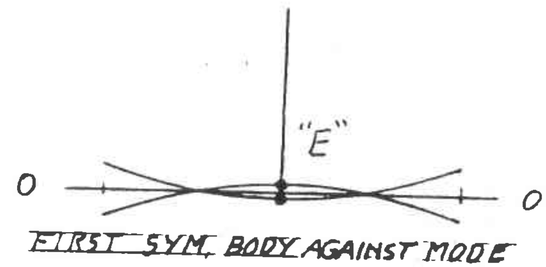
It should be noted that Case B symmetrical (body with) mode is highly improbable due to the requirement that the body bending first mode must be very low to match the wing frequency and the nose and tail must deflect a long way to make the C.G. of the airplane not plunge up and down while flapping (as required by Sir Isaac). Case B requires a very elastic fuselage.

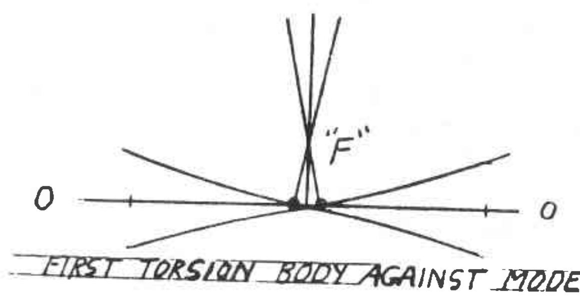
Second or higher modes of flapping are not considered since the chord lengths per cycle are so low that flutter could not occur. Also the admittance factor would be low.

TAIL MODES





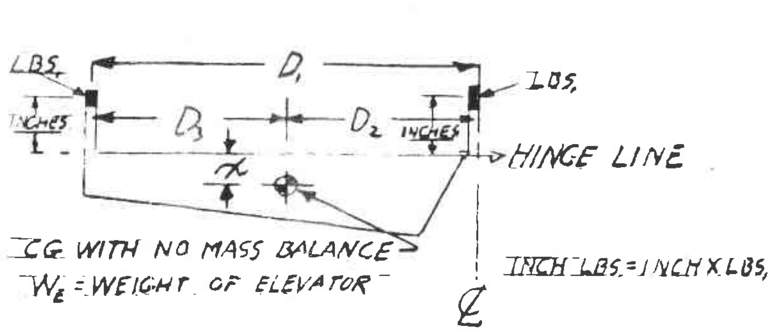




For tail surfaces all the body with and against modes are probable since the tail is generally small compared to the wing and the tail frequencies are close to the fuselage bending and torsional frequencies.

So the safest answer for tail surfaces that must fly a reasonably high speeds is to balance such that all first mod are balanced dynamically. This can be accomplished on the elevator by 3 mass balances, 1 on each tip and 1 in the cent

First estimate or measure the spanwise C. G. of one elevator.



The inch lbs. of balance at the tip is $= \frac{D_2}{D_1} \times (x \times W_E)$

and the inch lbs. at the center balance is $= \frac{D_3}{D_1} \times (x \times W_E)$

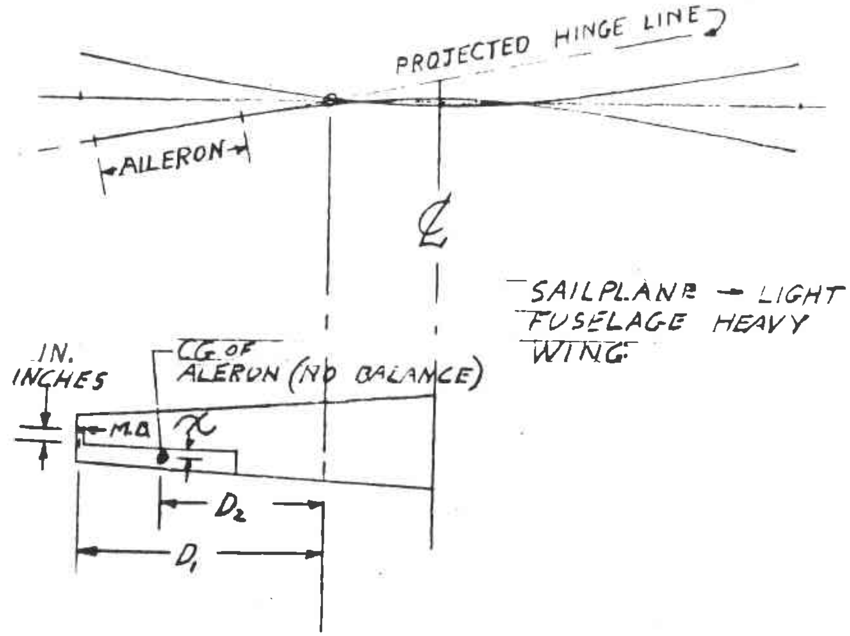
If the center balance weight is one weight for both elevators then of course the center balance is 2 TIMES THE ABOVE

