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Evaluation of the Wright 1902 Glider Using Full-Scale Wind-Tunnel Data

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A reproduction of the Wright 1902 glider was wind-tunnel tested at the NASA Langley Research Center Full-Scale Tunnel through a range of conditions similar to those experienced 102 years ago. Data were collected over a range of angle of attack from −4 to 20 deg, and a range of sideslip angles from −20 to 20 deg. Both the canard and wing warp controls were actuated to determine control power, and a canard-only test was performed to augment estimates of pitch control power. An optimal lift/drag of 7.4:1 in ground effect was determined, comparing favorably with the Wrights' observations. This aircraft displayed levels of longitudinal and lateral control power that allowed, for the first time, control of an aircraft without reliance on weight shift. Because of its low pitch inertia and damping, the aircraft is sensitive in pitch and requires constant pilot attention to maintain an attitude. Laterally, the aircraft shows adequate roll rate and turn coordination, although a moderate sideslip will develop in turning maneuvers due to the lack of vertical surface area. Validation of the handling characteristics was aided through the use of a six-degree-of-freedom, commercially available flight simulator.

Nomenclature

Æ	=	aspect ratio
B_{C_X}	=	bias error for X coefficient
C_D	=	drag coefficient
C_L	=	lift coefficient
C_{L0}	=	aircraft lift coefficient intercept
$C_{L\alpha}$	=	aircraft lift coefficient slope
C_l	=	roll coefficient
C_{lp}	=	roll coefficient/roll rate
$C_{l\beta}$	=	roll coefficient/sideslip angle
$C_{l\delta w}$	=	roll coefficient/warp deflection
C_m	=	moment coefficient
C_{mq}	=	pitch coefficient/pitch rate
$C_{m\alpha}$	=	aircraft pitching moment coefficient slope
C_n	=	yaw coefficient
C_{n_β}	=	yaw coefficient/sideslip angle
C_{Y_β}	=	side force coefficient/sideslip angle
I_{yy}	=	pitching moment of inertia
L/D	=	lift to drag ratio
n_z	=	vertical load factor
P_{C_X}	=	precision error for X coefficient
Q	=	dynamic pressure
Re	=	Reynolds number
U_{C_X}	=	uncertainty for X coefficient
α	=	angle of attack
β	=	sideslip
δ_c	=	canard deflection
δ_w	=	warp deflection
θ	=	aircraft pitch angle

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Introduction

UPON returning from Kitty Hawk after the 1901 glider flights, the Wright brothers immediately began sorting through conflicting results. This marked the beginning of an intense period of scientific investigation. The 1901 glider was marginally controllable, and its lifting capacity for a given angle of attack was much less than predicted. The Wrights brought into question all existing theories on lift and realized they would need to develop new theories to achieve successful, well-controlled glides. Ironically, it was the failure of the 1901 glider that prompted the Wrights to explore new theories and designs that would allow them to fly a powered machine by 1903.

Fundamental research in airfoil lift and drag was conducted by the Wrights shortly after their return from Kitty Hawk in late 1901. The Wrights used a shop-built wind tunnel that cleverly measured lift and drag coefficients using two aerodynamic balances. One balance was designed to measure the lift/drag (L/D) ratio, and the other balance measured lift, so that the drag could be determined from the available data. This was not the first wind tunnel in use,[1] but it was the first one to have accurately measured both lift and drag. Over 200 airfoil shapes were examined in the tunnel, resulting in a few shapes that would continue to be used in wing and propeller designs of their subsequent aircraft.

The 1902 glider design embodies features that were found in the wind tunnel to improve lift and performance, including airfoil shape, the position of maximum camber, and increased wing aspect ratio. The low Reynolds number found in the 30-mph (13.4-m/s) wind tunnel with airfoil sections measuring a few square inches in area resulted in the Wrights selecting thin airfoils for their designs. It was not until testing by Prandtl and others in Germany in the latter

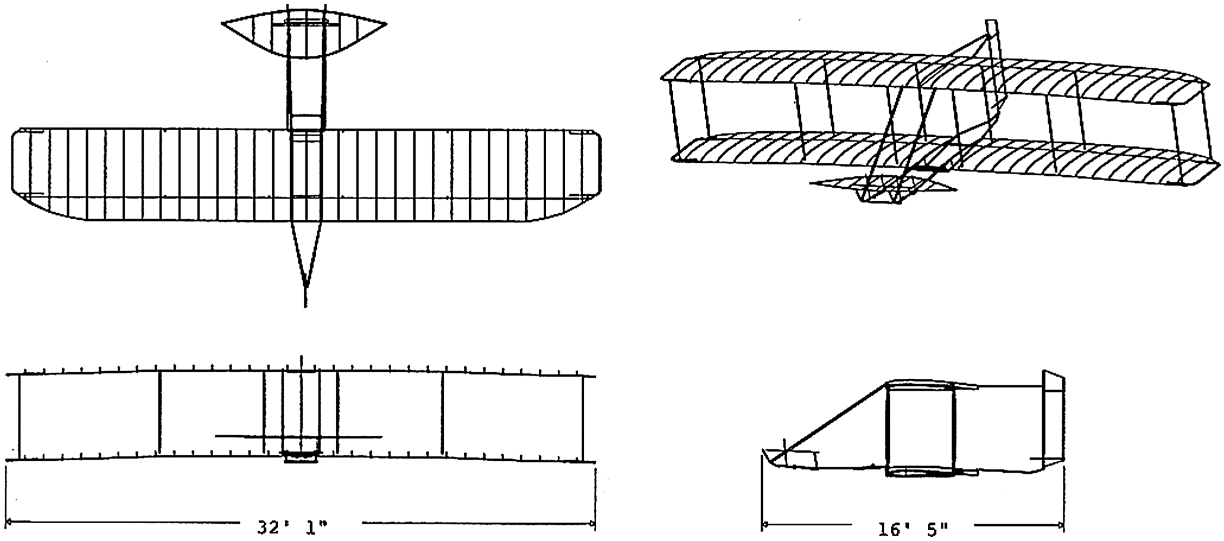

Fig. 1 Three-view of Wright 1902 glider.

