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A METHOD FOR ESTIMATING THE ROLLING MOMENT DUE TO SPIN RATE FOR ARBITRARY PLANFORM WINGS

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ABSTRACT

The application of aerodynamic theory for estimating the force and moments acting upon spinning airplanes is of interest. For example, strip theory has been used to generate estimates of the aerodynamic characteristics as a function of spin rate for wing-dominated configurations for angles of attack up to 90 degrees. This work, which had been limited to constant chord wings, is extended here to wings comprised of tapered segments. Comparison of the analytical predictions with rotary balance wind tunnel results shows that large discrepancies remain, particularly for those angles-of-attack greater than 40 degrees.

NOMENCLATURE

b	wing span
c	wing chord
c_o	wing segment chord at the in-board edge
C_L	lift coefficient
C_ℓ	rolling moment coefficient = $\frac{L}{qb \text{ area of segment}}$
C_N	normal force coefficient
C_{N_o}	constant term in normal force coefficient equation
$C_{N_{\sin\alpha}}$	coefficient in normal force coefficient equation
h	slope of linear taper equation
L	rolling moment
p	roll rate, $\Omega \cos \alpha$
q	dynamic pressure
q_ℓ	local dynamic pressure
r	yaw rate, $\Omega \sin \alpha$
V	velocity
V_ℓ	local velocity
u,v,w	velocity components, center of wing
x,y,z	coordinates
α	angle of attack
α_ℓ	local angle of attack
ρ	air density
Ω	rotation rate

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INTRODUCTION

The use of parameter estimation in modeling aircraft dynamics has been quite successful for many mathematical models of flight. Parameter estimation is most readily applied when linear models representing small perturbations from straight equilibrium paths are appropriate. Flight data is most accurate in this regime and the mathematical model is the simplest.[1]

Parameter estimation becomes more complex in application to spinning aircraft. Modeling nonlinear aerodynamics, including rotational flow effects, is much more difficult and many more unknown parameters are introduced.[2] In order to reduce the large number of unknowns it is helpful to apply strip theory of reference 3. Strip theory "links the wing airfoil section characteristics to the rolling and yawing moment of the wing in spinning flight."[1]

In reference 1, strip theory provided a mathematical model that was used to determine the rolling moment of a wing in spinning flight. Calculated rolling moment forces due to the wing were about 50 percent larger than the experimental rotary balance spin-tunnel measurements of a wing-dominated aircraft. It is the purpose of this paper to expand the existing mathematical model of a spinning wing in order to more closely represent an aircraft in spinning flight, and to further explore the limitations and possibilities of the more general model. Specifically, the strip theory technique of reference 1 will be extended to wings comprised of tapered segments. The same limitation of reference 1 will be used in that the flow angle at each strip location is independent of the incremental lift at other locations.

DISCUSSION

In order to decrease the complexity of estimating the rolling moment due to spinning, the authors in reference 1 restricted their analysis to the rolling moment produced by an untapered wing of a wing-dominated aircraft. In this paper the approach is extended to wings of arbitrary planform by considereing a wing to be made up of sections of differing taper.

Let us first consider the local flow characteristics for the general spanwise location y, shown in figure 1.

$$V_\ell^2 = (u - ry)^2 + (w + py)^2$$

$$\alpha_\ell = \arctan\left(\frac{w + py}{u - ry}\right) = \arcsin\left(\frac{w + py}{\sqrt{(u - ry)^2 + (w + wy)^2}}\right)$$

and

$$q_\ell = \frac{\rho}{2}\left[(u - ry)^2 + (w + py)^2\right]$$

For wings having a constant taper, the wing chord can be represented by a linear equation:

$$c = c_o - hy \qquad \text{for} \qquad y > 0$$

$$c = c_o + hy \qquad \text{for} \qquad y < 0$$

The equation for rolling moment for a single strip would be:

$$dL = -\frac{\rho}{2}\left[(u - ry)^2 + (w + py)^2\right] C_N \left(c_o \pm hy\ y\ dy\right)$$

For the entire wing the rolling moment becomes:

$$L = -\frac{\rho}{2}\int_{-\frac{b}{2}}^{\frac{b}{2}} \left[(u - ry)^2 + (w + py)^2\right] C_N(y)\left(c_o \pm hy\right) y\ dy$$

In order to easily represent aerodynamic data at high angles of attacks, the normal force coefficient (fig. 2) is given the form:

$$C_N(\alpha) = C_{N_o} + C_{N_{\sin\alpha}} \sin\alpha$$

It follows then that a single wing section over which the normal force equation is applicable will have the following contribution to rolling moment:

$$\Delta L = -\frac{\rho c_o C_{N_o}}{2}\int_{y_{lower}}^{y_{upper}} \left[(u - ry)^2 + (w + py)^2\right] y\ dy \mp$$

$$\frac{h\rho C_{N_o}}{2}\int_{y_{lower}}^{y_{upper}} \left[(u - ry)^2 + (w + py)^2\right] y^2 dy -$$

$$\frac{\rho c_o C_{N_{\sin\alpha}}}{2}\int_{y_{lower}}^{y_{upper}} \sqrt{(u - ry)^2 + (w + py)^2}\ (w + py) y\ dy \mp$$

$$\frac{h\rho C_{N_{\sin\alpha}}}{2}\int_{y_{lower}}^{y_{upper}} \sqrt{(u - ry)^2 + (w + py)^2}\ (w + py) y^2 dy$$

After integrating,

$$\Delta L = -\frac{\rho c_o C_{N_o}}{2}\ \frac{1}{4}\left[A\left(y_{upper}^4 - y_{lower}^4\right) + \frac{1}{3} B\left(y_{upper}^3 - y_{lower}^3\right)\right.$$

$$\left. + \frac{1}{2} C\left(y_{upper}^2 - y_{lower}^2\right)\right] \mp \frac{h\rho C_{N_o}}{2}\left[\frac{1}{5} A\left(y_{upper}^5 - y_{lower}^5\right)\right.$$

$$+ \quad \frac{1}{4} B\left(y_{upper}^4 - y_{lower}^4\right)\Bigg] + \frac{1}{3} C\left(y_{upper}^3 - y_{lower}^3\right)\Bigg]$$

$$- \quad \frac{w\rho c_o C_N \sin\alpha}{2} \int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\; y\, dy - \frac{\rho C_N \sin\alpha}{2} \left(pc_o \pm wh\right) \int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\; y^2 dy$$

$$\mp \quad \frac{p\rho h C_N \sin\alpha}{2} \int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\; y^3 dy$$

where:

$$A = p^2 + r^2 = \Omega^2$$

$$B = -2ur + 2wp = 0$$

$$C = u^2 + w^2 = V^2$$

$$\Phi = Ay^2 + By + C$$

$$\int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, dy = \frac{1}{4A}\Bigg[\left(2Ay_{upper}\right) + B \sqrt{Ay_{upper}^2 + By_{upper} + C}$$

$$- \left(2Ay_{lower}\right) + B \sqrt{Ay_{lower}^2 + By_{lower} + C}\Bigg]$$

$$+ \frac{4AC - B^2}{8A\sqrt{A}} \log\left[\frac{2Ay_{upper} + B + 2\sqrt{A^2 y_{upper}^2 + ABy_{upper} + AC}}{\sqrt{2Ay_{lower} + B + 2\sqrt{A^2 y_{lower}^2 + ABy_{lower} + AC}}}\right]$$

