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This technical report has been reviewed and is approved for publication.

EMERIL D. LEFEVER, Chief
Integration Division
Directorate of Design Analysis
Deputy for Development Planning

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<td>Flying Qualities</td>
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<td>This report explains the methods involved in estimating aircraft moments of inertia for preliminary design purposes. Assumptions that were made for this procedure and the derivation of equations that evolved from these assumptions are included. An example using the method on the C-5A aircraft is shown. This procedure requires a knowledge of the major aircraft group weights, the location of major components (landing gear, avionics bay, etc.), geometry information, and inertias of some major subsystem items. Using</td>
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this data, the moments of inertia about the roll, pitch, and yaw axes are calculated as well as the roll-yaw cross-product of inertia.
FOREWORD

The purpose of this work was to establish a method to predict aircraft inertia suitable to preliminary design. It must be applicable to all types of military aircraft and be usable with the level of information normally available during preliminary design.

The material in this report was compiled as a part of the continuing methods development effort under project AFSD00X00ON, Flying Qualities Methodology and Development. The effort was accomplished within ASD/XRHI by Charles Lanham while a cooperative student under the direction of Wayne M. O'Connor.
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APPENDIX

REFERENCES
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<tr>
<th>TERM</th>
<th>DEFINITION</th>
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<tbody>
<tr>
<td>$b_1$</td>
<td>span of surface panel from root chord to tip*</td>
</tr>
<tr>
<td>$b_2$</td>
<td>span of surface panel from root chord to break*</td>
</tr>
<tr>
<td>$b_3$</td>
<td>span of surface panel from break chord to tip</td>
</tr>
<tr>
<td>$b_4$</td>
<td>span of wing fuel tank</td>
</tr>
<tr>
<td>$c_r$</td>
<td>length of surface root chord*</td>
</tr>
<tr>
<td>CREW&lt;sub&gt;cg&lt;/sub&gt;</td>
<td>perpendicular distance from YZ plane of the remote axes to the crew center of gravity.</td>
</tr>
<tr>
<td>$c_2$</td>
<td>length of surface chord at break</td>
</tr>
<tr>
<td>$c_3$</td>
<td>length of most inboard chord for wing fuel tank</td>
</tr>
<tr>
<td>$d$</td>
<td>average diameter of payload</td>
</tr>
<tr>
<td>$I$</td>
<td>moment of inertia of a group or component about the remote axes</td>
</tr>
<tr>
<td>$I_{cg}$</td>
<td>moment of inertia of total aircraft about its own center of gravity</td>
</tr>
<tr>
<td>$I_o$</td>
<td>moment of inertia of a group or component about its own center of gravity</td>
</tr>
<tr>
<td>$I_1$</td>
<td>moment of inertia of a surface about the leading edge of the root chord or break chord*</td>
</tr>
<tr>
<td>$I_{4}$</td>
<td>moment of inertia of a fuel tank about the leading edge of the most inboard tank chord</td>
</tr>
<tr>
<td>$l_c$</td>
<td>length of fuselage center section</td>
</tr>
<tr>
<td>$l_e$</td>
<td>length of nacelle (for buried engines just length of engine)</td>
</tr>
<tr>
<td>$l_f$</td>
<td>longitudinal length of fuselage tank</td>
</tr>
<tr>
<td>$l_n$</td>
<td>length of fuselage nose cone</td>
</tr>
<tr>
<td>$l_p$</td>
<td>length of fuselage tail cone</td>
</tr>
<tr>
<td>$l_v$</td>
<td>length of item used as a volume in fuselage</td>
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<thead>
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<tr>
<td>R</td>
<td>average fuselage radius ( \frac{S_{\text{max}}}{\pi} )</td>
</tr>
<tr>
<td>( R_e )</td>
<td>average nacelle radius (for buried engines use radius of engine)</td>
</tr>
<tr>
<td>( R_v )</td>
<td>average radius of item used as a volume in fuselage</td>
</tr>
<tr>
<td>( S_c )</td>
<td>wetted area of fuselage center section</td>
</tr>
<tr>
<td>( S_l )</td>
<td>external store or tank length</td>
</tr>
<tr>
<td>( S_n )</td>
<td>wetted area of fuselage nose cone</td>
</tr>
<tr>
<td>( S_r )</td>
<td>average radius of external tank or store</td>
</tr>
<tr>
<td>( S_t )</td>
<td>wetted area of fuselage tail cone</td>
</tr>
<tr>
<td>( t_b )</td>
<td>thickness of surface at break chord ( \left( \frac{r}{c} \right) )</td>
</tr>
<tr>
<td>( t_f )</td>
<td>thickness of wing fuel tank at most inboard chord</td>
</tr>
<tr>
<td>( t_{f0} )</td>
<td>thickness of wing fuel tank at most outboard chord</td>
</tr>
<tr>
<td>( t_r )</td>
<td>thickness of surface at root chord ( \left( \frac{r}{c} \right) )</td>
</tr>
<tr>
<td>( t_t )</td>
<td>thickness of surface at tip chord ( \left( \frac{r}{c} \right) )</td>
</tr>
<tr>
<td>( W_c )</td>
<td>weight of fuselage center section (structure only)</td>
</tr>
<tr>
<td>( W_{dc} )</td>
<td>total weight of contents to be distributed throughout the fuselage</td>
</tr>
<tr>
<td>( W_e )</td>
<td>total propulsion group weight divided by the number of engines</td>
</tr>
<tr>
<td>( W_{ff} )</td>
<td>weight of fuel in the fuselage</td>
</tr>
<tr>
<td>( W_{fw} )</td>
<td>weight of fuel in both wing fuel tanks</td>
</tr>
<tr>
<td>( W_h )</td>
<td>weight of total horizontal tail group</td>
</tr>
<tr>
<td>( W_i )</td>
<td>weight of both surface inboard of break</td>
</tr>
<tr>
<td>( W_n )</td>
<td>weight of fuselage nose cone (structure only)</td>
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<tr>
<td>( W_0 )</td>
<td>weight of both surfaces outboard of break</td>
</tr>
<tr>
<td>( W_p )</td>
<td>weight of one point mass</td>
</tr>
<tr>
<td>( W_{pc} )</td>
<td>total weight of point masses in the fuselage center section</td>
</tr>
<tr>
<td>( W_{pnc} )</td>
<td>total weight of point masses in nose and tail cones</td>
</tr>
<tr>
<td>( W_s )</td>
<td>weight of fuselage structure</td>
</tr>
<tr>
<td>( W_{st} )</td>
<td>weight of external fuel tank or store</td>
</tr>
<tr>
<td>( W_t )</td>
<td>weight of fuselage tail cone (structure only)</td>
</tr>
<tr>
<td>( W_v )</td>
<td>weight of total vertical tail group</td>
</tr>
<tr>
<td>( W_{v0} )</td>
<td>weight of one volume of mass</td>
</tr>
<tr>
<td>( W_w )</td>
<td>weight of total wing group</td>
</tr>
<tr>
<td>( X_{F1} )</td>
<td>distance from the wing fuel tank leading edge at most inboard tank chord to the longitudinal tank center of gravity</td>
</tr>
<tr>
<td>( X_{F2} )</td>
<td>perpendicular distance from YZ plane of the remote axes to leading edge of wing fuel tank most inboard chord</td>
</tr>
<tr>
<td>( X_P )</td>
<td>perpendicular distance from YZ plane of the remote axes to engine center of gravity</td>
</tr>
<tr>
<td>( X_{S1} )</td>
<td>distance from the surface leading edge at root chord to the longitudinal surface center of gravity</td>
</tr>
<tr>
<td>( X_{S2} )</td>
<td>(surface with leading and/or trailing edge break) distance from the surface leading edge of root chord to the longitudinal center of gravity for the surface section inboard of the break. *</td>
</tr>
<tr>
<td>( X_{S3} )</td>
<td>(surface with leading and/or trailing edge break) distance from the surface leading edge break chord to the longitudinal center of gravity for the surface section outboard of the break.</td>
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<tr>
<td>( X_{S4} )</td>
<td>perpendicular distance from YZ plane of the remote axes to leading edge of surface root chord</td>
</tr>
<tr>
<td>( X_{S5} )</td>
<td>perpendicular distance from YZ plane of the remote axes to leading edge of surface break chord</td>
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<td>$\bar{X}$</td>
<td>perpendicular distance from Z axis to aircraft center of gravity</td>
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<tr>
<td>$\overline{X}$</td>
<td>distance from a defined reference point to the surface longitudinal center of gravity</td>
</tr>
<tr>
<td>$Y_{F1}$</td>
<td>distance from the wing fuel tank most inboard chord to the spanwise tank center of gravity</td>
</tr>
<tr>
<td>$Y_{F1}$</td>
<td>perpendicular distance from the XZ plane of wing fuel tanks most inboard chord to the spanwise tank center of gravity. ((Y_{F1} \cos \theta))</td>
</tr>
<tr>
<td>$Y_{F2}$</td>
<td>perpendicular distance from XZ plane of the remote axes to most inboard chord of wing fuel tank</td>
</tr>
<tr>
<td>$Y_P$</td>
<td>perpendicular distance from XZ plane of the remote axes to engine center of gravity</td>
</tr>
<tr>
<td>$Y_{S1}$</td>
<td>distance along span of surface from root chord to center of gravity</td>
</tr>
<tr>
<td>$Y_{S1}$</td>
<td>perpendicular distance from XZ plane of surface root chord to spanwise center of gravity ((Y_{S1} \cos \theta))</td>
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<tr>
<td>$Y_{S2}$</td>
<td>distance along span of surface from root chord to center of gravity for inboard surface</td>
</tr>
<tr>
<td>$Y_{S2}$</td>
<td>perpendicular distance from XA plane of surface root chord to spanwise center of gravity for inboard surfaces ((Y_{S2} \cos \theta))</td>
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<tr>
<td>$Y_{S3}$</td>
<td>distance along span of surface from break chord to center of gravity for outboard surface</td>
</tr>
<tr>
<td>$Y_{S3}$</td>
<td>perpendicular distance from XA plane of surface break chord to spanwise center of gravity for outboard surfaces ((Y_{S3} \cos \theta))</td>
</tr>
<tr>
<td>$Y_{S4}$</td>
<td>perpendicular distance from XZ plane of the remote axes to the surface root chord</td>
</tr>
<tr>
<td>$Y$</td>
<td>distance from some reference point a surface spanwise center of gravity</td>
</tr>
<tr>
<td>$Z_b$</td>
<td>perpendicular distance from XY plane of the remote axes to fuselage centerline</td>
</tr>
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<td>DEFINITION</td>
</tr>
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<td>-------</td>
<td>-------------------------------------------------------------------------------------------------------------------------------------------</td>
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<tr>
<td>ZF</td>
<td>perpendicular distance from XY plane of the remote axes to wing fuel at most inboard chord</td>
</tr>
<tr>
<td>ZF2</td>
<td>$(YF1 \sin \theta)$ (needed only for wing internal tanks with anhedral or dihedral) perpendicular distance from the XY plane of the surface root chord to the vertical center of gravity</td>
</tr>
<tr>
<td>ZP</td>
<td>perpendicular distance from XY plane of the remote axes to engine center of gravity</td>
</tr>
<tr>
<td>ZS1</td>
<td>perpendicular distance from XY plane of the remote axes to root chord of surface</td>
</tr>
<tr>
<td>ZS3</td>
<td>$(YS1 \sin \theta)$ perpendicular distance from the XY plane of the surface root chord to the vertical surface center of gravity. *</td>
</tr>
<tr>
<td>ZS4</td>
<td>$(YS2 \sin \theta)$ perpendicular distance from the XY plane of the surface root chord to the vertical center of gravity of the surface panel inboard of the break. *</td>
</tr>
<tr>
<td>ZS5</td>
<td>$(YS3 \sin \theta)$ perpendicular distance from the XY plane of the surface break chord to the vertical center of gravity of surface panel outboard of the break</td>
</tr>
<tr>
<td>$\bar{Z}$</td>
<td>perpendicular distance from X axes to aircraft center of gravity</td>
</tr>
<tr>
<td>$\bar{z}$</td>
<td>distance from some reference point to a vertical surface center of gravity</td>
</tr>
<tr>
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<td>DEFINITION</td>
</tr>
<tr>
<td>-------</td>
<td>-------------------------------------------------------------------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>ZF</td>
<td>perpendicular distance from XY plane of the remote axes to wing fuel at most inboard chord</td>
</tr>
<tr>
<td>ZF2</td>
<td>(YF1 sin θ) (needed only for wing internal tanks with anhedral or dihedral) perpendicular distance from the XY plane of the surface root chord to the vertical center of gravity</td>
</tr>
<tr>
<td>ZF3</td>
<td>perpendicular distance from XY plane of the remote axes to engine center of gravity</td>
</tr>
<tr>
<td>ZS1</td>
<td>perpendicular distance from XY plane of the remote axes to root chord of surface</td>
</tr>
<tr>
<td>ZS3</td>
<td>(YS1 sin θ) perpendicular distance from the XY plane of the surface root chord to the vertical surface center of gravity. *</td>
</tr>
<tr>
<td>ZS4</td>
<td>(YS2 sin θ) perpendicular distance from the XY plane of the surface root chord to the vertical center of gravity of the surface panel inboard of the break, *</td>
</tr>
<tr>
<td>ZS5</td>
<td>(YS3 sin θ) perpendicular distance from the XY plane of the surface break chord to the vertical center of gravity of surface panel outboard of the break</td>
</tr>
<tr>
<td>Z</td>
<td>perpendicular distance from X axes to aircraft center of gravity</td>
</tr>
<tr>
<td>Z'</td>
<td>distance from some reference point to a vertical surface center of gravity</td>
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**LIST OF SYMBOLS (con. 'd)**