Note inch lbs. of mass balance is the distance from the hingeline to the C. G. of the mass balance times the weight of the mass balance.

If possible use the same technique on the rudder; you will not have the doubling up in the center. It is of course OK to distribute the mass balance spanwise along the surface proportional to the chordwise unbalance distribution.

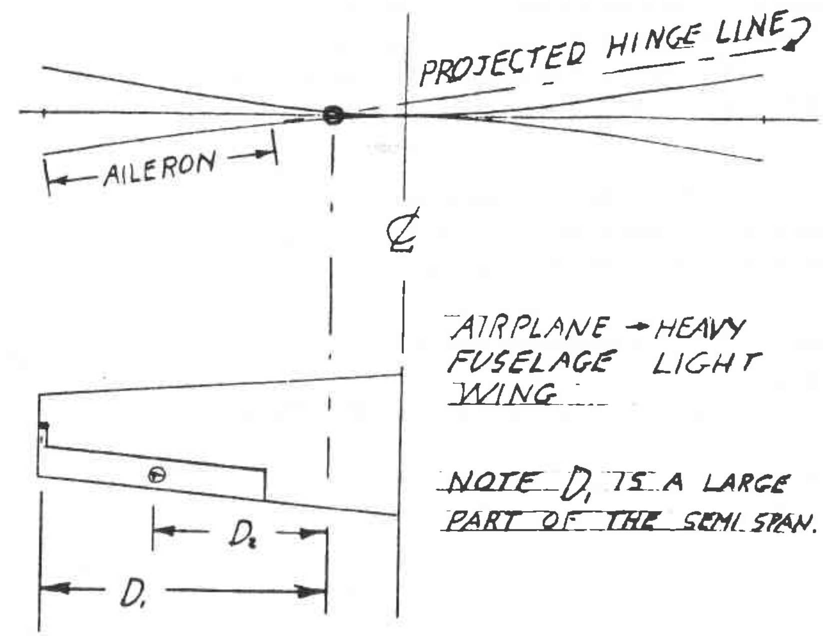
Aileron balance is a much simpler problem than tail surface balancing since ailerons do not generally run all the way in to the fuselage from the tip and the body motion even for Case B is small compared to the wing flapping excursion. Some high speed airplanes (Mach .8+) have balances only at the aileron tip. This is OK if the aileron torsional stiffness is high. (Natural frequency of the aileron in torsion including mass balance is considerably above the wing flapping frequency.)

Of course distributed mass balance proportional to the distributed mass unbalance is always OK. A full span aileron could have the same problem in the A mode as an elevator in the E mode if the wing were extremely heavy and the body light, but also the body mode in bending would have to be close in frequency to the wing flapping mode frequency. The likelihood of the above is essentially zero. So generally if the mass balance is sufficient to satisfy the following, the system will not flutter due to aileron unbalance.



THE MIN. M.B. (MASS BALANCE) WEIGHT IS

$$M.B. \times IN. = \frac{D_2}{D_1} \times x \times \left(\text{WEIGHT OF AILERON WITHOUT M.B.}\right)$$



Some words of warning about mass balancing.