$$\int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\; ydy = \frac{1}{3A} \Bigg[\left(Ay_{upper}^2 + By_{upper} + C\right)^{3/2}$$

$$- \left(Ay_{lower}^2 + By_{lower} + C\right)^{3/2}\Bigg]$$

$$- \frac{B}{2A} \int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, dy$$

$$\int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, y^2 dy = \frac{6Ay_{upper} - 5B}{24A^2} \left(Ay_{upper}^2 + By_{upper} + C\right)^{3/2}$$

$$- \frac{6Ay_{lower} - 5B}{24A^2} \left(Ay_{lower}^2 + By_{lower} + C\right)^{3/2}$$

$$- \frac{4AC - 5B^2}{16A^2} \int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, dy$$

$$\int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, y^3 dy = \left(\frac{y_{upper}^2}{5A} - \frac{7By_{upper}}{40A^2} + \frac{7B^2}{48A^3} - \frac{2C}{15A^2}\right) \left(Ay_{upper}^2 + By_{upper} + C\right)^{3/2}$$

$$- \left(\frac{y_{lower}^2}{5A} - \frac{7By_{lower}}{40A^2} + \frac{7B^2}{48A^3} - \frac{2C}{15A^2}\right) \left(Ay_{lower}^2 + By_{lower} + C\right)^{3/2}$$

$$- \left(\frac{7B^3}{32A^3} - \frac{3CB}{8A^2}\right) \left(\int_{y_{lower}}^{y_{upper}} \sqrt{\Phi}\, dy\right)$$

The terms in the normal force equation, C_{N_o} and $C_{N_{\sin\alpha}}$, correspond to the local angle-of-attack ranges (see fig. 2) listed in Table 1 from reference 1:

Angle-of-Attack	C_{N_o}	$C_{N_{\sin\alpha}}$
-164° to -16°	-.5	1.0
-16° to -10.5°	-1.6	-3.0
-10.5° to 10.5°	0	5.8
10.5° to 16°	1.6	-3.0
16° to 164°	.5	1.0

Table 1

The wing span locations having local angles of attack of -16, -10.5, 10.5 and 16 degrees are determined by:

$$y_{boundary} = \frac{w - u \tan(\alpha \text{ boundary})}{- p - r \tan(\alpha \text{ boundary})}$$

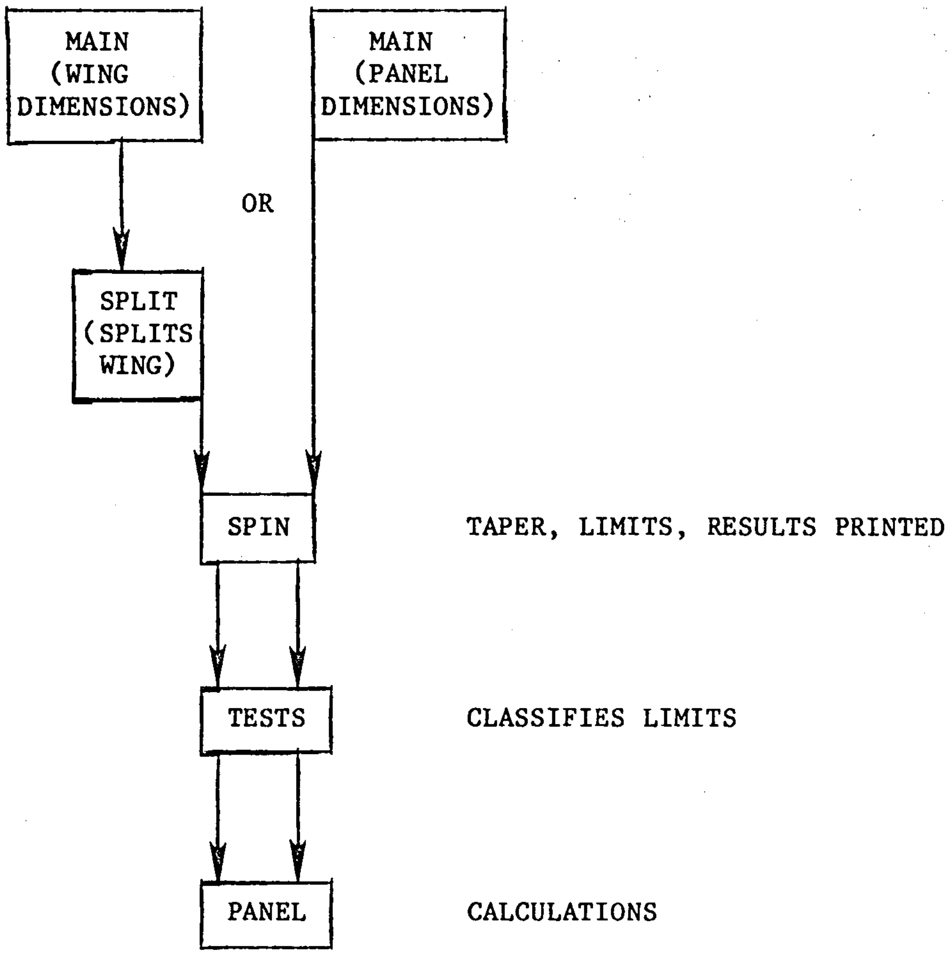
These will serve as limits of integration in the above equations if they fall in the confines of the panel being considered. If they do not, the boundaries of the panel will be used as limits.

The program used to calculate the rolling moment of the wing using the above equations is listed in the appendix. It is a series of subroutines that will calculate the rolling moment coefficient of any tapered section of a flat wing given the following data: the two boundary chord lengths of each panel; the distance of these chords from the origin; the air density; the velocity of the aircraft; the wingspan; and the area of the wing. There is an option to calculate the rolling moment coefficient of a single panel, or both symmetrical panels having the given dimensions.

The spin subroutine accepts the dimensions of the panel and calculates the slope of the linear equation describing wing taper (h). It then computes the limits of integration along the panel. These limits are sent to the intermediate tests subroutine. Tests classifies the limits and sends only those that are within the bounds of the desired panel(s) to the panel subroutine. The panel subroutine does the actual rolling moment calculation of the panel between the limits using the above equations. The split subroutine is an optional subroutine which, given the dimensions of the wing, will split a wing into its component panels and send each panel in succession to the spin subroutine.

With this program, a wing comprised of tapered panels can be modeled, panel by panel. Through a simple modification, the program can accumulate the total rolling moment of an aircraft wing due to each panel at a selected angle-of-attack. For the airplane shown in figure 3, this was done at an angle-of-attack of 14 degrees in order to obtain figure 4. Figure 4 is a plot of the total rolling moment coefficient of the wing of the aircraft, as well as the rolling moment coefficient of each of the wing's component panels as a function of nondimensional spin rate. The bottom curve of figure 4 represents the total rolling moment coefficient for the airplane of figure 3 at 14 degrees angle-of-attack.