<table>
<thead>
<tr>
<th>TERM</th>
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<tbody>
<tr>
<td>$\Lambda_{L1}$</td>
<td>sweep of surface leading edge at root</td>
</tr>
<tr>
<td>$\Lambda_{L2}$</td>
<td>sweep of surface leading edge outboard of break</td>
</tr>
<tr>
<td>$\Lambda_{T1}$</td>
<td>sweep of surface trailing edge at root *</td>
</tr>
<tr>
<td>$\Lambda_{T2}$</td>
<td>sweep of surface trailing edge outboard of break</td>
</tr>
<tr>
<td>$\Lambda_{L3}$</td>
<td>sweep of wing fuel tank leading edge</td>
</tr>
<tr>
<td>$\Lambda_{T3}$</td>
<td>sweep of wing fuel tank trailing edge</td>
</tr>
<tr>
<td>$\rho$</td>
<td>density of fuel</td>
</tr>
<tr>
<td>$\theta$</td>
<td>angle in degrees between plane of surface and XY plane of remote axes (positive for dihedral, negative for anhedral)</td>
</tr>
</tbody>
</table>

* root chord can be defined as either theoretical or exposed

(see section II A)
SECTION I
INTRODUCTION

The purpose of this procedure is to determine the inertias of an aircraft at the preliminary design level so that the dynamic performance (flying qualities) can be examined. The purpose of analyzing dynamic performance at the preliminary design level is to insure adequate control surface sizing, develop control surface sizing rules for parametric design studies, and to determine the complexity of the flight control system necessary to adequately perform all required maneuvers. To do this, the procedure must be able to provide reasonably accurate estimates for different fuel and loading states.