First, mass balances must be attached to the surface sufficiently strong to stand buffeting due to stall at high speed and stiff enough so that the mass balance resonating against the surface being balanced produces a frequency considerably above the expected flapping frequency of the surface that the control surface is attached to.

Some notes on flight flutter testing:

My advice is to cross-exam the design to make sure that there is no possibility of flutter at 1.5 x VNE of the machine, then make a flight flutter demonstration, not a test. There is only one case where over balancing of control surfaces will cause flutter and that is where the over balance causes an apparent unbalance.

Look at case E: a tail with full static balance on the elevator all at the tip with a light aft fuselage so the fuselage motion is large. The inboard largest part of the elevator sees the mass balance at the tip as if it were on the trailing edge of the elevator due to its opposite motion. (Explanation of above: Full static balance means that you balance around the hinge line and in this example put all the balance at the tips.) Full static balance does not mean that you do not have a flutter problem.

Afterthoughts:

There are many effects not covered in this overview of the physics of flutter:

1 - The effects of structural damping.

2 - Transonic effects (imbedded supersonic inclosures with shock down to subsonic) causing non-linear aerodynamics.

3 - Body pitching coupling with flapping & torsion and/or control surface motion.

4 - Separated flow and stall flutter.

These effects are not generally problems with sailplanes and light aircraft.

Irv Culver

tions of Swept Wings of Various Taper Ratios. B.C. Wallner. NACA RM L8A26, July 1948.

28. A Simple Approximate Method for Obtaining Spanwise Lift Distribution over Swept Wings. F.W. Diederich. RM L7I07, May 1948.

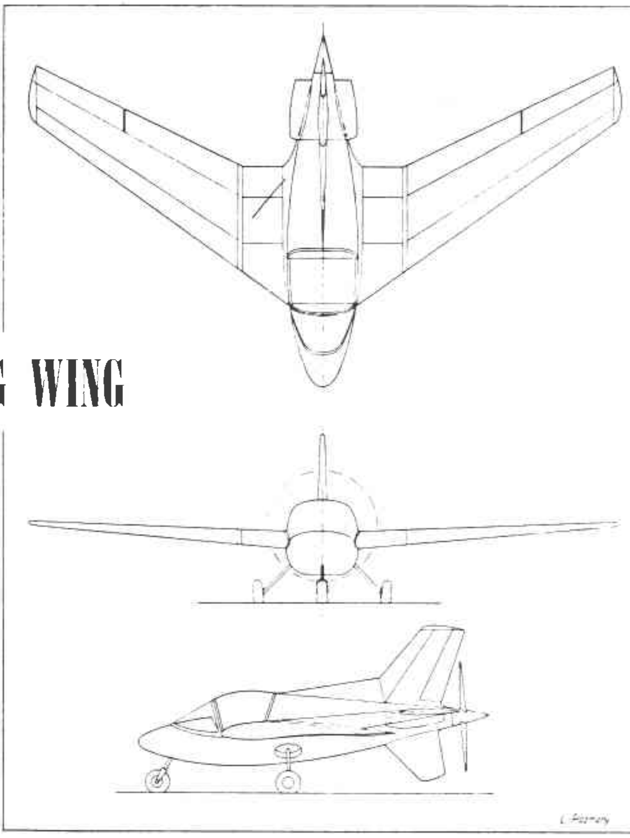
29. Comparison Between Measured and Theoretical Span Loading on a Moderately Swept-Forward and Moderately Swept Back Semispan Wings. Mendelsohn and Brewer. NACA TN 1351, 1947.