The following flowchart is a diagram of the program:



In a typical light, wing-dominated aircraft such as the one illustrated in figure 3, the panels that cause the greatest moment are the outer panels as is shown in figure 4 and in table 2. The upper curve in figure 4 represents the rolling moment contribution of the inner panels, the next curve represents the contribution of the middle panels and the third curve from the top is the contribution of the outer panels. This figure is for a fixed angle-of-attack while the rotation rate varies. On the other hand, table 1 shows the relationship between the panels when the rotation rate is fixed and the angle-of-attack is varied. The data of table 1 and Figure 4 clearly show that the outer panels contribute from 78% to 97% of the total rolling moment. Of course, this is expected since these panels are larger than the others, have the longest moment arm, and experience the greatest variation in dynamic pressure.

Figure 5 shows the improvement caused by taking into account wing taper as compared to the values obtained with a constant chord. There is significant improvement in the data, particularly at higher rates of rotation. The upper curves are the spin-tunnel test data. Obviously, improvements in the model must be made before the method can be considered acceptable. It is interesting to note (see figures 5 and 6) that there is little difference when the wing of the aircraft in figure 3 is simplified in the calculations to two large trapezoidal panels instead of six smaller ones. However, the multi-panel approach is more accurate and is applicable to the more general case.

In reference 1, it was noted that at angles-of-attack around 50 degrees the experimental rolling moments were autorotative at low rotation rates. The calculated data of reference 1 did not represent this phenomenon. The plot of 30 and 50 degrees angle-of-attack in figure 7 shows that the new calculated data does not show autorotative moments either. With the theory being used here, it would be impossible to obtain autorotative moments except over an angle-of-attack range of 10.5 to 16 degrees since the slope of the line of normal force coefficient vs. angle-of-attack (fig. 2) is always positive except over this range. Note that figures 5 and 6 show an autorotative moment at low rates of rotation both in the test data and in the calculated data for 14 degrees angle-of-attack. However, an extension must be made to this simplified aerodynamic theory for higher angles-of-attack.[1]

The amount of error in the mathematical representation of a spinning wing has been decreased by describing the wing as a set of tapered panels. However, the errors are still large. The next step might be to consider the contribution of the tail section to the rolling moment. Since the program calculates the rolling moment of any tapered panel, the three tail panels could be input in order to determine the tail effects. The present method will compute the rolling moment for swept-wing configurations since rolling moment is independent of sweep. However, an extension of the model should also incorporate pitching moment. Of course, this method will not hold for aircraft where body effects cannot be neglected. The effects of the body would have to be considered by some other method such as the strip theory of reference 5. Improved estimates of aerodynamic moments would be expected if the induced flow effects on the flow angles were included in the formulation. Past results and future extentions promise further improvements in predicting the aerodynamic forces and moments of spinning airplanes.

CONCLUDING REMARKS

Mathematical representations of nonlinear phenomena such as the aerodynamics of a spinning aircraft are characterized by having large numbers of unknown parameters. Analytical methods such as strip theory can be used to reduce the number of unknown parameters. In this paper, strip theory is applied to compute aerodynamic forces for a wing composed of several variable taper trapezoidal panels in order to obtain a model structure which requires only the unknowns of the normal force equations. Although the error is decreased significantly by using strip theory in this manner as compared to approximating the wing as untapered, there is still much more to be done in order to analytically predict aerodynamic force of spinning aircraft. In order to extend the model further, many new parameters would have to be added. Also, it is clear that aerodynamic theory for angles-of-attack greater than 40 degrees must be improved since it is impossible to predict the results of spin-tunnel rotary balance tests with strip theory methods.

Since the program that calculates the data is general enough to accept any wing panel of an aircraft, the revised model is currently useful in comparing the effects on a panel of changing parameters such as rotation rate, angle-of-attack, velocity, taper, etc. It is also useful for comparing aircraft components. However, the error between anlytical predictions and the experimental data is still too large to consider the strip theory representation to be an effective model of a spinning aircraft.

APPENDIX

Listed in the following pages is the program used to calculate the rolling moment due to spin rate for an arbitrary planform wing. Inputs to the program are set in a short main program which calls the subroutines necessary for the calculations.

The first page shows an example of the simplest case where a single panel is input to the program. Variable CHORD is the inboard chord length and CHTIP is the outboard chord length. D1 and D2 are spanwise distances from the center line of the fuselage to CHORD and CHTIP respectively. AREA refers to the area of the entire wing containing the panel, and SPAN is the wing span. RHO is air density and VEL is the velocity of the aircraft. The last integer tells the program whether to compute the rolling moment for panel with the given dimensions on the positive side of the aircraft (0), the negative side of the aircraft (1) or both (2). Normally, this value will be 2, except when isolation a single wing panel is desired.

The second page shows a case where the panels of an entire wing will be input to the program. In this case, variable CHORD is the root chord, CH1 and CH2 are the chord lengths at the point of wing taper change, and CHTIP is the chord length at the wing tip. D1 and D2 are the spanwise distances from the center line to CH1 and CH2 respectively. FUSE is the width of the fuselage.