The method described in this report has been incorporated into the ASD/XR Interactive Computer Design (ICAD) system. The flying qualities analysis portion of this system is described in Reference 1.
SECTION II

BACKGROUND

A. Basic Moment-of-Inertia Theory

Moment of inertia is the measure of resistance to angular acceleration, as mass is the measure of resistance to linear acceleration. Moment of inertia may be mathematically derived as follows.

If torque is expressed as the product of force and radius \( T = Fr \) and the following substitutions are made: \( F = ma \) and \( a = \alpha r \) then \( T = mar^2 \) or \( T = I\alpha \) where \( \alpha \) is the linear acceleration, \( \alpha \) is the angular acceleration, and \( m \) is the mass.

The term \( mr^2 \) is defined as the moment of inertia \( (I) \) and this equation may be written \( T = I\alpha \).

If a body of mass \( m \) is caused to rotate about a remote axis \( y \) the following relationship exists: \( I_y = mr^2 = m(x^2 + z^2) \).

However, since mass \( m \) not only offers resistance to rotation about the \( y \) axis but also offers resistance to rotation about its own centroidal axis, the total inertia of \( m \) about \( y \) is \( I_y = mr^2 + I_{oy} \) where \( I_{oy} \) is the inertia of \( m \) about its own centroidal axis.

When the full angular momentum equations are developed, there are nine \( I_o \) terms given in general by: \( I_{o x_i x_j} = \sum x_i x_j dm \) where \( x_i, x_j \) can be \( x, y, \) or \( z \). Since the symmetric terms are equal, e.g., \( I_{o xy} = I_{o yx} \), there are actually six independent moments of inertia.
For most aircraft problems, the vehicle is symmetric about the XZ plane. Although there are asymmetries in equipment locations which give rise to some non-zero values, it can be assumed for preliminary design purposes that \( I_{xy} \) and \( I_{yz} \) are zero. There are some configurations where this assumption obviously is not correct, such as skewed wings. For these aircraft the additional terms should be calculated. This method is limited to predicting the four remaining moments of inertia, \( I_x', I_y', I_z', \) and \( I_{xz'} \).

B. Method Description

There are three steps involved in obtaining these moments of inertia:

1) Allocate the total aircraft weight to six separate groups:
   
   a. wing group
   b. horizontal tail group
   c. vertical tail group
   d. fuselage group
   e. propulsion group
   f. additional items

   The level of detail of the weight breakdown given in Table 9 of Section II is adequate for determining the inertias. This allocation primarily involves distributing the subsystems throughout the aircraft without identifying the actual location of each wire, cable, line, or component. Since this is done on an "historical" or "accepted design practice" basis, adjustments may be needed for designs with unusual concepts or distributions.

2) Calculate the moment of inertia of each group about its own centroid and then transfer these inertias to a set of remote axes.

3) Locate the aircraft center of gravity, sum the inertias, and translate them back to the aircraft center of gravity to obtain the desired moments of inertia. The last two steps are described in detail in Section III.
C. Weight Allocation

Allocation of the total aircraft weight to the major groups is accomplished by a package of rules extracted from a structural weight estimation program (SWEEP) written by Rockwell International and from statistical data. The aircraft items are distributed as shown in Table 1.

<table>
<thead>
<tr>
<th>Item</th>
<th>Fraction in Fuselage</th>
<th>Fraction in Wing</th>
<th>Fraction Horiz</th>
<th>Fraction Vert</th>
<th>Fraction With Engine</th>
<th>Fraction With Items</th>
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</thead>
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<tr>
<td>Horiz tail structure</td>
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<tr>
<td>Nose gear</td>
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<td>-</td>
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<tr>
<td>Engine Nacelle &amp; Pylons</td>
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<td>-</td>
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<tr>
<td>Other structure</td>
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</tbody>
</table>

Note D = dependent on input location definition.
Items which need further discussion:

a. Fuel System - Distribute between the fuselage and wing group according to the fraction of fuel weight contained in each group.

b. Surface Controls: Summarized in table:

Table 2
SURFACE CONTROL WEIGHT ALLOCATION

<table>
<thead>
<tr>
<th>Configuration Code</th>
<th>Wing</th>
<th>Horizontal Tail</th>
<th>Vertical Tail</th>
<th>Fuselage Cockpit</th>
<th>Fuselage Distributed</th>
</tr>
</thead>
<tbody>
<tr>
<td>W, H, V*</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>0, 0, 0</td>
<td>0.532</td>
<td>0.128</td>
<td>0.124</td>
<td>0.038</td>
<td>0.178</td>
</tr>
<tr>
<td>0, 0, 1</td>
<td>0.457</td>
<td>0.110</td>
<td>0.247</td>
<td>0.033</td>
<td>0.153</td>
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<td>0, 1, 0</td>
<td>0.464</td>
<td>0.239</td>
<td>0.108</td>
<td>0.034</td>
<td>0.155</td>
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<td>0, 1, 1</td>
<td>0.406</td>
<td>0.209</td>
<td>0.220</td>
<td>0.029</td>
<td>0.136</td>
</tr>
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<td>1, 0, 0</td>
<td>0.608</td>
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<td>1, 1, 0</td>
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<td>0.029</td>
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<td>1, 0, 1</td>
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<td>0.094</td>
<td>0.213</td>
<td>0.028</td>
<td>0.131</td>
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<td>1, 1, 1</td>
<td>0.482</td>
<td>0.182</td>
<td>0.192</td>
<td>0.026</td>
<td>0.118</td>
</tr>
</tbody>
</table>

*W, wing 0 = fixed 1 = variable sweep
H, horizontal 0 = elevator type 1 = all moveable type
V, vertical tail 0 = rudder type 1 = all moveable type

c. Trapped fuel - Distribute between fuselage and wing group according to fraction of fuel weight contained in each group.

d. Air induction system - Add weight to fuselage group if engines are buried. Add to engine group if engines are podded.

e. Wing structure - If there is a wing carry-through structure, the weight should be added to the wing group - Otherwise only the exposed wing structure is in the wing group.
SECTION III
GROUP INERTIAS

Before the centroidal inertias \( I_0 \)'s) of each group can be calculated and then translated to the remote set of axes, a certain amount of component location and geometry information must be known. Everything should be referenced in accordance with the chosen set of remote axes (see sketch). The exact position of these axes can be varied, but to make the calculations easiest, the Z axis should be located at the nose of the aircraft.

Below is a list of additional components whose X, Y, and Z locations have to be determined if they are to be included. All geometry information that is needed is included with the discussion of each major group.

1) Main and nose landing gear
2) Auxiliary power unit
3) Air conditioning
4) Auxiliary gear
5) Gun
6) Crew
7) Weapons
8) Fuel system (Centroid of fuselage fuel tank)
9) Avionics bays
10) Radar
11) Furnishings & Equipment (centroid of total group or centroids of major items)
12) Photographic equipment
13) Other equipment
14) Liquid nitrogen
15) Miscellaneous items
16) Fuselage store and tank pylons
17) Fuselage external stores and tanks
18) Wing store and tank pylons
19) Wing external stores and tanks
20) Internal Payload

Using this information we can now proceed to calculate the moments of inertia of the separate groups about their own centroidal axes and translate these to the remote axes.

A. Surfaces

Wing, horizontal tail, and vertical tail groups are all common surfaces. To define the shape of the surface the normal planview (one side of wing and horizontal since they're symmetrical) is used. The equations are derived for a trapezoidal panel with the thickness varying linearly from root to tip. If a surface has edge or thickness breaks, it should be separated into inner and outer trapezoidal panels with the inertia of each calculated separately. The thickness is assumed constant as you go from leading to trailing edge and equal to the maximum for that section.
The inertia equations for this volumetric shape are derived (see Appendix Section 1) with the assumption that all surfaces lie in planes parallel to the XY plane of the remote axes.