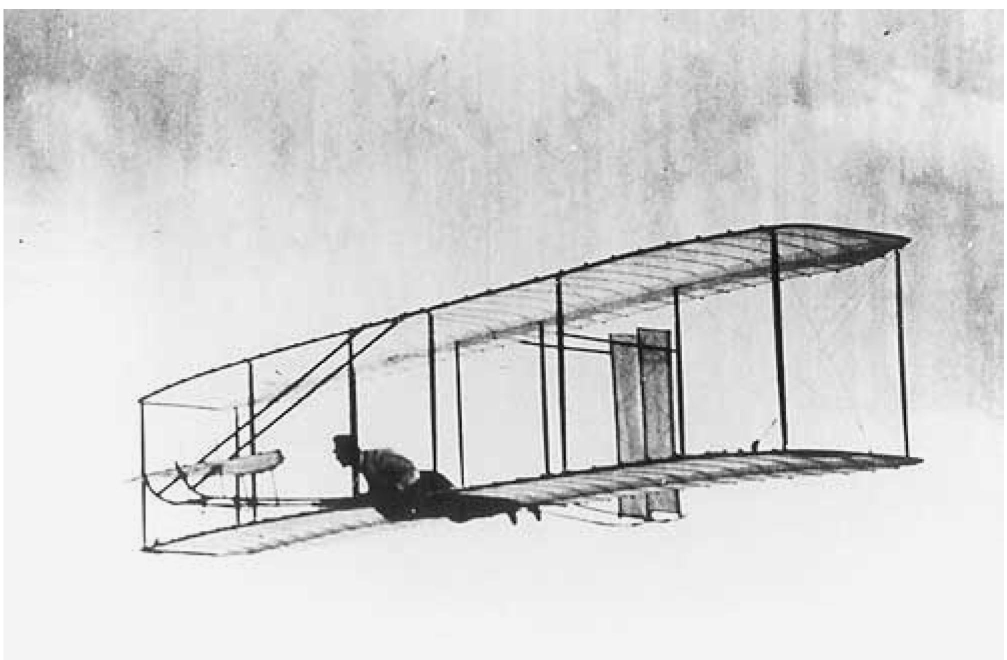

Fig. 2 Wright 1902 glider in flight, fixed-tail configuration.

part of the following decade that thicker airfoils were discovered to be more suitable at typical flying speeds for passenger carrying and combat aircraft.[1]

Interestingly, the actual shape used for the 1902 glider was not tested, but was an extrapolated shape based on their extensive testing and flight experience.[2] The wing was based on the number 12 airfoil, showing the "highest dynamic efficiency,"[1] or highest L/D of their tested surfaces. Instead of the 1/20 camber of airfoil 12, however, the 1902 glider featured a 1/24 camber, a modification most likely based on the results of their 1901 flights that showed improved pitch stability with a lower-camber wing.

The Wrights returned to Kitty Hawk in late summer of 1902 with the improved glider and made hundreds of flights during the months of September and October. The new glider featured a 32.1-ft (9.78-m) wingspan, 5.0-ft (1.52-m) geometric chord, and weighed 112 lb (50.8 kg). The wing area was 303 ft^2 (28.2 m^2) and the aspect ratio was 6.8, greatly increased over the $\mathit{AR} = 3.4$ of the 1901 glider. Like the 1901 glider, lateral control was affected by wing warping, although the pilot control was now in the form of a hip cradle. The canard consisted of a 14.4-ft^2 (1.34-m^2). lenticular shape, mounted with a central pivot located 70 in. (1.78 m) in front of the wing leading edge. Pitch control was accomplished by rotating a wooden bar directly in front of the wing leading edge with a connection made to the canard via actuating lines and pulleys. Figure 1 shows the glider planform and Fig. 2 shows the glider in flight.

At first, a tail consisting of two vertical surfaces with a total area of 11.2 ft^2 was incorporated in the design. The adverse yaw of the 1901 glider suggested a vertical tail would solve the problem; however, it was only part of the solution. The fixed tail would yaw the aircraft into the direction of its sideslip, resulting in a spiraling descent and impact the Wrights referred to as "well digging" (Ref. 2, p. 173). Despite this problem, the gliding results were encouraging and Wilbur Wright wrote (Ref. 3, p. 261) optimistically of their flight test program to Octave Chanute:

> The distances were not measured accurately except in a few cases. The usual length was 150 ft. to 225 ft. The times were 10 to 12 s for the most part, though we measured but few. In fact, we have spent but little time in measurements and have consequently greatly increased the amount of practice, which we now consider to be the only thing now lacking to attain soaring flight.