As was mentioned in the text, the program can be modified to accumulate the total rolling moment of a multi-paneled wing. To do this, a one-dimensional array can be defined and placed inside the main loop of the spin subroutine such that each time spin is called with a new panel's dimensions, the nine values of the panel's rolling moment coefficient array (CLW) are added to the new array for a selected angle-of-attack (corresponding to an iteration of the main loop). For more than one angle-of-attack, a two-dimensional array would be necessary.

```
      PROGRAM MAIN         74/74    OPT=1                            FTN 4.8+552         84/03/02. 08.5

                      PROGRAM MAIN(INPUT,OUTPUT)
                      CHORD=3.41
                      CHTIP=1.81
                      CH1=2.97
                      CH2=2.97
                      D1=1.045
                      D2=10.0
                      SPAN=20.
                      FUSE=2.09
                      RHO=.002378
                      VEL=100.
                      AREA=52.2
                      CALL SPIN(CHORD,CHTIP,D1,D2,AREA,SPAN,RHO,VEL,2)
                      STOP
                      END


      SYMBOLIC REFERENCE MAP (R=1)

ENTRY POINTS
 4140  MAIN

VARIABLES      SN  TYPE            RELOCATION
 4223  AREA        REAL                                   4211  CHORD      REAL
 4212  CHTIP       REAL                                   4213  CH1        REAL
 4214  CH2         REAL                                   4215  D1         REAL
 4216  D2          REAL                                   4220  FUSE       REAL
 4221  RHO         REAL                                   4217  SPAN       REAL
 4222  VEL         REAL

FILE NAMES         MODE
    0  INPUT                       2054  OUTPUT

EXTERNALS              TYPE    ARGS
       SPIN                      9

STATISTICS
  PROGRAM LENGTH                  215B     141
  BUFFER LENGTH                  4007B    2055
```

```
      PROGRAM MAIN         74/74    OPT=1                            FTN 4.8+552         84/03/05. 09.(

                      PROGRAM MAIN(INPUT,OUTPUT)
                      CHORD=3.41
                      CHTIP=1.99
                      CH1=2.97
                      CH2=2.97
                      D1=2.22
                      D2=4.66
                      SPAN=20.
                      FUSE=2.09
                      RHO=.002378
                      VEL=100.
                      AREA=55.04
                      CALL SPLIT(CHORD,CH1,CH2,CHTIP,D1,D2,SPAN,FUSE,RHO,AREA,VEL)
                      STOP
                      END


      SYMBOLIC REFERENCE MAP (R=1)

ENTRY POINTS
 4140  MAIN

VARIABLES      SN  TYPE            RELOCATION
 4224  AREA        REAL                                   4212  CHORD      REAL
 4213  CHTIP       REAL                                   4214  CH1        REAL
 4215  CH2         REAL                                   4216  D1         REAL
 4217  D2          REAL                                   4221  FUSE       REAL
 4222  RHO         REAL                                   4220  SPAN       REAL
 4223  VEL         REAL

FILE NAMES         MODE
    0  INPUT                       2054  OUTPUT

EXTERNALS              TYPE    ARGS
       SPLIT                    11

STATISTICS
  PROGRAM LENGTH                  216B     142
  BUFFER LENGTH                  4007B    2055
```

```
      C
            SUBROUTINE SPIN(CH1,CH2,D1,D2,AREA,SPANW,RHO1,VEL,ITOG)
      C
      C     ***************************************************************
      C     * THIS SUBROUTINE GENERATES NON-DIMENSIONALIZED VALUES FOR THE *
      C     * ROLLING MOMENT OF A SPINNING AIRCRAFT WING PANEL AT ANGLES OF *
      C     * ATTACK FROM 0 TO 90 DEGREES AND AT NON-DIMENSIONALIZED ROTATION *
      C     * RATES FROM 0 TO .9.  THE MAIN ROGRAM MUST DEFINE VALUES FOR THE *
      C     * BOUNDARY CHORDS, THE DISTANCES TO THESE CHORDS FROM THE ORIGIN, *
      C     * THE WINGSPAN, THE VELOCITY OF THE AIRCRAFT, THE AIR DENSITY, THE *
      C     * AREA OF THE WING, AND AN INTEGER TO SPECIFY WHETHER OR NOT THE *
      C     * USER WANTS TO DESCRIBE A NEGATIVE PANEL, POSITIVE PANEL OR BOTH. *
      C     * THE PRIMARY PURPOSE OF THIS SUBROUTINE IS TO COMPUTE THE LIMITS *
      C     * OF INTEGRATION ON THE PANEL.                                   *
      C     ***************************************************************
      C
      C DEFINITION OF VARIABLES
      C
      C     ALPHA - THE ANGLE OF ATTACK IN RADIANS
      C     ALPHD - THE ANGLE OF ATTACK IN DEGREES
      C     A1    - THE FIRST LOCATIONAL ANGLE OF ATTACK - 10.5 DEGREES
      C     A2    - THE SECOND LOCATIONAL ANGLE OF ATTACK - 16 DEGREES
      C
      C     CH1, CH2 - TWO LIMIT CHORDS OF THE PANEL
      C            (CH2 IS LARGER THAN CH1)
      C
      C     CLW - NON-DIMENSIONALIZED ARRAY FOR ROLLING MOMENT
      C     CNSA - CONSTANT TERM IN NORMAL FORCE COEFFICIENT EQUATION
      C     CNO - COEFFICIENT IN NORMAL FORCE COEFFICIENT EQUATION
      C
      C     D1 - DISTANCE TO CH1
      C     D2 - DISTANCE TO CH2
      C
      C     ELW - ROLLING MOMENT
      C
      C     M - SLOPE OF THE LINEAR EQUATION DESCRIBING THE WING TAPER
      C     ITOG - 0 FOR POSITIVE PANEL; 1 FOR NEGATIVE PANEL; 2 FOR BOTH
      C
      C     LA,LB,LC,LD,LE - LOWER LIMITS OF INTEGRATION
      C     UA,UB,UC,UD,UE - UPPER LIMITS OF INTEGRATION
      C
      C     OMEGA - ROTATION RATE
```



```
      C     P - ROLL RATE
      C     QUE - DYNAMIC PRESSURE
      C     R - YAW RATE
      C     RHO - AIR DENSITY
      C
      C     SPAN - WING SPAN
      C
      C     U - COMPONENT OF VELOCITY
      C     VEL - VELOCITY
      C     W - COMPONENT OF VELOCITY
      C     WB2V - NON-DIMENSIONALIZED ROTATION RATE
      C
      C     YM1 - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF -10.5 DEGREES
      C     YM2 - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF -16   DEGREES
      C     Y1  - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF  10.5 DEGREES
      C     Y2  - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF  16   DEGREES
      C
      C
            DIMENSION CLW(50)
        700 FORMAT(11E12.4)
            REAL LA,LB,LC,LD,LE,MFUSE
            COMMON // A,B,C,W,P,M,MFUSE,SPAN,CNO,CNSA,ELW,CHORD,RHO
      C
      C
            A1=10.5/57.3
            A2=16./57.3
            RHO=RHO1
            SPAN=SPANW
            MFUSE=D1
            M=(CH1-CH2)/(D2-D1)
            CHORD=CH1+M*(ABS(D1))
            QUE=.5*RHO*VEL*VEL
            ALPHA=-2./57.3
      C
      C BEGINNING FIRST LOOP
      C
            DO 1 I=1,46
      C
      C    INCREMENTING ANGLE OF ATTACK BY 2 DEGREES
      C
            ALPHA=ALPHA+2./57.3
            ALPHD=57.3*ALPHA
```

```
      W=VEL*SIN(ALPHA)
      U=VEL*COS(ALPHA)
      WB2V=0.
C
C  BEGINNING SECOND LOOP
C
      DO 2 J=1,9
C
C    INCREMENTING ROTATION RATE
C
      WB2V=WB2V+.1
      OMEGA=WB2V*2.*VEL/SPAN
      P=OMEGA*COS(ALPHA)
      R=OMEGA*SIN(ALPHA)
      A=P*P+R*R+.000000001
      B=-2.*U*R+2.*W*P
      C=U*U+W*W
      YM2=(-W+U*TAN(-A2))/(P+R*TAN(-A2))
      YM1=(-W+U*TAN(-A1))/(P+R*TAN(-A1))
      Y1 =(-W+U*TAN( A1))/(P+R*TAN( A1))
      Y2 =(-W+U*TAN( A2))/(P+R*TAN( A2))
      LA=777.
      LB=777.
      LC=777.
      LD=777.
      LE=777.
      UA=777.
      UB=777.
      UC=777.
      UD=777.
      UE=777.
      ELW=0.
C
C  COMPUTING THE LIMITS UA & LA FOR CNO=-.5 AND CNSA=1.0
C
      IF(-D2-YM2)11,11,100
   11 LA=-D2
      IF(D2-YM2)12,12,13
   13 UA=YM2
      GO TO 10
   12 UA=D2
   10 CONTINUE
```



```
      CNO=-.5
      CNSA=1.0
      CALL TESTS(UA,LA,ITOG)
  100 CONTINUE
C
C  COMPUTING THE LIMITS UB & LB FOR CNO=-1.6 AND CNSA=-3.
C
      IF(-D2-YM1)14,14,150
   14 IF(D2-YM2)150,150,16
   16 IF(-D2-YM2)17,17,18
   18 LB=-D2
      GO TO 19
   17 LB=YM2
   19 IF(D2-YM1)21,21,20
   21 UB=D2
      GO TO 15
   20 UB=YM1
   15 CONTINUE
      CNO=-1.6
      CNSA=-3.
      CALL TESTS(UB,LB,ITOG)
  150 CONTINUE
C
C  COMPUTING THE LIMITS UC & LC FOR CNO=0.0 AND CNSA=5.8
C
      IF(-D2-Y1)22,22,230
   22 IF(D2-YM1)230,230,24
   24 IF(-D2-YM1)25,25,26
   26 LC=-D2
      GO TO 27
   25 LC=YM1
   27 IF(D2-Y1)28,28,29
   28 UC=D2
      GO TO 23
   29 UC=Y1
   23 CONTINUE
      CNO=0.
      CNSA=5.8
      CALL TESTS(UC,LC,ITOG)
  230 CONTINUE
C
C  COMPUTING THE LIMITS UD & LD FOR CNO=1.6 AND CNSA=-3.
```

```
      C
            IF(-D2-Y2)30,30,310
         30 IF(D2-Y1)310,310,32
         32 IF(-D2-Y1)33,33,34
         34 LD=-D2
            GO TO 35
         33 LD=Y1
         35 IF(D2-Y2)36,36,37
         36 UD=D2
            GO TO 31
         37 UD=Y2
         31 CONTINUE
            CNO=1.6
            CNSA=-3.
            CALL TESTS(UD,LD,ITOG)
        310 CONTINUE
      C
      C  COMPUTING THE LIMITS UE & LE FOR CNO=.9 AND CNSA=1.0
      C
            IF(D2-Y2)380,380,39
         39 IF(-D2-Y2)40,40,41
         41 LE=-D2
            GO TO 42
         40 LE=Y2
         42 UE=D2
            CNO=.9
            CNSA=1.0
            CALL TESTS(UE,LE,ITOG)
        380 CONTINUE
      C
      C  PRINTING THE LIMITS TO THE DUMP
      C
            PRINT 700,LA,UA,LB,UB,LC,UC,LD,UD,LE,UE
      C
      C  COMPUTING THE NON-DIMENSI3NALIZED ROLLING MOMENT FROM THE ROLLING MOMENT.
      C  THIS WILL HAPPEN 9 TIMES AND THEN THE SUBROUTINE WILL EXIT INTO THE OUTER
      C  LOOP WHERE THE ANGLE OF ATTACK IS INCREASED BY 2 DEGREES.
      C
            CLW(J)=ELW/(QUE*AREA*SPAN)
          2 CONTINUE
      C
      C  PRINTING THE ANGLE OF ATTACK AND THE NON-DIMENSIONALIZED ROLLING MOMENT
```