HIGH PERFORMANCE SAILPLANE DATA, compiled by Bob Fronius

Type	Min sink	at (speed)	span	AR	L/D
HP-12E	1.8	47	54.6	21.6	39
HP-15	1.6	45	49.2	33	45
HP-19	1.6	40	49.2	21.4	42
Lamson	1.5	50	65.6	23	43
Mescalero			72	36	44
White Knight	1.7		49.2	18.1	34
Diamant 19	1.6		62.5	24	42
VHP-1	1.6	49	57	28.5	40
RHJ-8	1.9	50	53	23.2	39
DG-202/17	1.7		55.8	27.3	45.5
DG-400/17	1.76		55.8	27.3	45
Elfe	1.2	poor in weak air	52.5	20.4	40
BS-1	1.78	50	59	23	44
Libelle	1.8	47	49.2	23	35-39
Kestrel	1.8	51	55.7	25	43
304	1.34	57	49.2	22.8	43
604	1.64	45	72.17	29.8	49
Jantar	1.5	40	67.2	29.2	
48					
Lark IS 32	1.67		65.6	27.25	46
Cirrus	1.64	46	58.2	25	44
Nimbus II	1.6	47	66.6	28.6	49
Nimbus 3	1.71		75.13	32.3	55
ASW-12	1.6	53	60	26	47
ASW-15	1.8	42	49.2	20.45	
38					
ASW-17	1.64		66	27	48.5
ASW-22	1.35		78.9	37	57

HAVE YOU SEEN THESE WINGS? These pictures, from Al Backstrom's collection, show two low AR flying wing designs. Bob Fronius says that the one with endplates looks like one he saw at Santa Ana in 40-41. It also resembles a design by one Horten or Horton (not related to the German brothers). The other looks like the Lanier Paraplane of the same era. The truth is that we have no idea where these beasts come from. If you can enlighten us, call Bob with the information!

Drawing by
Geo. Collinge



AN EXPERIMENTAL PUSHER FLYING WING

by Ladislao Pazmany, EAA 2431

The "flying wing" idea is as old as aviation itself Even the early experimenters realized the apparent advantages of the configuration but lacked the knowledge and materials necessary for success. Raoul Hoffman's All-Wing design of 1934 illustrated in the December issue of the EXPERIMENTER was one adaptation of the idea to the lightplane field. In recent years we have had such efforts as the Horten and Fauvel gliders, the Northrop series, Backstrom's "Plank" and others.

In this article I'd like to discuss the possible advantages of this design solution and draw a comparison between it and the conventional approach. The accompanying three-view drawing and the artist's impression by George Collinge which appeared on the cover of the January issue of SPORT AVIATION outlines one possible solution of the flying wing application to light aircraft.

There are advantages to this design which can be demonstrated easily without too much calculation, such as reduced weight and reduced number of parts to manufacture. The weight reduction will result in a general performance improvement, and as an example the increase in the V-max will be calculated. The comparison will be made between a conventional two-place, side by side airplane and a pusher flying wing (P.F.W.), both using the same wing area and the same power.

Due to the elimination of the horizontal tail and part of the fuselage, a 50 lb. weight saving can be estimated. Thus our first comparison table will appear as follows:

	Conventional	P. F. W.
Gross weight	1200 lbs.	1150 lbs.
Wing area	114 sq. ft.	114 sq. ft.
Engine	85 hp	85 hp
Airfoil	NACA 63_2615	NACA 63_2615

Calculating the parasitic drag coefficient for each design reveals an interesting comparison:

FUSELAGE

Tabulating the equivalent flat plate area as follows:

	Conventional	P. F. W.
Skin friction & irregularities	0.315 sq. ft.	0.190 sq. ft.
Canopy	0.114 sq. ft.	0.114 sq. ft.
Engine installation	0.730 sq. ft.	0.730 sq. ft.
Total area	1.159 sq. ft.	1.034 sq. ft.

The drag coefficient for the fuselage will be (based on wing surface):

Conventional — $C_{Df} = \frac{1.159}{114} = .0102$

P. F. W. — — — $C_{Df} = \frac{1.034}{114} = .0090$

TAIL SURFACES

On the conventional airplane the following values will be found:

Horizontal Tail —	S_h =	18.0 sq. ft.
Vertical tail —	S_v =	9.7 sq. ft.
Total tail area —		27.7 sq. ft.