The lateral control problem was not improving, and on 3 October, Orville suggested to Wilbur that a movable tail may be the solution. The next day, they started construction of a single-surface rudder that was interconnected to the warping mechanism to provide yawing moment that, with a warp deflection, would result in a coordinated turn. Actually, it was the ability to exit a developed turn successfully that they most desired because the recovery from a low wing situation was not presently working. The new rudder greatly improved their ability to control the glider, and flights of over 300 ft became commonplace. In a five day period in October, they logged 250 flights and set a distance record of 622 ft in 26 s (Ref. 3, p. 280).

When they returned to Kitty Hawk the following year with a powered machine, the 1902 glider was flown for pilot proficiency. This time, a double, movable rudder was installed, and the time aloft record became 1 min 12 s. Having satisfied the requirements for controlled flight, the Wrights filed for patent 821, 393 on 23 March 1903 that described a three-axis controlled, biplane glider. The patent was issued 22 May 1906.

Wind-Tunnel Test Overview

To rediscover the events leading up to a successful powered flight in 1903, a reproduction of the 1902 Wright glider was fabricated at The Wright Experience™ facility in Warrenton, Virginia, for analytical investigation. This aircraft was wind tunnel tested at the NASA Langley Research Center Full-Scale Tunnel (LFST), operated since 1997 under a memorandum of agreement between Old Dominion University and NASA. This facility (formerly the NASA 30 by 60) is currently the largest operating wind tunnel in the United States.

The open jet test section of the LFST is semi-elliptical in cross section with a width of 18.29 m (60 ft) and a height of 9.14 m (30 ft) as shown in Fig. 3. The elevated ground board is 13 m (42.5 ft) wide by 16 m (52 ft) long and features a turntable with a diameter of 8.7 m (28.5 ft). Power is supplied by two 3 MW (4000 hp) electric motors driving two 11-m- (35.5-ft-) diam four-bladed fans.

Full-scale aircraft are supported on three struts, which are shielded from the flow to minimize tare loads and corrections. The two main struts are used as pivots to allow the aircraft to pitch by articulating the third (tail) strut. This entire assembly is mounted to the turntable and can be rotated in yaw. The struts transfer loads to the six-degree-of-freedom external balance located below the turntable. The balance utilizes load reduction linkages consisting of lever arms and knife-edge pivots ultimately driving balance beam scales with strain gauge outputs. The scale voltages are converted to aerodynamic coefficients using a 16-bit analog-to-digital personal computer based data acquisition system. Figure 4 shows the 1902 glider installed in the wind tunnel.

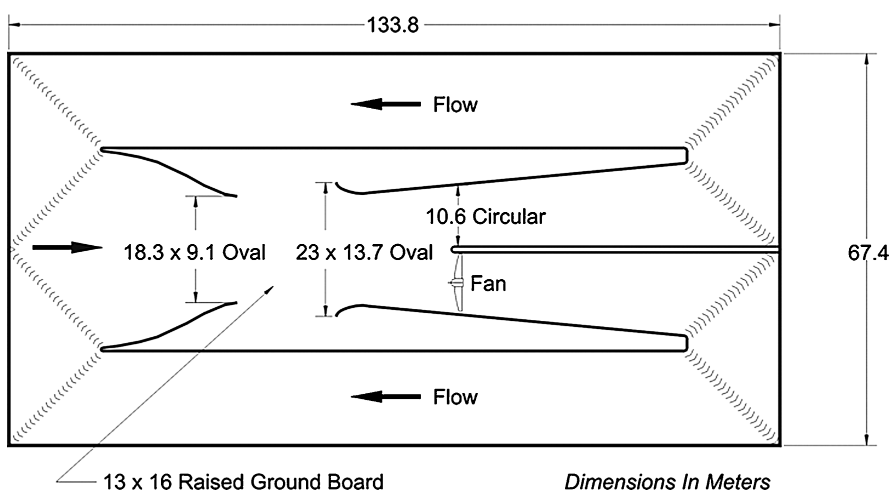

Fig. 3 LFST planform view.

Fig. 4 LFST with 1902 glider installation.

The 1902 glider was mounted to the three-post system via the inboard wing strut attach points. These are located on the bottom leading-edge spar and also at the center of the rear spar. Rod end bearings were used on the rear spar attach point to eliminate interference with the wing warping action. In flight, wing warping causes an equal but opposite rotation about the longitudinal axis of the front and rear spars. The top and bottom wings develop an equal but opposite rotation about the vertical axis. Fixing the bottom leading-edge spar creates a reference datum for use in conversion to the in-flight conditions. All data shown are corrected for support tares (primarily drag) and the boundary effect caused by the ground plane 4.1 m below the glider.

The bias limits for the external balance based on a nominal dynamic pressure of $Q = 1$ psf (48 Pa) were computed from Ref. 4:

$$B_{C_L} = 0.0086, \quad B_{C_D} = 0.0060, \quad B_{C_Y} = 0.011$$
$$B_{C_m} = 0.024, \quad B_{C_l} = 0.0039, \quad B_{C_n} = 0.0026 \quad (1)$$

The low dynamic pressure and flexible airframe lead to expected increases in uncertainty when compared to typical tests of rigid models. The test program allowed for repeated angle-of-attack sweeps to be performed in an effort to quantify the precision error. When the AIAA-recommended standard practice[5] for wind-tunnel data uncertainty estimation is used, the combined bias and precision errors lead to the following average estimates of uncertainty for the longitudinal coefficients over the angle-of-attack range tested:

$$P_{C_L} = 0.032, \quad P_{C_D} = 0.004, \quad P_{C_m} = 0.036 \quad (2)$$

$$U_{C_L} = \sqrt{B_{C_L}^2 + P_{C_L}^2} = 0.033, \quad U_{C_D} = \sqrt{B_{C_D}^2 + P_{C_D}^2} = 0.007$$

$$U_{C_m} = \sqrt{B_{C_m}^2 + P_{C_m}^2} = 0.043 \quad (3)$$

The test program budget did not support repeated measurements for the lateral aerodynamic coefficients, and hence, precision limits were not established.

