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

William A. Poppen, Jr.

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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
- \( c_0 \) 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
- \( C_{N_0} \) 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
- \( \alpha \) angle of attack
- \( \alpha_L \) local angle of attack
- \( \rho \) air density
- \( \Omega \) rotation rate
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.\textsuperscript{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.\textsuperscript{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."\textsuperscript{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 considering 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.

$$v_{\zeta}^2 = (u - ry)^2 + (w + py)^2$$

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

and

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

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

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

$$c = c_o + hy \quad \text{for} \quad y < 0$$
The equation for rolling moment for a single strip would be:

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

For the entire wing the rolling moment becomes:

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

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

\[ C_N(\alpha) = C_{N_0} + C_{N_s} \sin \alpha \]

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

\[ \Delta L = - \frac{\rho c_o C_{N_0}}{2} \int_{y_{\text{lower}}}^{y_{\text{upper}}} \left[ (u - ry)^2 + (w + py)^2 \right] y dy \]

\[ + \frac{h \rho c_o C_{N_s} \sin \alpha}{2} \int_{y_{\text{lower}}}^{y_{\text{upper}}} \sqrt{(u - ry)^2 + (w + py)^2} (w + py)y dy \]

After integrating,

\[ \Delta L = - \frac{\rho c_o C_{N_0}}{2} \frac{1}{4} \left[ A\left( y_{\text{upper}}^4 - y_{\text{lower}}^4 \right) + \frac{1}{3} B\left( y_{\text{upper}}^3 - y_{\text{lower}}^3 \right) \right]

\[ + \frac{1}{2} C\left( y_{\text{upper}}^2 - y_{\text{lower}}^2 \right) \frac{h \rho c_o C_{N_s} \sin \alpha}{2} \frac{1}{5} A\left( y_{\text{upper}}^5 - y_{\text{lower}}^5 \right) \]

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

\[- \frac{wpCOCNsina}{2} \int_{y_{lower}}^{y_{upper}} \sqrt{\phi} \ y \ dy - \frac{\rhoCNsina}{2} \left( pc_o \pm wh \right) \int_{y_{lower}}^{y_{upper}} \sqrt{\phi} \ y^2 \ dy \]

\[ = \frac{pohCNsina}{2} \int_{y_{lower}}^{y_{upper}} \sqrt{\phi} \ y^3 \ dy \]

where:

\[ A = p^2 + r^2 = \Omega^2 \]
\[ B = -2ur + 2wp = 0 \]
\[ C = u^2 + w^2 = V^2 \]
\[ \phi = Ay^2 + By + C \]

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

\[ - \left( 2Ay_{lower} \right) + B \sqrt{A^2 y_{lower}^2 + By_{lower} + C} \]

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

\[ \int_{y_{lower}}^{y_{upper}} \sqrt{\phi} \ y \ dy = \frac{1}{3A} \left[ \left( Ay_{upper}^2 + By_{upper} + C \right)^{3/2} \right] \]

\[ - \left( Ay_{lower}^2 + By_{lower} + C \right)^{3/2} \]
\[ - \frac{B}{2A} \int^{y_{\text{upper}}}_{y_{\text{lower}}} \sqrt{\phi} \, dy \]

\[ \int^{y_{\text{upper}}}_{y_{\text{lower}}} \sqrt{\phi} \, y^2 \, dy = \frac{6A y_{\text{upper}} - 5B}{24A^2} \left( \frac{A^2 y_{\text{upper}}}{\left( A^2 y_{\text{upper}} + B y_{\text{upper}} + C \right)^{3/2}} \right) \]

\[ - \frac{6A y_{\text{lower}} - 5B}{24A^2} \left( \frac{A^2 y_{\text{lower}}}{\left( A^2 y_{\text{lower}} + B y_{\text{lower}} + C \right)^{3/2}} \right) \]

\[ - \frac{4A C - 5B^2}{16A^2} \int^{y_{\text{upper}}}_{y_{\text{lower}}} \sqrt{\phi} \, dy \]

\[ \int^{y_{\text{upper}}}_{y_{\text{lower}}} \sqrt{\phi} \, y^3 \, dy = \left( \frac{y_{\text{upper}}^2}{5A} - \frac{7B y_{\text{upper}}}{40A^2} + \frac{7B^2}{48A^3} - \frac{2C}{15A^2} \right) \left( \frac{A^2 y_{\text{upper}}}{\left( A^2 y_{\text{upper}} + B y_{\text{upper}} + C \right)^{3/2}} \right) \]