In previous calculations the tail drag coefficient based on the wing area was found to be:

$$C_{Dt} = 0.0024$$

Due to the elimination of the horizontal tail and a slight increase in the vertical tail area, the P. F. W. tail area is estimated to be 15 sq. ft. Then the drag coefficient can be calculated thusly:

$$C_{Dt} = 0.0024 \times \frac{15}{27.7} = 0.0013$$

WING

Due to the possibility of obtaining a complete laminar flow over the entire wing because of the absence of the turbulent propeller slipstream, the following considerations can be made:

On the conventional airplane, 30% of the wing area is subjected to turbulent flow, while on the P. F. W. the 100% wing area can be considered as laminar.

On page 29 of NACA Report No. 824, Fig. 35, it is stated that the effect of the propeller slipstream turbulence increases the section drag coefficient by 50%. The values shown are for a 66(2 × 15) — 018 airfoil, and the C_{do} is increased from .0040 for the undisturbed airfoil to C_{do} = .0060 (mean value) for the disturbed airfoil. Then, on page 169 of the same report, the following is given relative to the airfoil considered in this comparison (NACA 63_2615):

C_{do} = .0103 @ RN = 6.000.000 & standard roughness

and for the disturbed airfoil we can calculate:

$$C_{do} = .0103 + \frac{(.0103 \times 50)}{100} = .0154$$

Then the wing parasitic drag coefficient for the conventional airplane will be:

$$.70 \times .0103 = .0079$$
$$.30 \times .0154 = .0046$$
$$C_{do} = .0125$$

Thus the total parasitic drag coefficients for each airplane can be compared as follows:

	Conventional	P.F.W.
Fuselage	.0102	.0090
Tail surfaces	.0024	.0013
Wing	.0125	.0103
Total Parasitic Drag	.0251	.0206

Assuming that both airplanes will have the same tapered wing with an aspect ratio of 7, then the induced drag coefficient can be calculated. The wing and fuselage efficiency is found to be: e = .83. Then:

$$C_{Di} = \frac{CL^2}{r \times e \times AR} = \frac{CL^2}{r \times .83 \times 7} = 0.0548\ C_L^2$$

The total drag coefficient for each airplane will be:

Conventional - $C_D = .0251 + .0548\ C_L^2$

P. F. W. - - - $C_D = .0206 + .0548\ C_L^2$

The maximum speeds can be estimated for each airplane:

Conventional - V_{max} = 147 mph

P. F. W. - - - V_{max} = 157 mph

The lift coefficient for these speeds is determined by the following formula:

$$C_L = \frac{W/S}{0.00256\ V^2} \quad \text{where } W/S = \text{wing loading}$$

Calculation of the lift coefficient is as follows:

$$\text{Conventional - } C_L = \frac{1200/114}{.00256 \times 147^2} = .187$$

$$\text{P. F. W. - - - - } C_L = \frac{1150/114}{.00256 \times 157^2} = .159$$

The value of the total drag coefficient for each airplane will be:

Conventional - $C_D = .0251 + .0548 \times .187^2 = .0270$

P. F. W. - - - - $C_D = .0206 + .0548 \times .159^2 = .0219$

The value of the drag can be calculated with the following formula:

$$D = \frac{W}{C_L / C_D}$$

$$\text{Conventional - } D = \frac{1200}{.187/.0270} = 173 \text{ lbs.}$$

$$\text{P. F. W. - - - - } D = \frac{1150}{.159/.0219} = 158 \text{ lbs.}$$

The propeller efficiency for a tractor installation can be estimated as: η = .80, while for the pusher type it will be slightly smaller say: η = .78. Then the maximum speed can be determined with the following formula:

$$V_{max} = \frac{375 \times HPmax \times \eta}{D}$$

$$\text{Conventional } V_{max} = \frac{375 \times 85 \times .80}{173} = 148 \text{ mph}$$

$$\text{P. F. W. - - - } V_{max} = \frac{375 \times 85 \times .78}{158} = 156.5 \text{ mph}$$

The obtained values check out with the previously estimated, so the improvement in maximum speed is then determined:

$$V_{max} = 157 - 148 = 9 \text{ mph}$$

and in percentage:

$$V_{max}\ (\%) = \frac{9}{148} \times 100 = 6.1\%$$

The improvement obtained is not fantastic, but the airplane designer knows that any performance improvement in modern airplanes is built up through the summation of many small contributing factors. The pusher flying wing configuration would seem to offer many advantages which will result in improved performance. Certainly it merits close study and further experimentation.