Whereas enough baseline data were taken (zero warp, zero canard deflection) to provide a reasonable estimate of mean pitching moment, the same is not true for the canard-deflected cases. A canard-only test was run to obtain canard control power data because the full aircraft data did not prove to be sufficient. Details of this test are found in the "Longitudinal Results and Discussion" section.

The tunnel corrections applied to the data were implemented using a vortex lattice model of the aircraft mounted in the tunnel facility. The LFST is unique in that it contains a ground plane within an open section, so that flow constraint only occurs on one side of the test section. Corrections using the method of images,[6] implemented with vortex lattice gave an increase in drag of 3.2% and a decrease in lift of 2.6% when the aircraft is modeled in free air.

Test Plan

The test program included angle-of-attack (AoA) sweeps from $\alpha = -4$ to ± 20 deg and sideslip angles of $\beta = \pm 20$ deg. Control deflections for the canard of $\delta_c = \pm 20$ deg and wing warping $\delta_w = \pm 3.1$ deg (measured at the tip) allowed a description of available control power through all phases of flight. Dynamic pressures of $0.85 \leq Q \leq 2.8$ psf ($40 \leq Q \leq 134$ Pa) were used to simulate the range of operating speeds, with the high-end based on Wilbur Wright's statement, "As we were able to obtain speeds of more than 30 miles on hills of 10° or less. . .." (Ref. 3, p. 344).

The single, movable rudder configuration was used for most of the testing, with limited testing of the double rudder occurring late in the program. All double-rudder testing was performed without the wing warping interconnect, but with wing warp-only data, the effectiveness of the interconnected double rudder could be determined. Horizontal center of gravity was set at 18 in. aft of the leading edge, as noted in Orville's notebook (Ref. 3, p. 278), and vertical c.g. was estimated to be 16.1 in. (0.41 m) above the center of the lower front spar. Table 1 summarizes the runs.

Longitudinal Results and Discussion

Runs 1–9 represent the longitudinal runs, from which the baseline lift, drag, and pitching moment data were obtained. A third-order polynomial was fit to the lift data, showing an approximate straight-line lift curve slope in Fig. 5 of $C_{L\alpha} = 0.061$ per deg and an intercept of $C_{L0} = 0.021$. A maximum lift coefficient of $C_{L\max} = 1.0$ was measured at 19 deg. The drag polar is shown in Fig. 6, with minimum

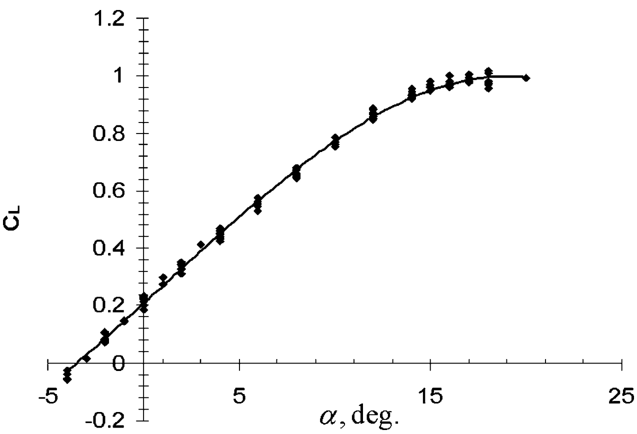

Fig. 5 Baseline lift coefficient for 1902 Wright glider.

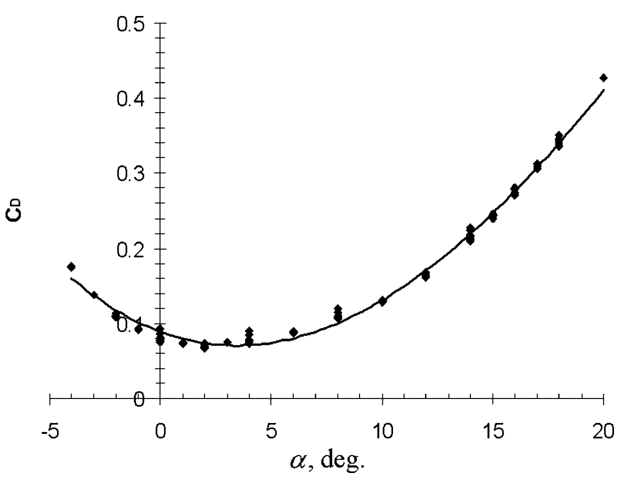

Fig. 6 Baseline drag coefficient for 1902 Wright glider.

