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ENGINEERING REPORT

RAPPORT D'INGÉNIERIE

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ENGINEERING REPORT
RAPPORT D'INGÉNIERIE

LP096/2009

Bush Caddy Analysis

Bush Caddy, [redacted]

Occurrence Date: 28-Jun-09

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1.0 INTRODUCTION

1.1 Description of Occurrence. While in flight approximately ten minutes after take-off, the amateur kit-built Bush Cadi (also spelled Bush Caddy) aircraft encountered turbulence and experienced an upset. Following recovery from the upset, the pilot observed that the right wingtip appeared to be missing. Inspection of the aircraft after landing revealed that the wingtip was not missing, but rather that the right wing had buckled at the strut attachment location and was bent upwards.

1.2 Engineering Services Requested. The damaged wing was submitted for engineering examination and analysis.

2.0 EXAMINATION

2.1 Description of Damage. Figure 1 shows the right wing as received. The upper caps of the front and rear spars were buckled just inboard of the strut attachment points. The buckles were located at lightening holes. Close visual examination of the lower spar caps and skins did not reveal any cracking.

2.2 Wing Loads and Stresses. The calculations in Appendix A estimated the spanwise shear force, bending moment, and end load in the wing when the aircraft was in steady level flight at its maximum weight of 2200 pounds. The highest stressed location in the wing was found to be at the point where the wing strut attaches, which also corresponds to the location of the failure in the present occurrence. The calculations in Appendix B show a stress analysis at the failure location. Although the wing spars in the occurrence wing failed in compression buckling, a theoretical buckling analysis with available techniques would not provide a sufficiently accurate result due to the complexity of the local geometry and loading. Therefore, rather than estimating the stress at which the present buckling failure occurred, a simpler stress analysis was conducted to estimate the stress at which the present structure would have failed in a tensile overstress mode of failure due to bending. This analysis determined that in steady level 1.0 g[1] flight at the maximum weight of 2200 pounds, the bending stress in the front spar was about one third of the value at which tensile overstress bending failure is experienced. As a first approximation, this suggests that the wing would fail in tensile overstress at roughly 3.0 g.

2.3 Kit Design. It was reported that after the company which originally designed and manufactured the kits with this particular wing design was sold to a new owner, the new owner recognized the weakness in the design and strengthened it. The new design of this wing uses a thicker front spar (0.063 versus 0.050 inches), thicker wing skins (0.025 versus 0.020 inches), and has four extra wing stringers. All the wing stringers are now continuous along the span rather than stopping at each rib and the rivet spacing is reduced. The older design is still in use on some airplanes constructed before the design change.

[1] 1.0 g equals one times the force of gravity

3.0 DISCUSSION

3.1 Margin of Safety for Tensile Overstress Failure. The analysis found that the wing of the occurrence aircraft would fail in tensile overstress at very roughly 3.0 g when the aircraft is at its maximum weight of 2200 pounds. As a point of comparison, aircraft certified in the Normal Category must be designed to withstand a manoeuvring load factor of 3.8 plus have a safety factor of 1.5 for a total of 5.7 g.[2] As a further comparison, aircraft designed to the Advanced Ultralight standard must be designed to withstand a manoeuvring load factor of 4.0 plus have a safety factor of 1.5 for a total of 6.0 g[3]. Therefore, for the case of bending failure by tensile overstress, the wing of the occurrence aircraft had about half the strength of aircraft designed to Normal Category or Advanced Ultralight standards.

3.2 Buckling Failure. In the present occurrence, the mode of failure was buckling. There were no indications of cracking or necking to suggest the tension side of the spar had approached tensile overstress failure limits. In a thin light structure such as this, it is not unusual for buckling to occur before tensile overstress limits are reached. Also contributing to buckling is the compressive end load introduced into the wing by the spar, so it is not unusual that the buckling failure should have occurred inboard of the strut attachment. It is considered that although the wing would have failed at roughly 3.0 g in tensile overstress, it would most probably have failed at less than 3.0 g in buckling.

4.0 CONCLUSIONS

4.1 The buckling failure location just inboard of the wing strut attachment point was consistent with expectations, this being the location with the most severe combination of bending moment and compressive end load.

4.2 Theoretical calculations and information from the kit manufacturer indicated that the occurrence wing would fail at less than 3.0 g, roughly half the load factor of a wing designed to Normal Category or Advanced Ultralight standards.