To take into account the fact that surfaces don't always lie in planes parallel to the XY plane but usually have some anhedral or dihedral:

\[
I_{1y} = \frac{Wb^3}{V} \left\{ \left[ (t_r-t_t)(\frac{c}{4} + \frac{b\tan\Lambda_T}{5} - \frac{b\tan\Lambda_L}{5}) \right] \\
+ \left[ t_r \left( \frac{c}{3} + \frac{b\tan\Lambda_T}{4} - \frac{b\tan\Lambda_L}{4} \right) \right] \right\} 
\]

(1)

\[
I_{1y} = \frac{Wb}{V} \left\{ \left[ t_r \left( \frac{c^3}{3} + bc\tan\Lambda_T \left( \frac{c}{2} + \frac{b\tan\Lambda_T}{3} \right) + \frac{b^3}{12} \left( \tan^3\Lambda_T - \tan^3\Lambda_L \right) \right) \right] \right\} - \left[ (t_r-t_t) \left( \frac{c^3}{6} + bc\tan\Lambda_T \left( \frac{c}{2} + \frac{b\tan\Lambda_T}{4} \right) + \frac{b^3}{15} \left( \tan^3\Lambda_T - \tan^3\Lambda_L \right) \right) \right] 
\]

(2)

\[
I_{1z} = I_{1x} + I_{1y} 
\]

(3)

\[
V = b \left\{ t_r \left[ c + \frac{b}{2} \left( \tan\Lambda_T - \tan\Lambda_L \right) \right] - (t_r-t_t) \left[ \frac{c}{2} + \frac{b}{3} \left( \tan\Lambda_T - \tan\Lambda_L \right) \right] \right\} 
\]

(4)

The inertia equations for this volumetric shape are derived (see Appendix Section 1) with the assumption that all surfaces lie in planes parallel to the XY plane of the remote axes.

To take into account the fact that surfaces don't always lie in planes parallel to the XY plane but usually have some anhedral or dihedral:

\[
\text{True } I_{1y} = (I_{1y} \cos \theta + I_{1z} \sin \theta) 
\]

(5)

\[
\text{True } I_{1z} = (I_{1y} \sin \theta + I_{1z} \cos \theta) 
\]

(6)
I_{lx} is not affected by dihedral.

The product of inertia $I_{lxz}$ is non-zero only if there is some dihedral or anhedral.

$$I_{lxz} = \frac{W}{V} t_r \sin \Theta \left[ \frac{c_r^2 b^2}{4} + \frac{c_r b^3}{3} \tan \Lambda_T + \frac{b^4}{8} \left( \tan^2 \Lambda_T - \tan^2 \Lambda_u \right) \right] - \frac{W}{V} (t_r - t_e) \sin \Theta \left[ \frac{c_r^2 b^2}{6} + \frac{c_r b^3}{4} \right] \tan \Lambda_T + \frac{b^4}{10} \left( \tan^2 \Lambda_T - \tan^2 \Lambda_u \right)$$

(7)

These equations calculate the inertias for the entire wing, horizontal, or vertical tail as long as the total group is used.

If a wing does not have a carry-through structure, the exposed wing should be used and the symbols should be defined accordingly. Otherwise, a theoretical wing should be used. All horizontal and vertical tails should be defined with exposed parameters. The equations shown here calculate the inertias for the entire inboard or out-board surfaces (left and right) as long as the total weight for each was used and the symbols were defined correctly.

Table 3 shows how to define the general symbols used in all equations dealing with surfaces, for each separate surface.

**Table 3. Surface Symbols**

Before $I_\perp$ can be translated back to obtain the surface $I_\odot$, the centroids of the surfaces must be known. All longitudinal surface centroids can be found by a method from DATCOM (See Appendix Section 2) as long as the parameters $c$, $b$, and $\Lambda$ are again properly defined for each surface.
**Table 3. Surface Symbols**

<table>
<thead>
<tr>
<th>GENERAL SYMBOL</th>
<th>DEFINED SYMBOLS</th>
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</thead>
<tbody>
<tr>
<td><strong>WING (NO BREAK)</strong></td>
<td><strong>INBOARD SURFACE</strong></td>
</tr>
<tr>
<td>$A_L$</td>
<td>$A_{L1}$</td>
</tr>
<tr>
<td>$A_T$</td>
<td>$A_{T1}$</td>
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<td>$X_{S1}$</td>
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<tr>
<td>$Y$</td>
<td>$Y_{S1*}$</td>
</tr>
</tbody>
</table>

* If $Y_{S4} = 0$ and $\theta = 0$ Set these $= 0$.

** If $Y_{S4} = 0$ and $\theta = 0$ Set these $= 0$.

*** Not needed if $\theta = 0$.

Before $I_1$ can be translated back to obtain the surface $I_0$, the centroids of the surfaces must be known. All longitudinal surface centroids can be found by a method from DATCOM (See Appendix, Section 2) as long as the parameters $c$, $b$, and $A_L$ are again properly defined for each surface.

\[
X_{S1}, X_{S2}, X_{S3} = \frac{(-C_a^2 + C_b^2 + C_c C_b + C_c^2)}{3 (C_b + C_c - C_a)} \sqrt{\left( K_o \right)}
\]

where $C_a$ is the smallest of the following values: $c$, $b \tan A_L$,
$b \tan A_L + c$ $C_b$ is the intermediate value; $C_c$ is the largest value $K_o = .703$ for a wing $K_o = .771$ for horizontal or vertical tail

9b
All spanwise surface centroids are assumed to be at the spanwise surface center of volume. These centroids are needed only for exposed wing surfaces, outboard surfaces and for vertical tails since these all have inertias that need to be translated (see Appendix, Section 2).

\[ YS_1, YS_2, YS_3 = \frac{b^2}{V} \left[ (t_x \left( \frac{C}{2} + \frac{b}{3} (\tan \Lambda_T - \tan \Lambda_L) \right) - \left( t_x - t \right) \frac{C}{3} + \frac{b}{4} \right] (\tan \Lambda_T - \tan \Lambda_L) \]  

(9)

All vertical center of gravity distances for surfaces with \( \theta = 0 \) are negligible because of the small thickness of surfaces compared with their length and span. For surfaces with either anhedral or dihedral the vertical center of gravity \( (z) \) no longer lies in the XY plane of the root or break chord and can be calculated by:

\[ ZS_3 = YS_1 \sin \theta \]  

(10)

\[ ZS_4 = YS_2 \sin \theta \]  

(11)

\[ ZS_5 = YS_3 \sin \theta \]  

(12)

With the surface centroid location known, \( I_z \) can be translated to the centroid and then to the remote axes. For wings (no break), horizontal, and vertical surfaces, the \( I_z \) values are calculated by:

\[ I_x = I_{1x} = W(YS_1^2) + W(ZS_3^2) + W(YS_1 + YS_4)^2 + W(ZS_3 + ZS_1)^2 \]  

(13)

\[ I_y = I_{1y} = W(XS_1^2) + W(ZS_3^2) + W(XS_1 + XS_4)^2 + W(ZS_3 + ZS_1)^2 \]  

(14)

\[ I_z = I_{1z} = W(XS_1^2 + YS_1^2) + W(XS_1 + XS_4)^2 + W(YS_1 + YS_4)^2 \]  

(15)

\[ I_{xz} = I_{1xz} = W(XS_1) (ZS_3) + W(XS_1 + XS_4) (ZS_3 + ZS_1) \]  

(16)

For inboard surfaces:

\[ I_x = I_{1x} = W(YS_2^2 + ZS_4^2) + W(ZS_4 + ZS_1)^2 + W(YS_2 + YS_4)^2 \]  

(17)

\[ I_y = I_{1y} = W(XS_2^2 + ZS_4^2) + W(XS_2 + XS_4)^2 + W(ZS_1 + ZS_4)^2 \]  