Table 1 Test plan: 1902 glider

Runs	Warp	Canard	Beta, deg	AoA minimum	AoA maximum	Q, psf
1–9	0	0	0	−4	20	0.85–2.8
10–13	0	$+\frac{1}{2}, -\frac{1}{2}$	0	−4	18	0.85–1.0
14–25	0	+Full, −full	0	−4	18	0.85–2.6
26–33	0	0	±10, ±20	−4	18	0.85–1.0
34–49	±Half, full	0	0	−4	18	0.85–1.8
50–61	±Half, full	0	±10, ±20	−4	18	0.85–1.0
62	± Full, no rudder	0	0	0	0	1.0
63–68	±Full double rudder only, no warp	0	0	0	18	1.0

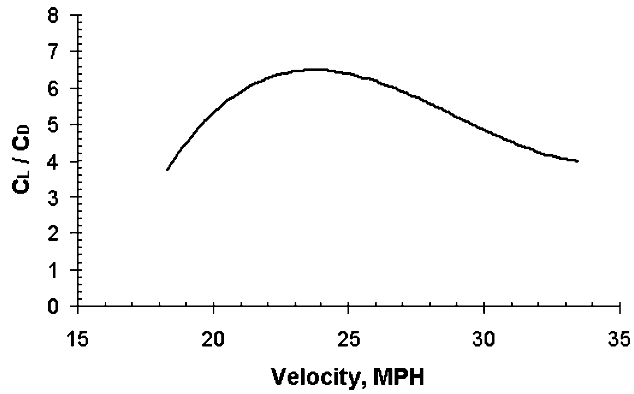

Fig. 7 Standard conditions, 1902 glider trim *L/D*.

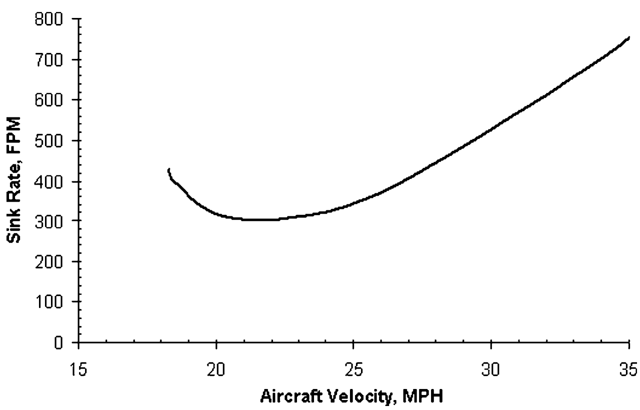

Fig. 8 Sink rate for 1902 glider, standard conditions.

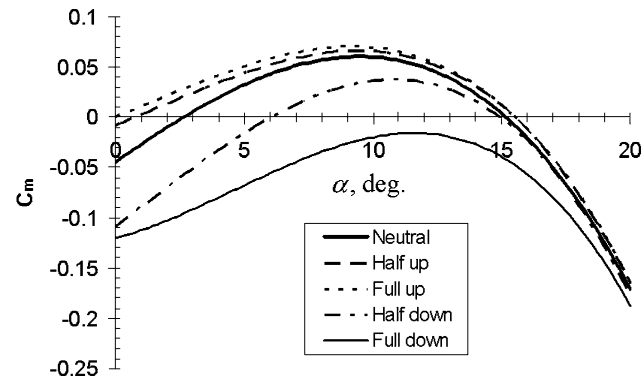

Fig. 9 Pitching moment and canard control power.

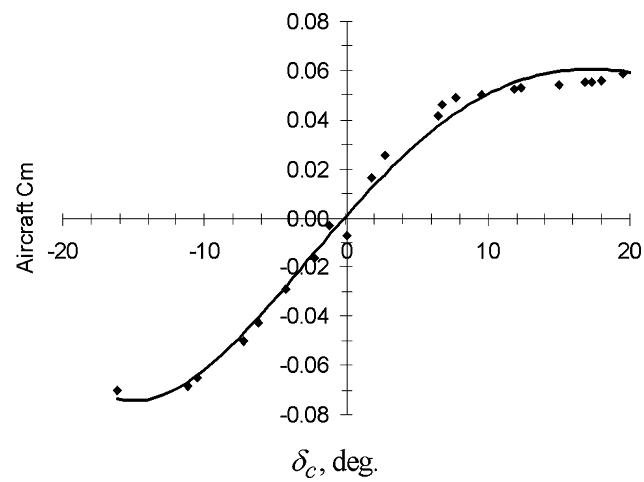

Fig. 10 Canard control power, canard-only test.

drag $C_{D\,\mathrm{min}} = 0.067$ at $\alpha = 1$ deg. Less sensitivity to dynamic pressure was observed during these tests than was the case with the 1901 glider,[7] and so most data were acquired at a $Q = 1.0$ psf (48 Pa) for both longitudinal and lateral cases.

The trim lift/drag ratio (L/D) of the glider is shown in Fig. 7, with an optimal $L/D = 6.5$ at a gliding speed of 24 mph and an AoA $= 6$ deg at standard conditions. All baseline data were taken with the canard angle set at zero, so drag corrections due to deflected trim canard angles were included in the L/D calculations. This correction was aided by the test of the canard separate of the airplane, discussed in detail later.