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

\[ - \left( \frac{7B^3}{32A^3} - \frac{3CB}{8A^2} \right) \left( \frac{y_{\text{upper}}^2}{5A} - \frac{7B y_{\text{upper}}}{40A^2} + \frac{7B^2}{48A^3} - \frac{2C}{15A^2} \right) \left( \frac{A^2 y_{\text{upper}}}{\left( A^2 y_{\text{upper}} + B y_{\text{upper}} + C \right)^{3/2}} \right) \]

The terms in the normal force equation, \( C_{N_0} \) and \( C_{N_{\sin \alpha}} \), correspond to the local angle-of-attack ranges (see fig. 2) listed in Table 1 from reference 1:

<table>
<thead>
<tr>
<th>Angle-of-Attack</th>
<th>( C_{N_0} )</th>
<th>( C_{N_{\sin \alpha}} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>-164° to -16°</td>
<td>-.5</td>
<td>1.0</td>
</tr>
<tr>
<td>-16° to -10.5°</td>
<td>-1.6</td>
<td>-3.0</td>
</tr>
<tr>
<td>-10.5° to 10.5°</td>
<td>0</td>
<td>5.8</td>
</tr>
<tr>
<td>10.5° to 16°</td>
<td>1.6</td>
<td>-3.0</td>
</tr>
<tr>
<td>16° to 164°</td>
<td>.5</td>
<td>1.0</td>
</tr>
</tbody>
</table>

Table 1
The wing span locations having local angles of attack of -16, -10.5, 10.5 and 16 degrees are determined by:

\[
Y_{\text{boundary}} = \frac{w - u \tan(\alpha \text{ boundary})}{-p - r \tan(\alpha \text{ boundary})}
\]

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:

```
MAIN (WING DIMENSIONS)  MAIN (PANEL DIMENSIONS)
  ↓               ↓
OR
SPLIT (SPLITS WING)
  ↓
SPIN
  ↓
TESTS
  ↓
PANEL
  ↓
TAPER, LIMITS, RESULTS PRINTED

CLASSIFIES LIMITS

CALCULATIONS
```

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 extensions 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 analytical 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.
SUBROUTINE SPINL(CH1,CH2,DI,E1,E2,SPAN,DIHGT,YAW,ITOG)

! This subroutine generates non-dimensionalized values for the
! rolling moment of a symmetric aircraft wing panel at angles of
! attack from 0 to 90 degrees and at non-dimensionalized rotation
! rates from 0 to 0.2. The main rotor must define values for the
! boundary chords, the distances to these chords from the blades,
! the area of the wing, and an integer to specify whether or not the
! user wants to describe a negative panel, positive panel on both,
! the primary purpose of this subroutine is to compute the limits
! of integration on the panel.

DEFINITION OF VARIABLES

ALPHA - THE ANGLE OF ATTACK IN RADIANS
ALPHA1 - THE ANGLE OF ATTACK IN DEGREES
ALPHA2 - THE SECOND LOCALLERT ANGLE OF ATTACK = 10.5 DEGREES
CH1, CH2 - TWO LIMIT CHORDS OF THE PANEL
CLH - NON-DIMENSIONIZED ARRAY FOR ROLLING MOMENT
CHMA - CONSTANT USED IN NORMAL FORCE COEFFICIENT EQUATION
CHND - COEFFICIENT IN NORMAL FORCE COEFFICIENT EQUATION
DL1 - DISTANCE TO CH1
DL2 - DISTANCE TO CH2
E1L - ROLLING MOMENT
H - SLOPE OF THE LINEAR EQUATION DESCRIBING THE WING TAPER
ITOG - 0 FOR POSITIVE PANEL; 1 FOR NEGATIVE PANEL; 2 FOR BOTH
LAX, LAY, LIAX, LLYA - LOWER LIMITS OF INTEGRATION
UX1, UY1, UX2, UY2 - UPPER LIMITS OF INTEGRATION
OMEGA - ROTATION RATE

P - ROLL RATE
Q - YAW RATE
R - AIR DENSITY
S - SPAN - WING SPAN
T - COMPONENT OF VELOCITY
U - COMPONENT OF VELOCITY
V - COMPONENT OF VELOCITY
W - COMPONENT OF VELOCITY
X - NON-DIMENSIONALIZED ROTATION RATE
Y - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF 10.5 DEGREES
Z - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF 0.5 DEGREES
THETA - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF 10.5 DEGREES
THETAP - - SPAN LOCATION HAVING AN ANGLE OF ATTACK OF 0.5 DEGREES

DIMENSION (CLH(90))