4.3 It was reported by the current kit manufacturer that wings like the one in the occurrence are no longer being sold, and that the design has been strengthened. However, the older design is still in use on some airplanes constructed before the design change.

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[2] Transport Canada Airworthiness Manual Section 523

[3] Light Aircraft Manufacturer's Association of Canada, Design Standards for Advanced Ultralight Aeroplanes

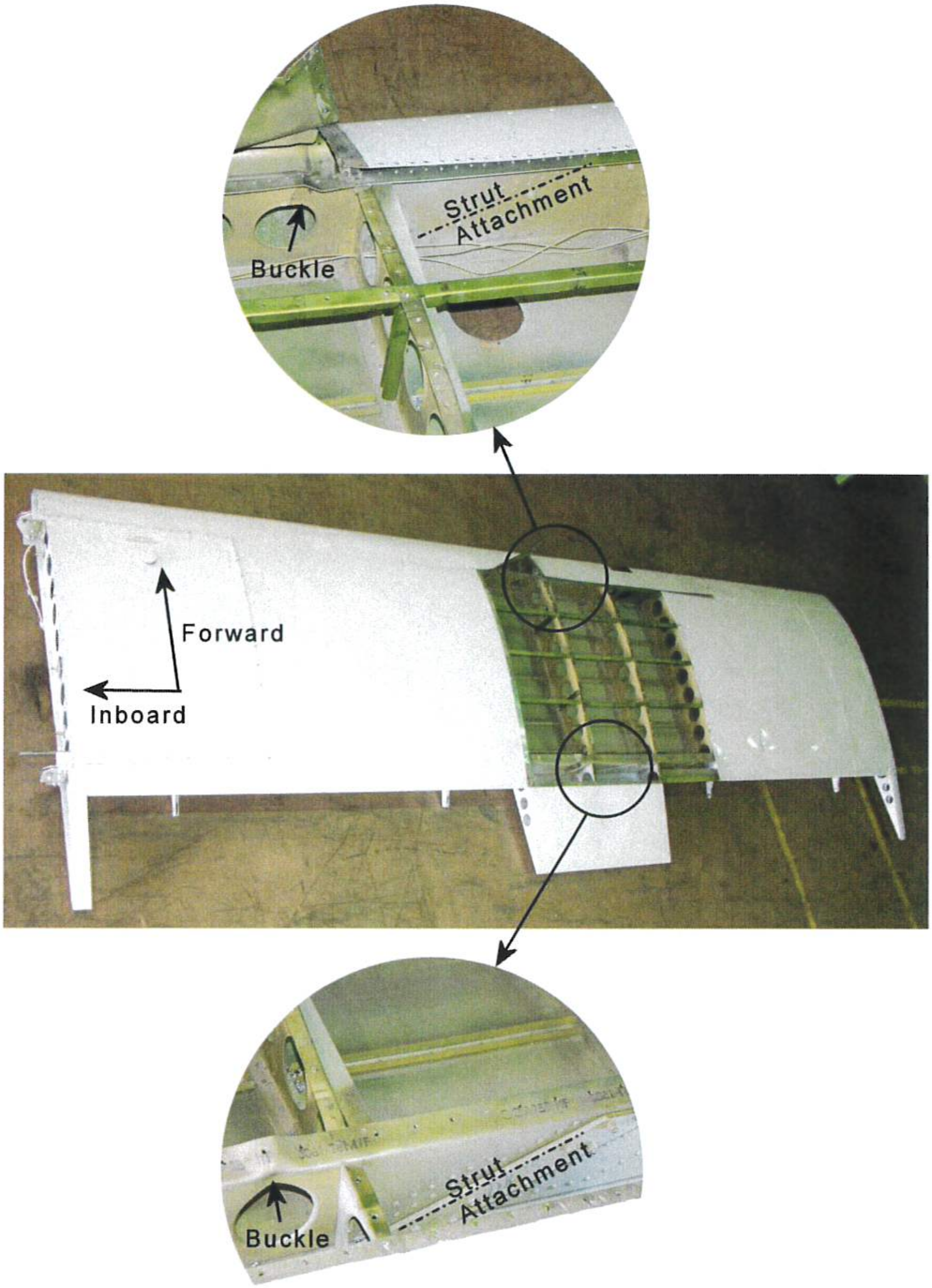

Figure 1: Right wing as received. The tops of both the front and rear spars were buckled just inboard of the strut attachment points. Strut attachment is indicated by dotted lines.