SUBROUTINE SPIN 74/74 OPT=1 FTN 4.8+552 84/03/05. 09.

```
      C
            PRINT 700,ALPHD,(CLW(J),J=1,9)
          1 CONTINUE
      C
      C  THE SUBROUTINE WILL END AT ANGLE OF ATTACK EQUAL TO 90 DEGREES
      C
            RETURN
            END
```

```
     SYMBOLIC REFERENCE MAP (R=1)

ENTRY POINTS
    3  SPIN

VARIABLES       SN  TYPE          RELOCATION
    0  A            REAL              / /          407  ALPHA    REAL
  411  ALPHD        REAL                             0  AREA     REAL                F.P.
  404  A1           REAL                           405  A2       REAL
    1  B            REAL              / /            2  C        REAL                / /
   13  CHORD        REAL              / /            0  CH1      REAL                F.P.
    0  CH2          REAL              F.P.         430  CLW      REAL     ARRAY
   11  CNSA         REAL              / /           10  CNO      REAL                / /
    0  D1           REAL              F.P.           0  D2       REAL                F.P.
   12  ELW          REAL              / /            5  H        REAL                / /
    6  HFUSE        REAL              / /          410  I        INTEGER
    0  ITOG         INTEGER           F.P.         414  J        INTEGER
  377  LA           REAL                           400  LB       REAL
  401  LC           REAL                           402  LD       REAL
  403  LE           REAL                           415  OMEGA    REAL
    4  P            REAL              / /          406  QUE      REAL
  416  R            REAL                            14  RHO      REAL                / /
    0  RHO1         REAL              F.P.           7  SPAN     REAL                / /
    0  SPANW        REAL              F.P.         412  U        REAL
  423  UA           REAL                           424  UB       REAL
  425  UC           REAL                           426  UD       REAL
  427  UE           REAL                             0  VEL      REAL                F.P.
    3  W            REAL              / /          413  W82V     REAL
  420  YM1          REAL                           417  YM2      REAL
```

```
C      *****************************************************************
C      *  THIS SUBROUTINE CLASSIFIES LIMITS SENT FROM THE SUBROUTINE SPIN *
C      *  INTO THREE CATAGORIES:                                        *
C      *       1) LIMITS THAT BOTH FALL ON THE NEGATIVE WING            *
C      *       2) LIMITS THAT BOTH FALL ON THE POSITIVE WING            *
C      *       3) LIMITS THAT FALL ON EITHER SIDE OF THE FUSELAGE       *
C      *  THESE LIMITS ARE SENT TO SUBROUTINE PANEL TO COMPUTE THE      *
C      *  ROLLING MOMENT.                                               *
C      *****************************************************************
C
       SUBROUTINE TESTS(UA,LA,ITOG)
       REAL LA
       COMMON // A,B,C,W,P,H,HFUSE,SPAN,CNO,CNSA,ELV,CHORD,RHO
C
C   ROUTINE TO CATCH LIMITS THAT ARE LESS THAN THE DISTANCE TO THE PANEL.
C
       IF(UA.LE.HFUSE.AND.UA.GE.(-HFUSE)) UA=-HFUSE
       IF(LA.LE.HFUSE.AND.LA.GE.(-HFUSE)) LA=HFUSE
       IF(UA.LE.LA) GO TO 1101
C
C   IF DOUBLE PANEL NOT REQUESTED, GO TO 1190
C
       IF(ITOG.LE.1) GO TO 1190
C
C   >>> DOUBLE PANEL <<<
C
C   BOTH LIMITS POSITIVE
C
       IF(UA.GE.HFUSE.AND.LA.GE.HFUSE) GO TO 1100
C
C   BOTH LIMITS NEGATIVE
C
       IF(UA.LE.-HFUSE.AND.LA.LE.-HFUSE) GO TO 1100
C
C   LIMITS THAT COME THROUGH ARE THOSE THAT ARE NOT ON THE SAME PANEL.
C   THEREFORE, THE ROLL MOMENT MUST BE CALCULATED FOR EACH SIDE SEPARATELY.
C
C      **** NEGATIVE WING ****
C
       ORIGA=UA
       ORIGB=LA
       UA=-HFUSE
```



```
       IF(UA.EQ.LA) GO TO 1007
       CALL PANEL(UA,LA)
C
C      **** POSITIVE WING ****
C
 1007  UA=ORIGA
       LA=HFUSE
       IF(UA.EQ.LA) GO TO 1101
       CALL PANEL(UA,LA)
       LA=ORIGB
       GO TO 1101
C
C   >>> SINGLE PANEL <<<
C
 1190  ORIGA=UA
       ORIGB=LA
       IF(ITOG.GT.0) GO TO 1155
       IF(UA.GE.HFUSE.AND.LA.GE.HFUSE) GO TO 1100
       IF(UA.LE.-HFUSE.AND.LA.LE.-HFUSE) GO TO 1101
       LA=HFUSE
       IF(UA.LE.LA) GO TO 1101
       CALL PANEL(UA,LA)
       LA=ORIGB
       GO TO 1101
 1155  IF(UA.GE.HFUSE.AND.LA.GE.HFUSE) GO TO 1101
       IF(UA.LE.-HFUSE.AND.LA.LE.-HFUSE) GO TO 1100
       UA=-HFUSE
       IF(UA.LE.LA) GO TO 1101
       CALL PANEL(UA,LA)
       UA=ORIGA
       GO TO 1101
 1100  CALL PANEL(UA,LA)
 1101  RETURN
       END
```