(18)

\[ I_z = I_{1z} = W(XS_2^2 + YS_2^2) + W(XS_2 + XS_4)^2 + W(YS_2 + YS_4)^2 \]  

(19)

9c
For outboard surfaces:

\[ I_{xz} = \sum I_{x} - W (X52) (ZS4) + W (X2 + ZS4) (ZS4 + ZS5) \] (20)

\[ I_x = I_{ox} + W (Y3 + b_2 \cos \theta + YS4)^2 + W (ZS1 + b_2 \sin \theta + ZS5)^2 \] (21)

\[ I_y = I_{oy} = W (X5 + X3)^2 + W (ZS1 + b_2 \sin \theta + ZS5)^2 \] (22)

\[ I_z = I_{oz} + W (Y3 + b_2 \cos \theta + YS4)^2 + W (X5 + X3)^2 \] (23)

\[ I_{xz} = I_{oxz} - W (ZS3) (ZS5) + W (X5 + ZS4) (ZS5 + ZS1) \] (24)

B. FUSELAGE

The fuselage data needed for inertia calculations is:

\[ l_n, l_c, l_t, R, Z, W_s, W_p, XS4, W_v, R_v, l_v, W_p, W_p \]

Fuselage weight is divided into four areas:

1. Structure
2. Distributed contents
3. Volumes of mass
4. Point masses

1. Structure. Fuselage structural weight includes wing carry-through structure (if it was added to the fuselage group) and air induction system weight (if you have buried engine installations). This weight is assumed to be distributed between a conical nose shell, open-ended right-cylindrical shell, and a conical tail shell. For buried engine installations, the conical tail shell is neglected. Fuselage structure is distributed to each geometric shape in proportion to the surface area. (\( \frac{\text{weight}}{\text{area}} \) = constant). Air induction
system weight should be added to the open ended right-circular shell.

Moments of inertia of fuselage structure about the remote axes (see Appendix, Section 3) are given by:

\[ I_x = \frac{R^2}{2} (W_n + 2W_c + W_t) + W_s (Z_b)^2 \] 

(25)

\[ I_y = \frac{R^2}{4} (W_n + 2W_c + W_t) + \frac{1}{n} \left( \frac{W_n}{2} + W_c + W_t \right) + \frac{1}{c} \left( \frac{W_c + W_t}{3} \right) + \frac{1}{t} \frac{W_t}{6} \]

\[ + \frac{1}{n} (W_c + 2W_t) + \frac{2}{3} \frac{1}{t} W_c + \frac{2}{3} \frac{1}{t} W_t + W_s (Z_b)^2 \]

(26)

\[ I_z = I_x - W_s (Z_b)^2 \] 

(27)

\[ I_{xz} = W_n \left( \frac{3}{4} \frac{1}{n} Z_b \right) + W_c Z_b \left( \frac{1}{n} + \frac{1}{c} \right) + W_t Z_b \left( \frac{1}{n} + \frac{1}{c} + \frac{1}{t} \right) \] 

(28)

2) Distributed Contents. This consists of four main items: electrical system, instruments and navigation, hydraulics, and surface controls. They are assumed to be randomly spread throughout the fuselage from the cockpit to the leading edge of the horizontal tail in the shape of an open ended right-cylindrical shell. Moments of inertia of distributed contents about the remote axes (see Appendix, Section 4) are given by:

\[ I_x = W_{dc} R^2 + W_{dc} (Z_b)^2 \] 

(29)

\[ I_y = W_{dc} R^2 + \frac{1}{6} (X_{4-CREW c_g})^2 + W_{dc} (X_{4-CREW c_g} + CREW c_g)^2 + W_{dc} Z_b^2 \] 

(30)

\[ I_z = I_y - W_{dc} (Z_b)^2 \] 

(31)

\[ I_{xz} = \frac{W_{dc}}{2} (X_{4} + CREW c_g) (Z_b) \] 

(32)
Here $W_{dc}$ is defined as the weight of surface controls allocated to the fuselage + weight of electrical system allocated to the fuselage + weight of hydraulic system allocated to the fuselage + 30% of weight of instruments and navigation allocated to the fuselage.

3) **Volumes of Mass** consist of items such as the fuel system in the fuselage, the avionics bay, and furnishings. It is left to the user to decide whether to use these as volumes or point masses because of the variability of the items. Either a cylindrical shell or a solid rectangular shape can be used. Moments of inertia of these volumes about the remote axes (see Appendix, Section 5) are given by:

Cylindrical shell -

\[
I_x = W_{vo} R_v^2 + W Z^2
\]

\[
I_y = W_{vo} \frac{(R_v^2 + 1/v^2)}{2} + W_{vo} (X^2 + Z^2)
\]

\[
I_z = W_{vo} \frac{(R_v^2 + 6/v^2)}{2} + W_{vo} X^2
\]

\[
I_{xz} = W_{vo} XZ
\]

Rectangular solid -

\[
I_x = \frac{W_{vd}}{12} (2R_v^2 + 2R_v^2) + W_{vo} Z^2
\]

\[
I_y = \frac{W_{vo}}{12} (1/v^2 + 2R_v^2) + W_{vo} (X^2 + Z^2)
\]

\[
I_z = \frac{W_{vo}}{12} (1/v^2 + 2R_v^2) + W_{vo} X^2
\]

\[
I_{xz} = \frac{W_{vo}}{} XZ
\]
4. **Point Masses.** Each point mass is generally considered separately for calculating inertias. Aggregate small items, such as troop provisions in cargo aircraft are handled differently. For roll \( I_x \) inertia the point mass total weight for aggregate items is distributed between a solid cone and a solid right circular cylinder. All aggregate point masses located in the nose or tail cone of the fuselage are put in the solid cone and all point masses in the center section are put in the solid right circular cylinder. For \( I_y \) and \( I_z \) they are lumped at some average location. The moment of inertia of point masses about the remote axes (see Appendix, Section 6) are given by:

\[
I_x = EW_p (Y^2 + Z^2)
\]

or

\[
I_x = \frac{W_{pc}}{2} R^2 + \frac{3}{10} W_{pnc} R^2 + (W_{pc} + W_{pnc}) (Z_b)^2
\]  

(41)

\[
I_y = EW_p (X^2 + Z^2)
\]  

(42)

\[
I_z = EW_p (X^2 + Y^2)
\]  

(43)

\[
I_{xz} = EW XZ
\]  

(44)

Items usually considered as point masses:

*Main and nose landing gear*

*Auxiliary power unit*

*Air Conditioning*

*Auxiliary gear*

*Gun*

*Crew*

*Armament*

*Surface controls assigned to cockpit*

*Radar*
Photographic
70% of instruments and navigation weight (locate at cockpit)

Other equipment

Liquid nitrogen

Miscellaneous items

C. Propulsion

The propulsion data needed for inertia calculations is: $W_e$, $R_e$, $l_e$, $X_0$, $Y_0$, $Z_0$, $I_o$ of engines.* The total group weight is divided by the number of engines; this is the weight of each engine and accessories. If the $I_o$'s of the engines are not known, they can be approximated by using a solid cylinder (see Appendix, section 9). The moments of inertia of each engine about the remote axes are given by:

$I_o$ approximated:

$I_x = \frac{W_e R_e^2}{2} + W_e (Y_0^2 + Z_0^2)$  \hspace{1cm} (45)

$I_y = \frac{W_e (3R_e^2 + l_e^2)}{12} + W_e (X_0^2 + Y_0^2)$  \hspace{1cm} (46)

$I_z = \frac{W_e (3R_e^2 + l_e^2)}{12} + W_e (X_0^2 + Y_0^2)$  \hspace{1cm} (47)

$I_{xz} = W_e (X_0) (Z_0)$  \hspace{1cm} (48)

$I_o$ input:

$I_x = I_{ox} + W_e (Y_0^2 + Z_0^2)$  \hspace{1cm} (49)

$I_y = I_{oy} + W_e (X_0^2 + Z_0^2)$  \hspace{1cm} (50)

$I_z = I_{oz} + W_e (X_0^2 + Y_0^2)$  \hspace{1cm} (51)

$I_{xz} = I_{oxz} + W_e (X_0) (Z_0)$  \hspace{1cm} (52)

---

*R_e and l_e are not needed if inertias are given.
D. Internal Fuel

1. Wing fuel tanks

Internal wing fuel is defined in the same manner as surfaces because of the wing fuel tank shape being similar to a surface. We assume the wing tank is full of fuel and has a constant density. The $I_1$ equations for surfaces (see Appendix, Section 2) can now be used as long as we substitute $\rho$ (density of fuel) for $\overline{W}$.

\[
I_{4x} = 2b^3 \rho \left[ -\left( \frac{t_r - t_c}{4} \right) \left( c + \frac{b}{5} (\tan \Lambda_T - \tan \Lambda_L) \right) + \frac{t_r}{3} \left( \frac{c}{4} + \frac{b}{4} (\tan \Lambda_T - \tan \Lambda_L) \right) \right] \quad (53)
\]

\[
I_{4y} = 2b^3 \rho \left[ \frac{t_r}{3} \left( \frac{c^3}{6} + b \tan \Lambda_T \left( \frac{c}{3} + \frac{b}{4} \tan \Lambda_L \right) \right) + b^3 \left( \frac{\tan^3 \Lambda_T - \tan^3 \Lambda_L}{12} \right) \right] + \frac{t_r}{4} \left( \frac{c^3}{6} + b \tan \Lambda_T \left( \frac{c}{3} + \frac{b}{4} \tan \Lambda_L \right) \right) \quad (54)
\]

\[
I_{4z} = I_{4x} + I_{4y} \quad (55)
\]

Again realizing wing fuel tanks may be at some dihedral angle:

\[
\text{TRUE } I_{4y} = (I_{4y} \cos \theta + I_{4z} \sin \theta) \quad (56)
\]

\[
\text{TRUE } I_{4z} = (I_{4y} \sin \theta + I_{4z} \cos \theta) \quad (57)
\]

$I_{4x}$ is not affected by dihedral. If there is dihedral, the product of inertia of the fuel is given by:

\[
I_{4xz} = 2\rho \sin \theta \left[ \frac{t_r}{12} \left( \frac{c^2 b^2}{12} \tan \Lambda_T + b^4 \left( \frac{\tan^2 \Lambda_T - \tan^2 \Lambda_L}{40} \right) \right) \right] + \frac{t_r}{6} \left[ \frac{c^2 b^2}{4} \tan \Lambda_T + b^4 \left( \frac{\tan^2 \Lambda_T - \tan^2 \Lambda_L}{10} \right) \right] \quad (58)
\]

These equations calculate the total $I_4$ for total wing fuel (right and left wing fuel tanks).
TABLE 4
WING INTERNAL FUEL TANK SYMBOLS

<table>
<thead>
<tr>
<th>General Symbol</th>
<th>Defined Symbol</th>
</tr>
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<tbody>
<tr>
<td>( \Lambda_L )</td>
<td>( \Lambda_{L3} )</td>
</tr>
<tr>
<td>( \Lambda_T )</td>
<td>( \Lambda_{T3} )</td>
</tr>
<tr>
<td>( b )</td>
<td>( b_4 )</td>
</tr>
<tr>
<td>( c )</td>
<td>( c_3 )</td>
</tr>
<tr>
<td>( \tau )</td>
<td>( \tau_f )</td>
</tr>
<tr>
<td>( \tau_t )</td>
<td>( \tau_{f_0} )</td>
</tr>
<tr>
<td>( \rho )</td>
<td>( JF4 = 0.2814 \text{ in} )</td>
</tr>
<tr>
<td>( X )</td>
<td>( X_F1 )</td>
</tr>
<tr>
<td>( Y )</td>
<td>( Y_F1 )</td>
</tr>
<tr>
<td>( Z )</td>
<td>( Z_F2 )</td>
</tr>
</tbody>
</table>

Again, as in the case for surfaces, the centroid of the fuel tank must be calculated. The centroid is assumed to be located at the center of volume of the tank. (See Appendix, Section 7.)

\[
X_{F1} = \frac{b}{v} \left\{ \left[ (\tau_r \left( \frac{c}{2} + \frac{b \tan \Lambda_T}{2} + \frac{b^2}{6} \left( \tan^2 \Lambda_T - \tan^2 \Lambda_L \right) \right] - \left[ (\tau_r - \tau_t) \left( \frac{c}{4} + \frac{c b \tan \Lambda_T}{3} + \frac{b^2}{8} \left( \tan^2 \Lambda_T - \tan^2 \Lambda_L \right) \right] \right) \right\} (59)
\]

\[
Y_{F1} = \frac{b^2}{v} \left[ \tau_r \left( \frac{c}{2} + \frac{b}{3} \left( \tan \Lambda_T - \tan \Lambda_L \right) \right) - \left[ (\tau_r - \tau_t) \left( \frac{c}{3} + \frac{b}{4} \right) \right] \right\} (60)
\]

The vertical fuel tank centroid is zero unless the wing has anhedral or dihedral, in which case:

\[ Z_{F2} = Y_{F1} \sin \theta \]

The fuel tank \( I_4 \) can be translated to obtain \( I_0 \) and then \( I \) by:

\[
I_x = I_{4x} - \rho_{FW} Y_{F1}^2 - \rho_{FW} (Z_{F2})^2 + \rho_{FW} (Y_{F1} + Y_{F2})^2 + \rho_{FW} (Z_{F2} + ZF)^2 \quad (61)
\]

16
\[
I_y = I_{4y} - W_{fw} (X_F^2 + Y_F^2) + W_{fw} (X_F + Y_F)^2 + W_{fw} (Z_F^2 + Z_F)^2
\] (62)

\[
I_z = I_{4z} - W_{fw} (X_F^2 + Y_F^2) + W_{fw} (X_F + Y_F)^2 + W_{fw} (Y_F + Y_F)^2
\] (63)

\[
I_{xz} = I_{4xz} - W_{fw} (X_F)(Z_F) + W_{fw} (X_F + Y_F)(Z_F + Z_F)
\] (64)

If there is more than one internal wing fuel tank, this total procedure can be used for each subsequent tank in the same manner.

2. Fuselage Fuel Tanks

Fuselage internal fuel is assumed to be in the shape of a solid right cylinder. These inertia calculations are to be used in aircraft flying qualities studies; only short period rolling motions will be examined, and the fuel will not attain any appreciable rotational motion during these maneuvers. The rolling inertia of the fuel about its own axis is therefore assumed to be zero. The moments of inertia of fuselage internal fuel about the remote axes (see Appendix, Section 8) is given by:

\[
I_{ox} = 0
\] (65)

\[
I_{oy} = \frac{W_{ff}}{12 \pi p_1 f} \left( \frac{W_{ff}}{12 \pi p_1 f} + 1_f \right)^2 + W_{ff} (X_f^2 + Z_f^2)
\] (66)

\[
I_{oz} = \frac{W_{ff}}{12 \pi p_1 f} \left( \frac{W_{ff}}{12 \pi p_1 f} + 1_f \right)^2 + W_{ff} X_f^2
\] (67)

\[
I_{oxz} = W_{ff} X_Z
\] (68)

E. Payload

1. Transport Payload

Payload inertia is estimated by using a solid rectangular mass or series of masses as were the volumes of mass in the fuselage. Moments of inertia for payload about the remote axes (see Appendix 5, Section 9) are given by:

\[
I_x = \frac{W}{12} (d_p^2 + d_p^2) + WZ^2
\] (69)
\[ I_y = \frac{W}{12} (l_p^2 + d_p^2) + W (x^2 + z^2) \]  
(70)

\[ I_z = \frac{W}{12} (l_p^2 + d_p^2) + WX^2 \]  
(71)

\[ I_{xz} = WXZ \]  
(72)

2. Internal Weapons

It is assumed that the inertia and locations of these items are given.