Appendix A: Calculation of Spanwise Loading

1.0 Aim. The aim of these calculations is to determine spanwise loading, shear force, bending moment, and end load for the wing.

2.0 Spanwise Loading. The planform of this wing is rectangular. Therefore, the spanwise airload distribution was assumed to be the average between rectangular loading and elliptical loading. In addition to the airloads which force the wing upwards, the inertial relief caused by the weight of the wing forces the wing downwards.

Rectangular Loading. To simplify the analysis, the comparatively small airload increment on the wing needed to compensate for the downwards load on the tail is neglected. Washout is also neglected. The maximum take-off weight (MTOW) is 2200 lbs, so half of the lift (1100 lbs) is assumed to be provided by each wing. If 1100 lbs is distributed rectangularly along the 184 inch span of the wing, the wing load is 1100/184=5.978 lbs/in-span as shown in the sketch below.

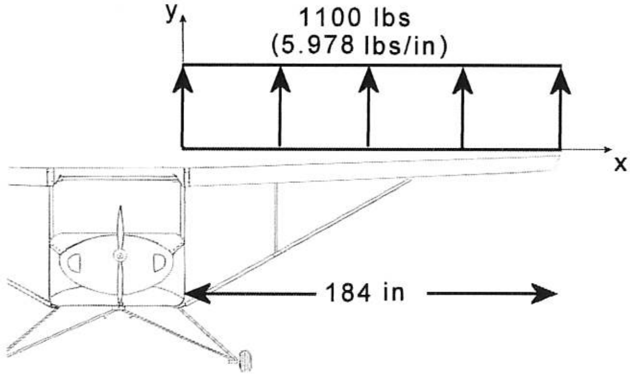

Figure A-1: Rectangular air load on wing.

The equation for the spanwise rectangular airload distribution is:

$$y = 5.978\, lbs\,/\,in$$

Elliptical Loading. The elliptical airload across one wing equals a quarter of the area of the ellipse as shown in the sketch below.

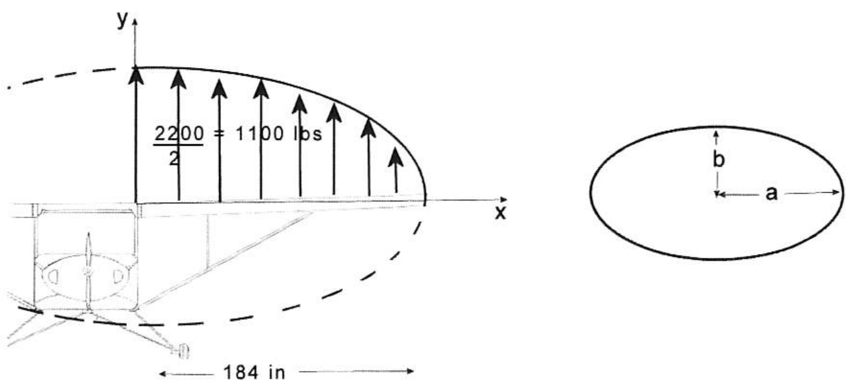

Figure A-2: Elliptical air load on wing.

The area of an ellipse (A) is given by the equation A = π(ab). The major radius of the ellipse (a) is 184 inches. The 1100 pound load is located within the quarter-area of the ellipse. Therefore, the minor radius (b) is calculated using this area equation as follows:

$$b = \frac{A}{\pi a} = \frac{(1100\,lbs \times 4)}{\pi(184\,in)} = 7.612\,lbs/in$$

The equation for the spanwise elliptical air load distribution is:

$$\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1$$

or

$$y = b\sqrt{1 - \frac{x^2}{a^2}} = 7.612\sqrt{1 - \frac{x^2}{(184)^2}}\ lbs/in$$

<u>Average Between Rectangular and Elliptical Loading</u>.

The average between the rectangular and elliptical loading is:

$$y = 0.5\left[5.978 + 7.612\sqrt{1 - \frac{x^2}{184^2}}\right] lbs/in$$

<u>Inertial Relief</u>.

The weight of the wing acts in the opposite direction as the lift, so must be subtracted to get the net loading. A wing weight of 111 pounds was estimated from weighing the wreckage.