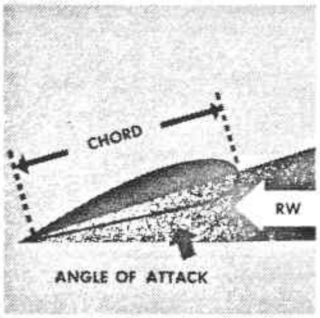
It must be emphasized that this design proposal is presented as an **idea** and needs further evaluation. The question has been raised regarding the problem of weight and balance on an aircraft of this type, since the CG travel of tailess aircraft is very limited. Possibly this can be improved by moving the passengers nearer to the CG. Additional study and analysis of this and other problems would be necessary before the final configuration could be arrived at, but since airplanes of this type have been built and flown successfully, it should be possible to evolve a suitable solution.

SPECIFICATIONS

Wingspan	**336 in.**
Length	**200 in.**
Height	**96 in.**
Width (wing folded)	**96 in.**
Weight	**1150 lb.**
Wing Area	**114 sq. ft.**
Engine . . . Continental	**C-85 hp**

PERFORMANCE

Maximum Speed	**156 mph**
Cruising Speed	**138 mph**
Stalling Speed	**50 mph**
Rate of Climb @ S. L.	**900 ft. /min.**
Service Ceiling	**16,500 ft.**
Range	**560 miles**



ANGLE of attack— The acute angle between the chord of an airfoil (wing) and the relative wind. (Note that the relative wind is not always parallel to the longitudinal axis.)

REFERENCES   Bruce Carmichael Collection

1. An Interim Report on the Stability and Control of Tailless Airplanes. Stability Research Division. NACA TR 796, 1944.

2. Preliminary Wind-Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept-Back Wings. W. Letko and A. Goodman. NACA TN 1046, April 1946

3. Effects of Sweepback and Aspect Ratio on Longitudinal Stability Characteristics of Wings at Low Speed. Shortal and Maggin. NACA TN 1093, July 1946.

4. Approximate Relations and Charts for Low Speed Stability Derivatives for Swept Wings. Tell and Queijo. NACA TN 1581, May 1948.

5. Estimate of the Effectiveness of Flap Type Controls on Sweptback Wings. Lawry and Schneiter. NACA TN 1674, August 1948.

6. Investigation at Low Speeds of the Effect of Aspect Ratio and Sweep on Static and Yawing Stability Derivatives of Untapered Wings. Goodman and Brewer. NACA TN 1672, August 1948.

7. A Comparison of the Lateral Motion Calculated for Tailless and Conventional Airplanes. C.W. Harper and A.L. Jones. NACA TN 1154, February 1947.

8. Notes on the Stability and Control of Tailless Airplanes. Robert T. Jones. NACA TN 837, December 1941.

9. Recent Tests of Tailless Airplanes. A. Lippisch. NACA TM 564, May 1930.

10. Chief Characteristics and Advantages of Tailless Airplanes. A.D. de Lajarte. NACA TM 794, May 1936.

11. The Dreieck I Tailless Airplane, a Low-Wing Cantilever Monoplane. NACA AC 159, March 1932.

12. Description of the Westland Hill Pterodactyl Mark IV. NACA MP 27, August 1931.

13. Determination of the Stability and Control Characteristics of a Tailless All-Wing Airplane Model with Sweepback in the Langley Free-Flight Tunnel. J.P. Campbell and C.L. Seacord Jr. NACA ACR L5A13, February 1946.

14. Determination of the Stability and Control Characteristics of a Straight-Wing Tailless Fighter-Airplane