F. Additional Items

1. External Stores and Tanks

Wing and fuselage store and tank pylons are to be used as point masses to calculate \( I_x \), \( I_y \), and \( I_z \). External wing and fuselage tanks and stores can be approximated by shells and solid right cylinders (see Appendix, Section 9) depending on whether the tanks are full or empty.

**TANKS:**

\[ I_z = \frac{W_{st}}{2} (SR^2 + SL^2) + W_{st} (Y^2 + X^2) \]  
(73)

\[ I_y = \frac{W_{st}}{2} (SR^2 + SL^2) + W_{st} (X^2 + Z^2) \]  
(74)

\[ I_x = W_{st} SR^2 + W_{st} (Y^2 + Z^2) \]  
(75)

\[ I_{xz} = W_{st} XZ \]  
(76)

**STORES:**

\[ I_x = \frac{W_{st}}{2} SR^2 + W_{st} (Y^2 + Z^2) \]  
(77)

\[ I_y = \frac{W_{st}}{12} (3 SR^2 + SL^2) + W_{st} (X^2 + Z^2) \]  
(78)

\[ I_z = \frac{W_{st}}{12} (3 SR^2 + SL^2) + W_{st} (X^2 + Y^2) \]  
(79)

\[ I_{xz} = W_{st} XZ \]  
(80)
G. Total Aircraft Inertias

The total inertia about the remote axes from all groups are now summed to achieve a complete inertia for the total aircraft. For this to be translated back to the center of gravity of the total vehicle, the center of gravity location must first be calculated. By definition:

\[
\bar{X} = \frac{\sum WX - \sum WZ}{\sum W} = \bar{Z} = \frac{\sum WZ}{\sum W} = 0
\] (81)

where \(X\) and \(Z\) are distances to the item or group centroid. All item and group weights and distances to the remote axes are already known, except for the fuselage structure longitudinal distances. These are given by:

\[
WX \text{ nose cone} = W_n \left(\frac{2}{3} l_n\right)
\] (82)

\[
WX \text{ center} = W_c \left(l_n + \frac{1}{2} l_c\right)
\] (83)

\[
WX \text{ tail cone} = W_t \left(l_n + l_c + \frac{1}{3} l_t\right)
\] (84)

\(W\) should equal the total aircraft weight. \(\bar{Y}\) is zero because of the already assumed symmetry of the aircraft. The translation of the total inertias to the aircraft center of gravity is then:

\[
I_{cgx} = I_x - WZ^2
\] (85)

\[
I_{cgy} = I_y - W(\bar{Z}^2 + \bar{X}^2)
\] (86)

\[
I_{cgz} = I_z - WX^2
\] (87)

\[
I_{cgz} = I_{xz} - WXZ
\] (88)

Results from the use of this method on various types of aircraft is given in Table 5. Data on these aircraft were obtained from References 3 - 6.
### Table 5. Summary Comparison

<table>
<thead>
<tr>
<th></th>
<th></th>
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<tr>
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<td></td>
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<td>129.0</td>
<td>747.2</td>
<td>762.2</td>
<td>822.3</td>
<td>835.6</td>
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<td>Air Superiority Takeoff</td>
<td>166.5</td>
<td>190.8</td>
<td>824.0</td>
<td>829.2</td>
<td>946.0</td>
<td>951.8</td>
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<td><strong>C-5A</strong></td>
<td></td>
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<td></td>
<td></td>
<td></td>
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<tr>
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<td>54246</td>
<td>101486</td>
<td>98853</td>
<td>146944</td>
<td>140694</td>
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<td>266755</td>
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<td>Max Fuel</td>
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<tr>
<td>Weight Empty</td>
<td>168</td>
<td>203</td>
<td>413</td>
<td>356</td>
<td>580</td>
<td>543</td>
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<tr>
<td>Ferry Mission Gross</td>
<td>293</td>
<td>279</td>
<td>604</td>
<td>608</td>
<td>817</td>
<td>891</td>
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<td><strong>B-52G</strong></td>
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<tr>
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<td>19380</td>
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<tr>
<td>Design Gross Weight</td>
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<td>64142</td>
<td>39520</td>
<td>37350</td>
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<td>92696</td>
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<tr>
<td><strong>Average Error (%)</strong></td>
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<td>5.7</td>
<td>6.8</td>
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</tr>
</tbody>
</table>

Average percent error of actual versus calculated values is 8.6%
III. Sample Problem: C-5A

Moments of inertia are first calculated for operating weight empty and then for basic flight design weight with maximum fuel. All units are pounds and inches. Basic geometry and weight data are given in Figures 1 and 2 and Tables 6 and 7. This data was taken from Reference 3.
Figure 2. C-5A Three-View
Table 6. C-5A Weight Statement

<table>
<thead>
<tr>
<th>Item Description</th>
<th>Weight (lb)</th>
</tr>
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<tbody>
<tr>
<td>Basic Structure</td>
<td>82044.8</td>
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<tr>
<td>Center Section</td>
<td></td>
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<tr>
<td>Intermediate Panel</td>
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</tr>
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<td>Outer Panel</td>
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</tr>
<tr>
<td>All. W. Total</td>
<td>21035.9</td>
</tr>
<tr>
<td>Flaps - Leading Edge</td>
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<tr>
<td>Slats - Trailing Edge</td>
<td>2474.2</td>
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<td>Tail Group</td>
<td></td>
</tr>
<tr>
<td>Stabilizer</td>
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<td>Elevator</td>
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<td>Rudder</td>
<td>675.1</td>
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<tr>
<td>Body Group (Including Manufacturing Variations)</td>
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<td>FUSELAGE</td>
<td>51948.8</td>
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<td>Gear Pads (Including MG and NKG Doors)</td>
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<tr>
<td>Cargo Floor (Including Tiedown Rings &amp; Receptacles)</td>
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<td><strong>TILLOON DEVICES</strong></td>
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<td><strong>PAYLOAD</strong></td>
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<td><strong>DELIVERABLE CARGO</strong></td>
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<td><strong>BAGGAGE</strong></td>
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<td><strong>AFT TOE RAMPS</strong></td>
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<td><strong>WATER - CREW</strong></td>
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26
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<th>Length - Overall (ft)</th>
<th>Height - Overall (ft)</th>
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| Loadable Cargocarryable Cargocarryable Nacelles 7-side Cutout |
|-----------------------|--------------------|-----------------|-----------------|-----------------|
| Fwd Ramp | Aft Ramp | Fwd Compl | Aft Compl | Fuselage | Main Tail | Tail Tip |
| 9.0      | 11.0     | 12.0      | 14.0      | 230.0     | 26.0      | 26.0     |

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<tr>
<th>Depth - Max (ft)</th>
<th>Width - Max (ft)</th>
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<th>Fuselage Volume (cu ft)</th>
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<th>Sweepback at 25° Chord (degrees)</th>
<th>Chord at Planform Break (in)</th>
<th>Chord at Planform Break (in) - Length</th>
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<th>Theoretical Tip Chord (in) - Length</th>
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<th>E. Hi C.D.</th>
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<th>Cleo Travel - Full Extended to Full Collapsed Time</th>
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<thead>
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<th>Maximum Gross Weight with Zero Wing Fuel</th>
<th>Maximum Flying Weight</th>
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<thead>
<tr>
<th>Limit Landing Sinking Speed (ft/Sec)</th>
<th>Limit Landing Sinking Speed (ft/Sec)</th>
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<table>
<thead>
<tr>
<th>Wing Lift Assumed for Landing Design Condition (per cent)</th>
<th>Stall Speed - Landing Configuration (per cent) (mph)</th>
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</table>