COMMON /AX,EY,FMX,FMY,SPAN,CM,CH1,CH2,DL1,DL2,CHORD,RND

C BEGINNING FIRST LOOP
DO 1 I=I+46
C INCREMENTING ANGLE OF ATTACK BY 2 DEGREES
ALPHA=ALPHA+2,P37
ALPHA=ALPHA

C END OF MAIN PROGRAM
SUBROUTINE SPIN TH $O$ OPT $I$  

BEGINNING SECOND LOOP

INCREMENTING ROTATION RATE

COMPUTING THE LIMITS UA & LA FOR CHD=+9 AND CHSA=+1.0

COMPUTING THE LIMITS UA & LA FOR CHD=+9 AND CHSA=+3.0

COMPUTING THE LIMITS UA & LC FOR CHD=+9.0 AND CHSA=+9.0

COMPUTING THE LIMITS UC & LC FOR CHD=+9.0 AND CHSA=+9.0

COMPUTING THE LIMITS UD & LC FOR CHD=+3.0 AND CHSA=+3.0

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SUBROUTINE PANEL

DESCRIPTION OF VARIABLES:

\( W \) - THE LENGTH OF THE LINEAR EQUATION DESCRIBING WING TAPER
\( PH\) - INTEGRAL OF THE SQUARE ROOT OF \( x^2 + xy + c \)
\( PHT\) - INTEGRAL OF \( x^2 \) SQUARE TIMES THE SAME
\( PHT3\) - INTEGRAL OF \( x^2 \) CUBED TIMES THE SAME
\( PHL\) - SQUARE ROOT OF \( x^2 + xy + c \) \& \( y \) THE LOWER LIMIT
\( PHU\) - SQUARE ROOT OF \( x^2 + xy + c \) \& \( y \) THE UPPER LIMIT

ENTRY POINTS

<table>
<thead>
<tr>
<th>PANEL</th>
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<tbody>
<tr>
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<tr>
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</tr>
<tr>
<td>11</td>
</tr>
<tr>
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</tr>
<tr>
<td>14</td>
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SYMBOLIC REFERENCE MAP (y=1)

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<th>VARIABLES</th>
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<tr>
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</tr>
<tr>
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</tr>
<tr>
<td>14</td>
<td>REAL</td>
<td>14</td>
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</tbody>
</table>

- 15 -
SUBROUTINE SPLIT

***************************************************************************
C             THIS SUBROUTINE Splits A wing into its individual panels .
C             AND TIES THE DIMENSIONS TO SUBROUTINE FITE. THE USER.
C             MUST SUPPLY IN THE MAIN PROGRAM VALUES FOR THE CORRESPOND
C             POINTS OF TAPER CHANGE, THE DISTANCES TO THESE CHORDS FROM
C             FOILAGE.
***************************************************************************
C
C SUBROUTINE SPLIT(CHORD,CH,CHISP,DI,DIISP,FUSE,HUN,AREA,VEL)
C
15         NFUSE=SFUSE
           HISPAN=SFSPAN
           CALL SPINCHORD(CH,CHISP,DI,DIISP,FUSE,HUN,AREA,VEL)
           CALL SPINCH(CH,DI,DIISP,FUSE,HUN,AREA,VEL)
           RETURN
20         END

SYMBOLIC REFERENCE MAP (PP-13)

ENTRY POINTS

  SPLIT

<table>
<thead>
<tr>
<th>VARIABLES</th>
<th>SX</th>
<th>TYPE</th>
<th>RELLOCATION</th>
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<tr>
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<td>F.P.</td>
</tr>
<tr>
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<td>CHISP</td>
<td>REAL</td>
<td>F.P.</td>
</tr>
<tr>
<td>0</td>
<td>CH</td>
<td>REAL</td>
<td>F.P.</td>
</tr>
<tr>
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<td>D3</td>
<td>REAL</td>
<td>F.P.</td>
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<td>FUSE</td>
<td>REAL</td>
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<td>0</td>
<td>RNO</td>
<td>REAL</td>
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<tr>
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EXTERNALS

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REFERENCES


Table 2
Rolling moment coefficients 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 Ch/2Y = 0.5.

<table>
<thead>
<tr>
<th>No.</th>
<th>Inner Panels</th>
<th>Middle Panels</th>
<th>Outer Panels</th>
<th>Total</th>
</tr>
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<tbody>
<tr>
<td>0</td>
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<td>-.00320</td>
<td>-.0167</td>
<td>-.0324</td>
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<td>-.02005</td>
<td>-.1793</td>
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<td>-.07434</td>
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Figure 1. Schematic Sketch of a Spinning Wing
Figure 1. Normal force coefficient, $C_n$, vs. angle of attack.

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

Figure 3. Drawing of 1/18-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 coefficients for 8 and 14 degrees angle of attack.
Figure 6. Effects of rotation rate on rolling moment coefficients for 8 and 18 degrees angle of attack.

Figure 7. Effects of rotation rate on rolling moment coefficients for 30 and 30 degrees angle of attack.
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.