Sink rate is shown in Fig. 8, with a minimum rate of 302 ft/min (1.53 m/s) at 24 mph (10.7 m/s). Figure 8 was generated using a pilot weight of 145 lb (65.8 kg), close to the Wrights' actual weights. With much of their gliding performed within a few feet of the ground, an analysis of the ground effect benefit was performed using vortex lattice theory. An improved L/D of 7.4 is realized at a flight altitude of 3 ft, resulting in a gliding angle of 7.7 deg. Wilbur Wright had reported in a letter to Octave Chanute on 3 November 1902 that they achieved a minimum gliding angle of 6.5 deg (Ref. 3, p. 281), but this was in the presence of a strong up-slope wind that may have affected their estimates. Only in stable gliding flight where the aircraft maintains a constant altitude with respect to the dune slope will the aircraft L/D be accurately measured, and that L/D will be equal to the slope gradient.

The pitching moment trend and canard control power is shown in Fig. 9, with the neutral canard data derived from nine alpha sweeps. Time and budget constraints allowed only two alpha sweeps from 2 to 18 deg with the canard fully deflected up and down, and only a single sweep was made at ± half deflection. With these limited canard-deflected runs, higher than acceptable uncertainties in pitching moment occurred because the canard-induced pitching moments are close to the balance bias error ($B_{C_m} = 0.024$). As a result, a canard-only test was run with a NASA Langley Research Center 711B balance to measure lift, drag, and moment from the canard more accurately. Canard control power on the aircraft was then determined using the measured coefficients and an upwash estimate from vortex lattice theory. Figure 10 shows the derived aircraft pitching moment at $\alpha = 0$ deg from the canard-only test as the canard is deflected over its $\delta_c = 20$ deg range.

The stall region of the canard, shown in Fig. 10, explains why Fig. 9 shows limited control power for positive canard deflections. The canard will already be stalled at $\alpha = 8$ deg with $\delta_c = 0$ deg (upwash estimated from vortex lattice to be about 2 deg). Wilbur Wright stated in 1901 that "...a smaller surface set at a negative angle in front of the main bearing surfaces, or wings, will largely counteract the effect of the fore and aft travel of the center of pressure (Ref. 3, p. 107). Whether or not it was intentional, the 1902 glider incorporated a canard that trims at negative angles for typical flight speeds. This had the beneficial effect of maximizing control power around trim conditions, allowing adequate nose-up and nose-down power for maneuvering. Figure 1 reveals a negative incidence in flight, and further investigation of the original design shows the fabric positioned on the bottom of the canard frame, allowing for some small camber effect as the covering billows under load.

In addition to using the canard-only data of Fig. 10 to define the canard control power, the baseline pitching moment curve in Figure 9 has been adjusted downward by $C_{m\,\mathrm{corr}} = -0.03$ to match flight-test data. In recent flights of the 1902 glider, performed under tow in Warrenton, Virginia, pilots adjusted their c.g. during flight to determine the absolute limits of controllability at the most-aft c.g. position. The wind-tunnel-measured aircraft pitching moment would not allow flight at these c.g. positions, and so the entire curve was shifted to correct for the two likely sources of bias error:

1) The data were obtained using pitch pause sweeps with alpha increasing in the presence of a slight binding in the balance system. Because of the relatively low-force levels generated by the glider, a binding condition of the balance system will have the effect of causing a significant parameter error that lags the trend with AoA.

2) There were changes in the aircraft geometry that occurred between the wind-tunnel test and flight test, such as fabric shrinkage. With the moment shift included, recent pilots' perception of aircraft handling is in good agreement with simulation based on the data.

The moment curve shows a similar characteristic that was observed in the 1901 glider test, with a strong nonlinearity typical in canard-configured biplanes. Instability is observed from 0 to 10 deg AoA, and stable flight is possible from 10 to 20 deg. Separation of flow on the wings caused by the sharp leading edge is suspected as the primary cause of the nose-down moment above 10-deg AoA. Because the aircraft flies near the neutrally stable condition, any change to pitch attitude will result in either increasing or decreasing the aircraft stability, requiring added vigilance on the part of the pilot.

Lateral Results and Discussion

The Wrights' development of a wing warping system to affect lateral control was one of their most important contributions to the science of flight. Warping is an aerodynamically efficient way to provide a rolling moment and has a minimal impact on weight and material. Figure 11 shows the warp control power of the 1902 glider, for full deflection ($\delta_w = \pm 3.1$ deg) and half-deflection ($\delta_w = \pm 1.6$ deg). The average roll control power at a trim AoA of 6 deg is $C_{l\delta w} = 0.0060$/deg, a value that is more than double that found in the 1901 glider.

With the inclusion in 1902 of a 5.8-ft^2 movable vertical surface that is interconnected to the warp control, the glider shows proverse yawing moment for half-warp deflection up to 13.5-deg AoA. At full warp deflection, the vertical surface power is overcome by adverse yaw at 10-deg AoA while the rudder has a measured deflection of 32 deg. Figure 12 shows the yawing moment for the single-rudder configuration, and Fig. 13 shows the benefit of the 1903 double-rudder configuration, providing positive yaw power up to 12.5-deg AoA for full warp deflection. The Wrights extended their successful experiments with the 1902 glider in 1903 for flight proficiency using

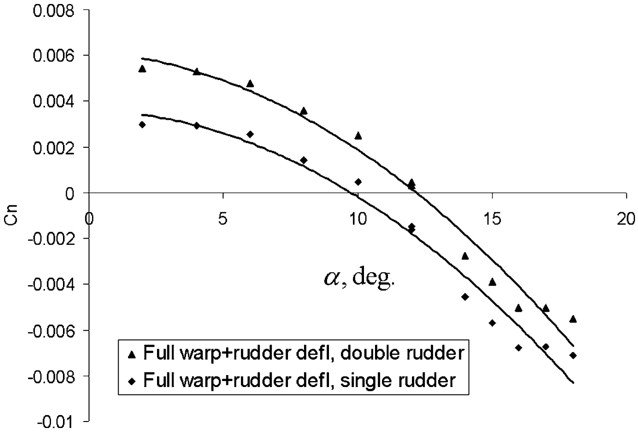

Fig. 13 Yaw moment due to warp δ_w.