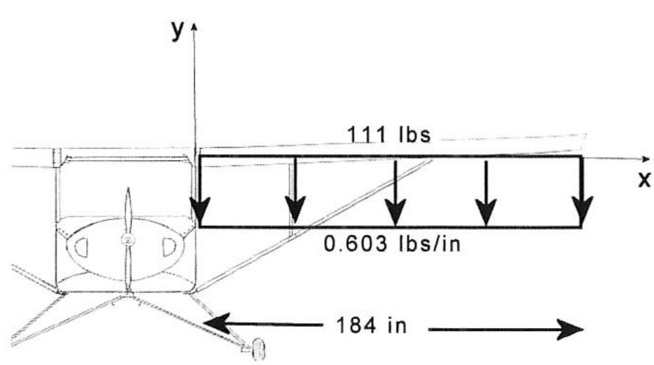

Figure A-3: Inertial relief on wing.

Net Spanwise Wing Load.

The inertial relief is now subtracted from the air load:

$$y = 0.5\left[5.978 + 7.612\sqrt{1 - \frac{x^2}{184^2}}\right] - 0.603\, lbs / in$$

When this equation is integrated across the 184-inch span, it results in a net upwards load of 989 pounds on the wing (ie. 1100 lbs air load acting up minus 111 lbs inertial relief acting down).

3.0 External Reactions. The external reactions at the wing root and strut are now calculated. The moment about the wing root is integrated across the span and then divided by the moment arm to the wing strut. This process determines that the vertical load at the strut is 905 pounds. Subtracting this from the upwards load on the wing of 989 pounds results in a load of 84 pounds at the wing root as shown in the following sketch.

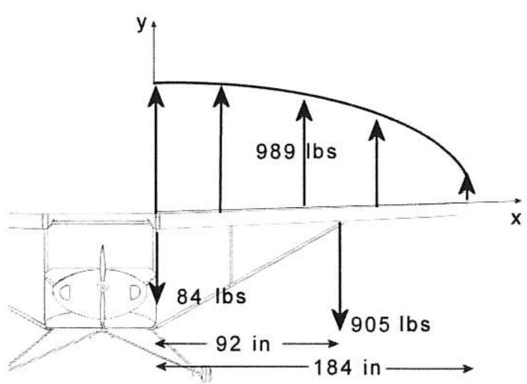

Figure A-4: External reactions on wing.

4.0 Shear and Bending Moment Diagrams.

The above loads are integrated once to determine the shear force and again to determine the bending moment. The results are as follows.

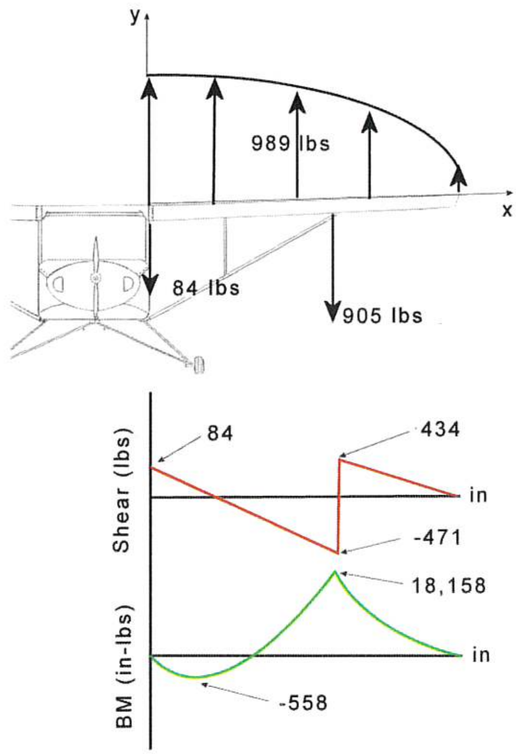

Figure A-5: Shear and bending moment diagrams.

5.0 Wing Strut Load. The above calculations only show the vertical component of the wing strut load. By taking vector components, the longitudinal load in the strut is now calculated as well as the compressive end load in the wing caused by the horizontal component of the wing strut load. As shown in the sketch below, the compressive end load in the wing is quite significant.

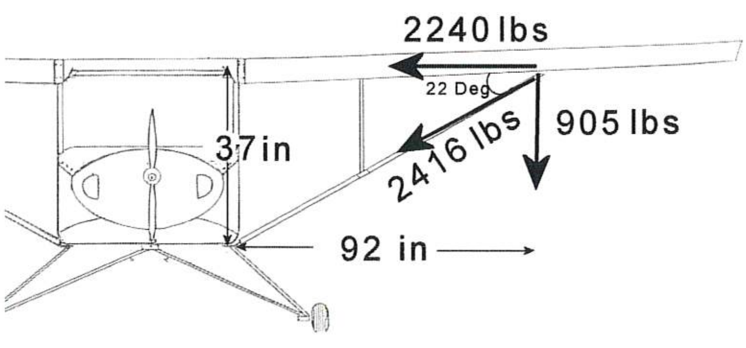

Figure A-6: Strut load and wing end load.