SYMBOLIC REFERENCE MAP (R=1)

```
C       ****************************************************************
C       *  THIS SUBROUTINE PERFORMS THE INTEGRATION FOR THE ROLLING MOMENT *
C       *  OF THE PANEL.  TWO LIMITS ARE SENT HERE FROM SUBROUTINE         *
C       *  TESTS THAT ARE BOTH EITHER ON A POSITIVE PANEL OR A NEGATIVE    *
C       *  PANEL.  THE SUBROUTINE IS CALLED EACH TIME A NEW SET OF LIMITS  *
C       *  ON THE PANEL IS COMPUTED.                                       *
C       ****************************************************************
C
C  DEFINITION OF VARIABLES
C
C       M - THE SLOPE OF THE LINEAR EQUATION DESCRIBING WING TAPER
C
C       PHI   - INTEGRAL OF THE SQUARE ROOT OF AY**2 + BY + C
C       PHIY  - Y TIMES THE INTEGRAL OF THE SAME
C       PHIY2 - INTEGRAL OF Y SQUARED TIMES THE SAME
C       PHIY3 - INTEGRAL OF Y CUBED TIMES THE SAME
C
C       RADL  - SQUARE ROOT OF AY**2 + BY + C | Y THE LOWER LIMIT
C       RADU  - SQUARE ROOT OF AY**2 + BY + C | Y THE UPPER LIMIT
C
        SUBROUTINE PANEL(UA,LA)
        REAL LA
        COMMON // A,B,C,W,P,H,HFUSE,SPAN,CNO,CNSA,ELW,CHORD,RHO
C
        RADU=SQRT(A*UA*UA+B*UA+C)
        RADL=SQRT(A*LA*LA+B*LA+C)
C
        PHI=(UA*.5+B*.25/A)*RADU-(LA*.5+B*.25/A)*RADL
        PHI=PHI+((4.*A*C-B*B)/(8.*A*SQRT(A)))*(ALOG(A*UA*2.+B+2.*SQRT(A)*R
       1ADU)-ALOG(A*LA*2.+B+2.*SQRT(A)*RADL))
C
        PHIY=.333333*(RADU**3-RADL**3)/A-.5*B*PHI/A
C
        PHIY2=((6.*A*UA    -5.*B)*RADU**3-(6.*A*LA-5.*B)*RADL**3)/(24.*A*A)
        PHIY2=PHIY2-(4.*A*C-5.*B*B)*PHI/(16.*A*A)
C
C
        TEMP=(UA**2)/(5*A)-(7*B*UA)/(40*A*A)+(7*B*B)/(48*A**3)
        PHIY3=(TEMP-(2*C)/(15*A*A))*RADU**3
        TEMP1=(LA**2)/(5*A)-(7*B*LA)/(40*A*A)+(7*B*B)/(48*A**3)
        PHIY3=PHIY3-(TEMP1-(2*C)/(15*A*A))*RADL**3
        PHIY3=PHIY3-((7*B**3)/(32*A**3)-(3*C*B)/(8*A*A))*PHI
```



```
C
C  LIMITS THAT ARE ON THE POSITIVE SIDE OF THE WING MUST GO THROUGH THE
C  EQUATIONS AT 2000 INSTEAD OF THOSE FOLLOWING BECAUSE THE NEGATIVE WING
C  IS DESCRIBED BY A DIFFERENT EQUATION THAN THE POSITIVE SIDE.
C
         IF(UA.GE.HFUSE.AND.LA.GE.HFUSE) GO TO 2000
C
         ELW=ELW-W*RHO*CHORD*.5*CNSA*PHIY
       $-RHO*CNSA*.5*(P*CHORD+W*H)*PHIY2-P*RHO*H*CNSA*.5*PHIY3
         ELW=ELW-RHO*CHORD*.5*CNO*(.25*A*(UA**4-LA**4)+.333333*B*(UA**3-LA*
       1*3)+C*.5*(UA*UA-LA*LA))-
       $H*RHO*CNO*.5*(.2*A*(UA**5-LA**5)+.25*B*(UA**4-LA**4)+.333333*C*
       $(UA**3-LA**3))
         GO TO 2050
  2000 ELW=ELW-W*RHO*CHORD*.5*CNSA*PHIY
       $-RHO*CNSA*.5*(P*CHORD-W*H)*PHIY2+P*RHO*H*CNSA*.5*PHIY3
         ELW=ELW-RHO*CHORD*.5*CNO*(.25*A*(UA**4-LA**4)+.333333*B*(UA**3-LA*
       1*3)+C*.5*(UA*UA-LA*LA))+
       $H*RHO*CNO*.5*(.2*A*(UA**5-LA**5)+.25*B*(UA**4-LA**4)+.33333*C*
       $(UA**3-LA**3))
  2050 RETURN
         END
```

SYMBOLIC REFERENCE MAP (R=1)