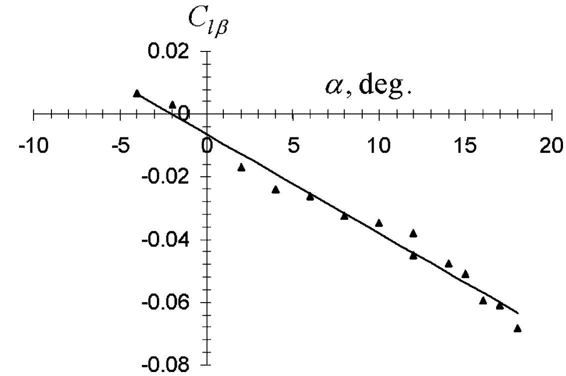

Fig. 14 Roll stability in sideslip.

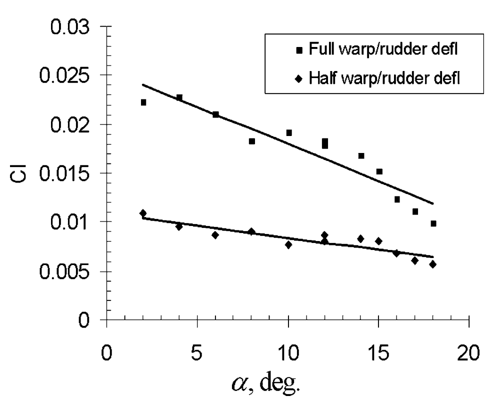

Fig. 11 Roll moment due to warp δ_w.

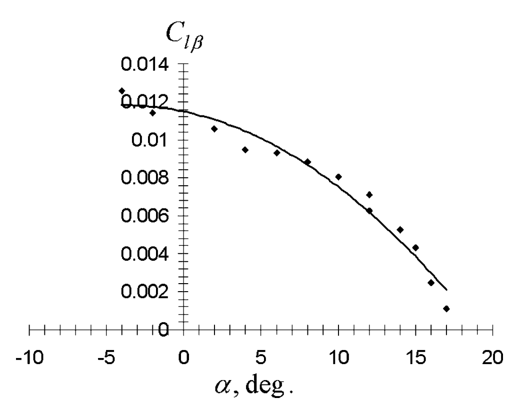

Fig. 15 Yaw stability in sideslip.

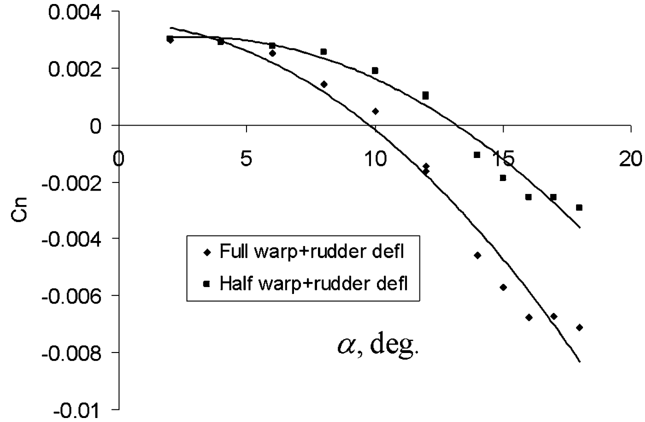

Fig. 12 Yaw moment due to warp δ_w, single-rudder configuration.

the improved vertical tail before their successful powered-machine-flights.

Even with the addition in 1902 of the vertical tail, it is obvious that when the aircraft reaches 10-deg AoA there will be a tendency for adverse yaw to take effect, potentially making a landing tricky. The 255-lb (116-kg) weight at which the Wrights flew corresponds to a speed of 21 mph (9.4 m/s).

In sideslip, the glider demonstrated typical performance found in modern aircraft, with negative $C_{l\beta}$ and positive $C_{n\beta}$ shown in Figs. 14 and 15. Despite the 4-in. anhedral angle built into the glider when the Wrights attempted to alleviate the glider's tendency to raise a wing when side slipping, the aircraft still proved to have slight dihedral stability. This stability is not desired in a ground effect machine, where a turn into a side gust is desirable to keep the airplane aligned with the freestream velocity. The rolling tendency away from a side gust will delay the recovery maneuver to keep the airplane flying into the wind.

A small side-force coefficient at trim AoA of $C_{Y\beta} = 0.12$/rad and the relatively small yaw stability coefficient $C_{n\beta}$ result in lateral wandering of the aircraft, complicated by inertial coupling and adverse yaw. This is further explored in the next section.