Appendix B: Calculation of Front Spar Stresses

1.0 Dimensions and Materials.

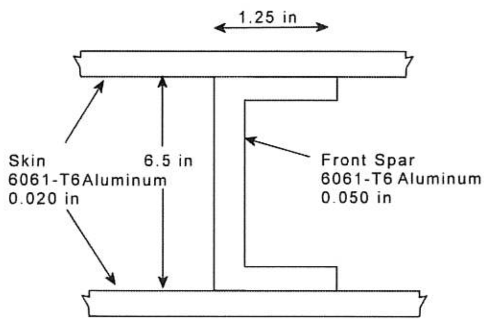

Figure B-1: Dimensions and materials of front spar.

2.0 Area Moment of Inertia.

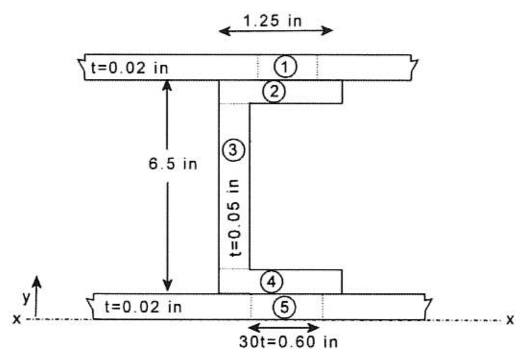

Figure B-2: Dimensions for area moment of inertia calculation.

Table B-1: Area moment of inertia calculation

Item	A	y	Ay	Ay^2	$bd^3/12$
1	0.012	6.53	0.078	0.512	------
2	0.0625	6.495	0.406	2.637	------
3	0.32	3.27	1.046	3.422	1.092
4	0.0625	0.045	0.003	------	------
5	0.012	0.01	------	------	------
Σ	0.469		1.533	6.571	1.092

$$I_{xx} = 6.571 + 1.092 = 7.663\,in^4 \quad \textit{Area Moment of Inertia about x axis}$$

$$\bar{y} = \frac{1.533}{0.469} = 3.269\,in \quad \textit{y coordinate of neutral axis}$$

$$I_{NA} = 7.633 - (0.469)(3.269)^2 = 2.621\,in^4 \quad \textit{Area Moment of Inertia about Neutral Axis}$$

3.0 Calculation of Stress.

The spars in the occurrence wing failed in compression buckling. Due to the complexity of the geometry and loading, a theoretical buckling analysis with available techniques would not provide a sufficiently accurate result. Therefore, rather than estimating the stress at which the present buckling failure occurred, a simpler stress analysis will be conducted to estimate the stress at which the present structure would have failed in a tensile overstress mode of failure. The end load is ignored to examine the worst case tensile stresses just outboard of the strut attach point. The bending shape factor is assumed to be 1.0 so is neglected.

From Appendix A, for an aircraft weighing 2200 lbs in steady level flight, the maximum bending moment in wing (M) is 18,158 in-lbs.

As a rough approximation assume 60% of this bending moment is reacted by the front spar (0.60 x 18,158 = 10,895 in-lbs).

The distance from the neutral axis to the most extreme fibre (c) is 3.25 in.

Therefore, the maximum bending stress (σ) on the front spar in steady level flight is:

$$\sigma = \frac{Mc}{I_{NA}} = \frac{(10{,}895\, in - lbs)(3.25\, in)}{2.621\, in^4} = 13{,}510\, psi$$

The following are the material allowables for 6061-T6[4]:

F_{TU} = 43,000 psi (Ultimate tensile strength)
F_{TY} = 38,000 psi (Yield Strength)

In steady level flight at its maximum weight of 2200 lbs, the maximum stress in the wing is roughly one third of the value required to cause a tensile overstress failure.

Therefore, as a rough first approximation, the wing would fail in tensile overstress at approximately 3.0 g.

[4] Military Standardization Handbook, Metallic Materials for Aerospace Vehicle Structures (Mil Handbook 5D)