ENTRY POINTS
3 PANEL

VARIABLES		SN TYPE	RELOCATION				
0	A	REAL	/ /	1	B	REAL	/ /
2	C	REAL	/ /	13	CHORD	REAL	/ /
11	CNSA	REAL	/ /	10	CNO	REAL	/ /
12	ELW	REAL	/ /	5	H	REAL	/ /
6	HFUSE	REAL	/ /	0	LA	REAL	F.P.
4	P	REAL	/ /	334	PHI	REAL	
335	PHIY	REAL		336	PHIY2	REAL	
340	PHIY3	REAL		333	RADL	REAL	
332	RADU	REAL		14	RHO	REAL	/ /

```
SUBROUTINE SPLIT      74/74  OPT=1                    FTN 4.8+552        84/03/09. 09.0

      C     ****************************************************************
      C     *  THIS SUBROUTINE SPLITS A WING INTO ITS INDIVIDUAL PANELS    *
      C     *  AND SENDS THE DIMENSIONS TO SUBROUTINE SPIN.  THE USER      *
      C     *  MUST SUPPLY IN THE MAIN PROGRAM VALUES FOR THE COMMON       *
      C     *  CHORD, THE TIP CHORD, AND TWO CHORDS IN-BETWEEN AT THE      *
      C     *  POINTS OF TAPER CHANGE, THE DISTANCES TO THESE CHORDS FROM *
      C     *  THE ORIGIN, THE AREA OF THE WING, THE WINGSPAN, AIR DEN-    *
      C     *  SITY, THE VELOCITY OF THE AIRCRAFT AND THE WIDTH OF THE     *
      C     *  FUSELAGE.                                                   *
      C     ****************************************************************
      C
            SUBROUTINE SPLIT(CHORD,CH1,CH2,CHTIP,D1,D2,SPAN,FUSE,RHO,AREA,VEL)
      C
            HFUSE=.5*FUSE
            HSPAN=.5*SPAN
            CALL SPIN(CHORD,CH1,HFUSE,D1,AREA,SPAN,RHO,VEL,2)
            CALL SPIN(CH1,CH2,D1,D2,AREA,SPAN,RHO,VEL,2)
            CALL SPIN(CH2,CHTIP,D2,HSPAN,AREA,SPAN,RHO,VEL,2)
            RETURN
            END
```

```
      SYMBOLIC REFERENCE MAP (R=1)

ENTRY POINTS
    3  SPLIT

VARIABLES       SN  TYPE            RELOCATION
    0  AREA         REAL                  F.P.        0  CHORD      REAL                 F.P.
    0  CHTIP        REAL                  F.P.        0  CH1        REAL                 F.P.
    0  CH2          REAL                  F.P.        0  D1         REAL                 F.P.
    0  D2           REAL                  F.P.        0  FUSE       REAL                 F.P.
  116  HFUSE        REAL                            117  HSPAN      REAL
    0  RHO          REAL                  F.P.        0  SPAN       REAL                 F.P.
    0  VEL          REAL                  F.P.

EXTERNALS            TYPE    ARGS
       SPIN                    9
```

REFERENCES

1. Taylor, Lawrence W., Jr.; and Pamadi, Bandu N.: Estimation of Parameters Involved in High Angle-of-Attack Aerodynamic Theory Using Spin-Flight Test Data. AIAA Atmospheric Flight Mechanics Conference, Gatlinburg, TN, August 1983.

2. Taylor, Lawrence W., Jr.: Applications of Parameter Estimation in the Study of Spinning Airplanes. AIAA 9th Atmospheric Flight Mechanics Conference, San Diego, CA, August 1982.

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4. Ralston, John N.: Rotary Balance Data for a Typical Single-Engine General Aviation Design for an Angle-of-Attack Range of 8 to 90. NASA Contractor Report 3246, March 1983.

5. Pamadi, Bandu N.: An Estimation of Aerodynamic Forces and Moments on an Airplane Model Under Steady State Spin Conditions. AIAA 9th Atmospheric Flight Mechanics Conference, San Diego, CA, August 1982.

α, deg.	Inner Panels	Middle Panels	Outer Panels	Total
0	-.00525	-.04320	-.1747	-.2236
2	-.00565	-.04095	-.1759	-.2225
4	-.00564	-.03505	-.1793	-.2200
6	-.00497	-.02923	-.1818	-.2160
8	-.00346	-.02434	-.1830	-.2108
10	-.00180	-.02017	-.1825	-.2045
12	-.000525	-.01620	-.1825	-.1992
14	-.000417	-.01220	-.1796	-.1922
16	-.000768	-.00824	-.1724	-.1814
18	-.000197	-.00451	-.1553	-.1600
20	-.000508	-.00239	-.1378	-.1407
22	-.000882	-.00198	-.1202	-.1231
24	-.000890	-.00394	-.1024	-.1072

Table 2

Rolling moment coefficient for the contributing trapezoidal panels on the main wing of the airplane shown in figure 3 at angles of attack from 0 to 24 degrees and $\Omega b/2V = 0.5$.

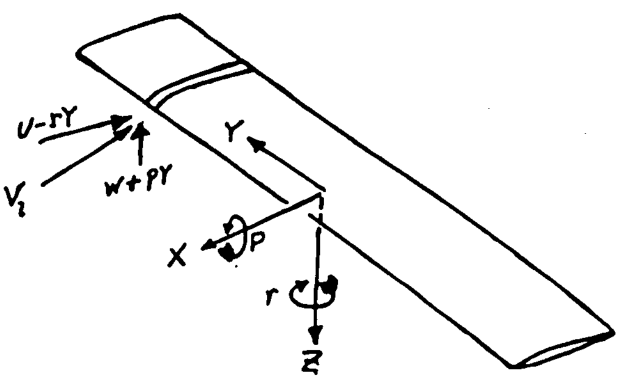

Figure 1. Schematic Sketch of a Spinning Wing

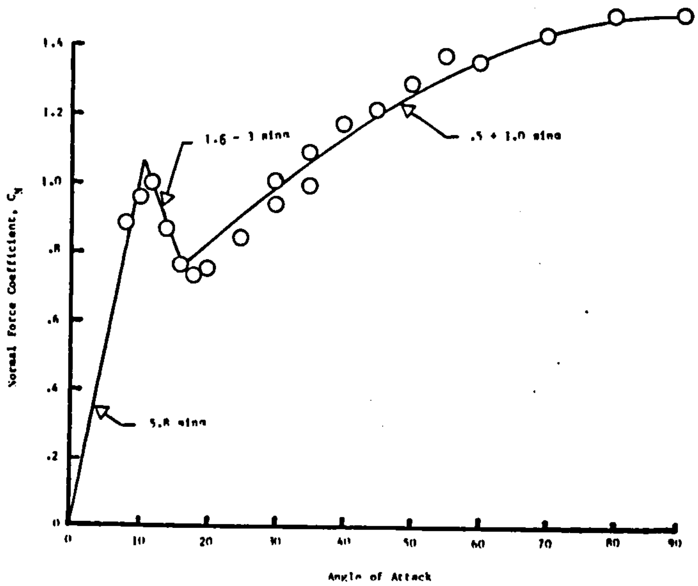

Figure 2. Model and measured normal force coefficient of reference 6.

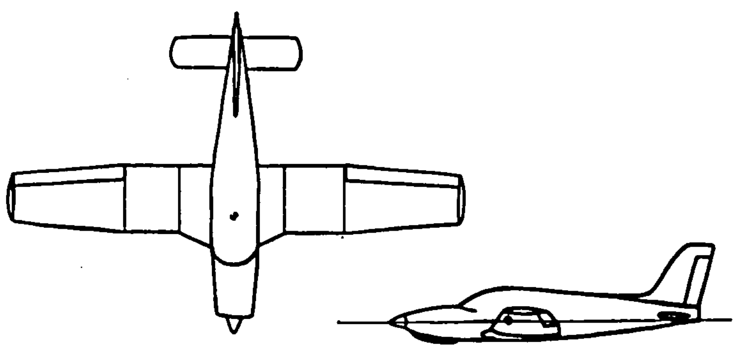

Figure 3. Drawing of 1/16-scale typical light airplane of reference 6. Note the 6 distinct panels on the main wing.