Flight Test and Simulation

The development of a 1902 glider simulator and recent flights in the machine have allowed for a quantitative evaluation of its handling characteristics. Five pilots have flown this particular aircraft under towed and free-flight conditions and provided feedback to the first author regarding handling characteristics. The first author has logged approximately 80 flights in the aircraft, with the majority being conducted under tow behind a ground vehicle in near-calm conditions. Free flights were accomplished at Nags Head, North Carolina, on Jockey's Ridge in wind speeds ranging from 10 to 20 kn (5.1–10.3 m/s). Center of gravity variations of 7.3 cm forward and 3.8 cm aft of nominal were tested for longitudinal handling evaluation. Figure 16 shows the glider in towed flight.

Longitudinally, the aircraft's most notable characteristic is its quick pitch response. With pitch inertia estimated to be very low with $I_{YY} = 25\ \mathrm{lb} \cdot \mathrm{s}^2 \cdot \mathrm{ft}$ and pitch damping estimated from pilot experience to be $C_{mq} = 3.0$, the aircraft has a very light feel and can suddenly pitch up or down due either to a pilot command or a gust. For comparison purposes, a Cessna 182 Skylane four-passenger aircraft has a pitch inertia of $I_{YY} = 1346\ \mathrm{lb} \cdot \mathrm{s}^2$ ft and a pitch damping value of $C_{mq} = -12.4$. In addition to the liveliness in pitch, the pilot has little mechanical advantage in the operation of the canard, adding to the challenge of controlling pitch excursions.

The moment curve shows that changes in the AoA have a dramatic effect on stability. A pitch-up maneuver will result in more stability, as AoA increases in response to an increasing n_z. When the aircraft pitches down, the load factor decreases, which lowers the angle of attack, causing instability. This variable stability requires added attention when close to the ground, as a pitch-down excursion can cause an irrecoverable departure.

Figure 17 shows a simulated 1-s pitch doublet for $\delta_c = \pm 9.5$, -6.7 deg, with the AoA α and aircraft pitch angle θ shown. The

Fig. 16 Wright 1902 glider in towed flight.

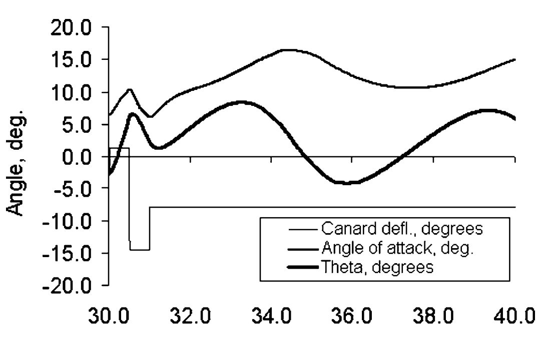

Fig. 17 Pitch doublet performed on applied research simulator.

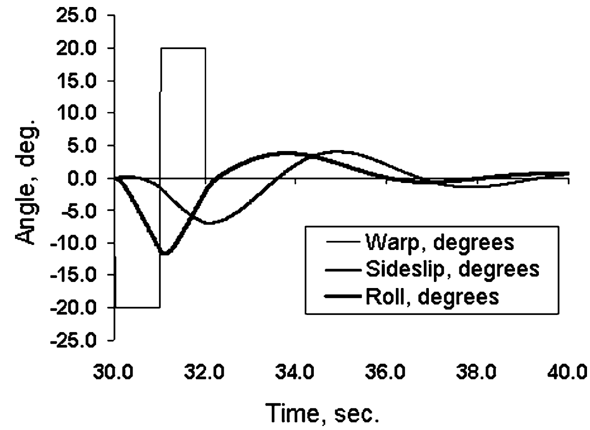

Fig. 18 Roll doublet performed on applied research simulator.

aircraft enters a stable, deep stall in such a maneuver. This occurs because of the two trim points found on the moment curve for most canard positions. Without pilot input, the aircraft will seek the stable trim point and remain there.

Roll doublets were performed in the simulator and on the towed machine to evaluate lateral handling characteristics. Because the tow bridle restricted yaw motion on the actual aircraft by inducing a counteracting yawing moment, flight observations were focused on the rolling behavior only.

Referencing both the towed flights and the glider simulator provided a roll damping estimate of $C_{lp} = -0.8$, a value that indicates stable roll behavior for the light aircraft. Figure 18 shows a simulated warp doublet that results in a roll rate of about 11 deg/s for the full warp deflection of $\delta_w = \pm 3.1$ deg. The sideslip due to adverse yaw at the initiation of the doublet is minor, showing that the moveable rudder is quite effective in coordinating turns. On the recovery, however, the rudder control power plus the favorable inertial coupling results in a continued sideslip buildup that produces a skidding turn. The lack of vertical surfaces on the aircraft tends to exacerbate the sideslip as well.

Lateral control can disappear when the aircraft encounters a gust. A measured 4-mph gust is capable of overpowering the lateral control. Because this aircraft has little pendulum stability, an upset requires immediate attention from the pilot.

Conclusions

The 1902 glider established the Wrights as the first true aeronautical engineers. Having used a systematic approach to solve the problem of controlled, manned flight, they developed an essentially modern aircraft the basic design of which has changed little in 100 years. When the materials and construction method used for this durable aircraft are considered, its maximum L/D of 7.4 was a remarkable achievement and a testimonial to the analysis and design talents of the Wrights.

This aircraft served the Wright brothers in many ways: It demonstrated an effective three-axis control system, it verified the efficiency of an airfoil and wing design from their wind-tunnel-test program, and it provided valuable pilot training time in preparation for the flights of the 1903 powered machine. In all, its contribution to the first manned, powered flight is perhaps as significant as the 1903 Flyer itself.

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