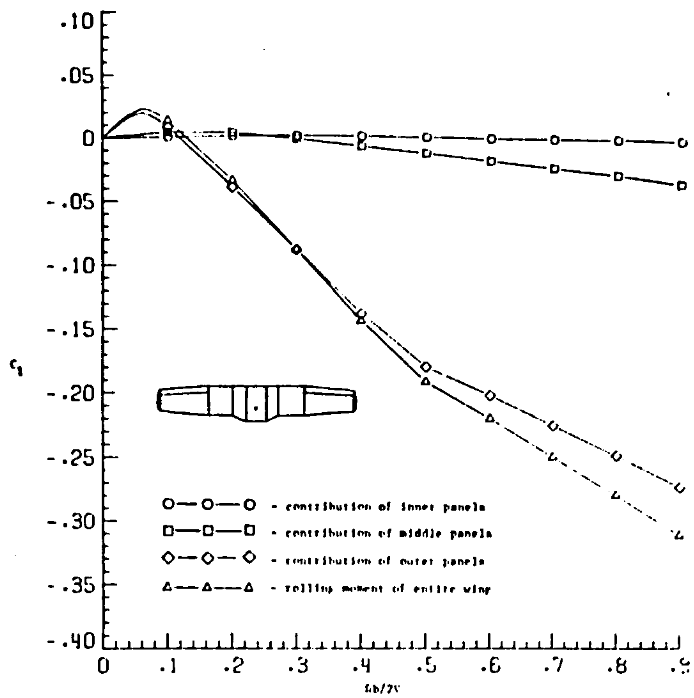

Figure 4. Contribution of wing panels to the rolling moment for 14 degrees angle of attack.

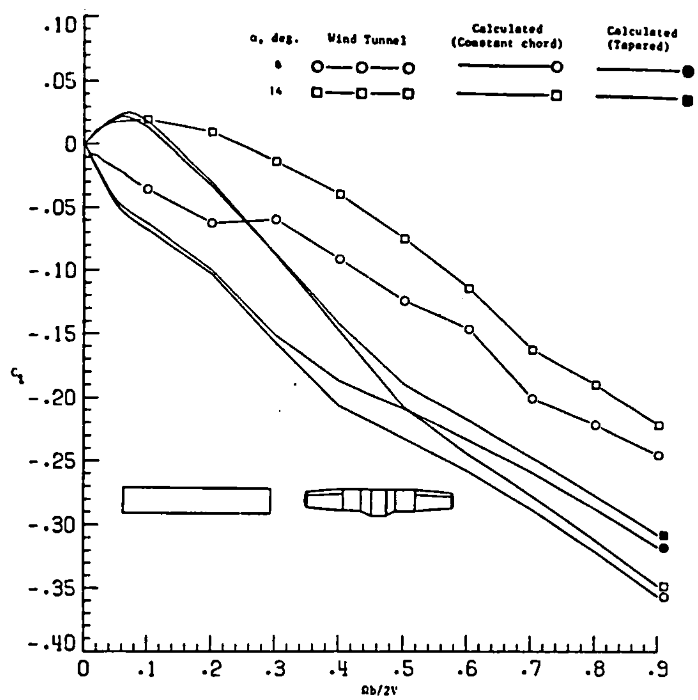

Figure 5. Effects of rotation rate on rolling moment coefficient for 8 and 14 degrees angle of attack.

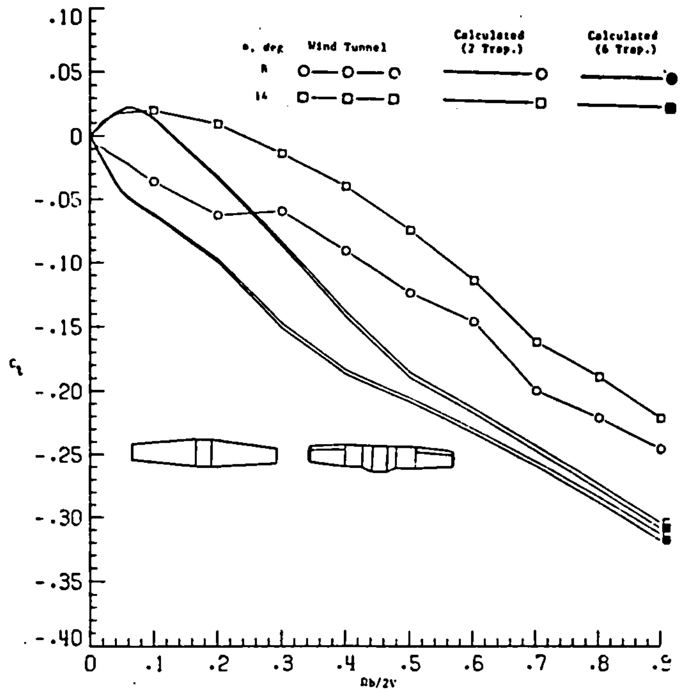

Figure 6. Effects of rotation rate on rolling moment coefficient for 8 and 14 degrees angle of attack.

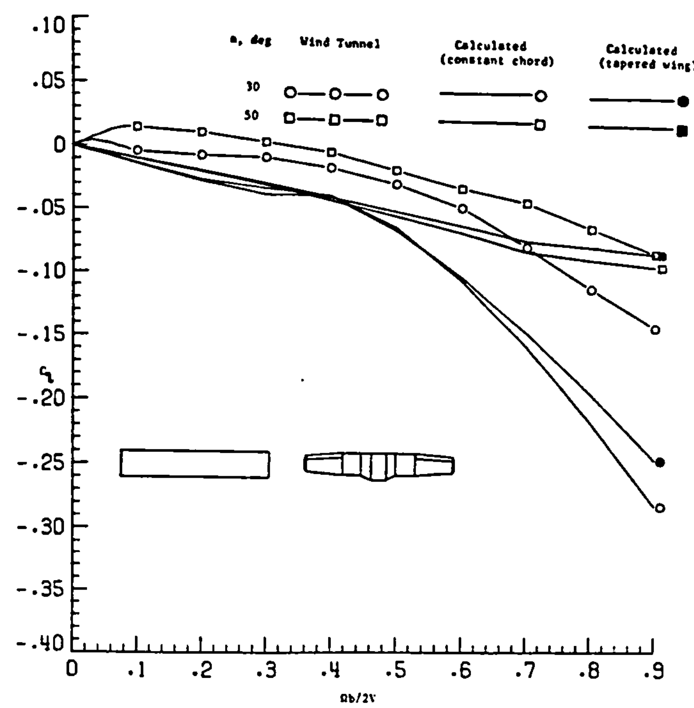

Figure 7. Effect of rotation rate on rolling moment coefficient for 30 and 50 degrees angle of attack.

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15. Supplementary Notes

16. Abstract

The application of aerodynamic theory for estimating the force and moments acting upon spinning airplanes is of interest. For example, strip theory has been used to generate estimates of the aerodynamic characteristics as a function of spin rate for wing-dominated configurations for angles of attack up to 90 degrees. This work, which had been limited to constant chord wings, is extended here to wings comprised of tapered segments. Comparison of the analytical predictions with rotary balance wind tunnel results shows that large discrepancies remain, particularly for those angles-of-attack greater than 40 degrees.

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