<table>
<thead>
<tr>
<th>Catching Cargo Compartment - Ultimate Design Press - Differential Lift (PSI)</th>
<th>12.54</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>28067.1</td>
</tr>
</tbody>
</table>

27
<table>
<thead>
<tr>
<th>GROSS WEIGHT CONDITION</th>
<th>WEIGHT (LB)</th>
<th>P.S.</th>
<th>% MAC</th>
<th>B.L.</th>
<th>W.L.</th>
<th>INCHES BELOW MAC</th>
<th>$I_x$ (Lb-In² x 10^-6)</th>
<th>$I_y$ (Lb-In² x 10^-6)</th>
<th>$I_z$ (Lb-In² x 10^-6)</th>
<th>$I_{xz}$ (Lb-In² x 10^-6)</th>
<th>$\alpha$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight Empty</td>
<td>325,263</td>
<td>1400</td>
<td>39.3</td>
<td>1</td>
<td>252</td>
<td>63</td>
<td>57,478.5</td>
<td>99,560.9</td>
<td>145,183.5</td>
<td>10,721.4</td>
<td>0.87</td>
</tr>
<tr>
<td>Basic Weight Plus Troop Kit</td>
<td>335,537</td>
<td>1405</td>
<td>40.7</td>
<td>1</td>
<td>254</td>
<td>61</td>
<td>57,865.4</td>
<td>101,742.1</td>
<td>147,586.2</td>
<td>11,013.8</td>
<td>0.91</td>
</tr>
<tr>
<td>Operating Weight</td>
<td>728,452</td>
<td>1393</td>
<td>37.5</td>
<td>1</td>
<td>252</td>
<td>63</td>
<td>57,909.0</td>
<td>101,485.9</td>
<td>146,943.8</td>
<td>10,697.5</td>
<td>0.76</td>
</tr>
<tr>
<td>Operating Weight &amp; Troop Kit</td>
<td>337,033</td>
<td>1402</td>
<td>39.9</td>
<td>1</td>
<td>254</td>
<td>61</td>
<td>57,991.9</td>
<td>102,999.2</td>
<td>148,411.1</td>
<td>10,951.2</td>
<td>0.81</td>
</tr>
<tr>
<td>Basic Flight Design-Max-Cargo</td>
<td>728,000</td>
<td>1379</td>
<td>37.7</td>
<td>0</td>
<td>250</td>
<td>65</td>
<td>150,214.2</td>
<td>150,202.5</td>
<td>233,950.9</td>
<td>12,051.6</td>
<td>5.11</td>
</tr>
<tr>
<td>Basic Flight Design-Max. Fuel</td>
<td>728,000</td>
<td>1379</td>
<td>37.7</td>
<td>0</td>
<td>276</td>
<td>39</td>
<td>170,866.5</td>
<td>124,743.5</td>
<td>279,748.2</td>
<td>10,618.1</td>
<td>5.62</td>
</tr>
<tr>
<td>Maximum Design-Maxim-Cargo</td>
<td>769,000</td>
<td>1350</td>
<td>25.8</td>
<td>0</td>
<td>246</td>
<td>69</td>
<td>149,599.9</td>
<td>158,894.7</td>
<td>291,557.3</td>
<td>12,322.0</td>
<td>0.92</td>
</tr>
<tr>
<td>Maximum Design-Maxim-Fuel</td>
<td>769,000</td>
<td>1371</td>
<td>31.5</td>
<td>0</td>
<td>271</td>
<td>44</td>
<td>171,415.7</td>
<td>133,034.9</td>
<td>287,845.4</td>
<td>11,064.1</td>
<td>5.38</td>
</tr>
<tr>
<td>Ferry Mission (Zero Cargo)</td>
<td>665,333</td>
<td>1393</td>
<td>37.5</td>
<td>0</td>
<td>285</td>
<td>30</td>
<td>169,830.5</td>
<td>109,899.6</td>
<td>265,343.4</td>
<td>9,741.7</td>
<td>5.76</td>
</tr>
<tr>
<td>Landplane Landing (Max. Cargo)</td>
<td>695,850</td>
<td>1311</td>
<td>23.4</td>
<td>0</td>
<td>232</td>
<td>85</td>
<td>91,010.0</td>
<td>154,948.7</td>
<td>230,265.2</td>
<td>12,272.7</td>
<td>5.00</td>
</tr>
<tr>
<td>Personnel/Peaoadload</td>
<td>728,000</td>
<td>1370</td>
<td>31.2</td>
<td>0</td>
<td>253</td>
<td>62</td>
<td>150,180.0</td>
<td>151,885.0</td>
<td>285,509.3</td>
<td>11,974.0</td>
<td>0.02</td>
</tr>
<tr>
<td>Typical Vehicle</td>
<td>728,000</td>
<td>1343</td>
<td>34.8</td>
<td>0</td>
<td>269</td>
<td>66</td>
<td>149,575.0</td>
<td>151,391.6</td>
<td>285,070.4</td>
<td>10,991.5</td>
<td>0.41</td>
</tr>
<tr>
<td>YAC Passenger</td>
<td>728,000</td>
<td>1369</td>
<td>31.0</td>
<td>0</td>
<td>266</td>
<td>49</td>
<td>166,286.6</td>
<td>151,574.2</td>
<td>301,287.4</td>
<td>12,191.9</td>
<td>2.12</td>
</tr>
<tr>
<td>** Kost Forward C.G.</td>
<td>713,904</td>
<td>1379</td>
<td>22.9</td>
<td>0</td>
<td>249</td>
<td>66</td>
<td>139,966.0</td>
<td>152,015.6</td>
<td>275,469.4</td>
<td>12,351.7</td>
<td>5.17</td>
</tr>
<tr>
<td>*** Kost Aft C.G. (Gear Up)</td>
<td>683,904</td>
<td>1406</td>
<td>41.0</td>
<td>0</td>
<td>246</td>
<td>69</td>
<td>135,513.4</td>
<td>145,574.8</td>
<td>265,053.0</td>
<td>10,261.4</td>
<td>4.50</td>
</tr>
</tbody>
</table>

L.E.M.A.C. = F.S. 1254.24
R.A.C. = 370.52 Inches

* Angle of Inclination of the principal axis (nose down) with respect to the air vehicle "x" axis.

** Represents the most critical forward center of gravity condition with respect to the allowable center of gravity limits for the 2.5g maximum cargo mission.

*** Represents the most critical aft center of gravity condition with respect to the allowable center of gravity limits for the 2.5g maximum cargo mission.
The weights have been reallocated for inertia calculation as shown in Table 9, and pertinent geometry items are given in Table 10.

<table>
<thead>
<tr>
<th>WING GROUP</th>
<th>WEIGHT</th>
<th>X,Y,Z (when needed)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>82045</td>
<td></td>
</tr>
<tr>
<td>Surface Controls</td>
<td>3796</td>
<td></td>
</tr>
<tr>
<td>Fuel System</td>
<td>2458</td>
<td></td>
</tr>
<tr>
<td>Anti-Ice</td>
<td>229</td>
<td></td>
</tr>
<tr>
<td>Trapped Fuel</td>
<td>562</td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>89090</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>FUSELAGE GROUP</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>116048</td>
<td></td>
</tr>
<tr>
<td>Distributed:</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Surface Control</td>
<td>1270</td>
<td></td>
</tr>
<tr>
<td>Inst. &amp; Navi.</td>
<td>281</td>
<td></td>
</tr>
<tr>
<td>Hydraulic</td>
<td>2666</td>
<td></td>
</tr>
<tr>
<td>Electrical</td>
<td>2761</td>
<td></td>
</tr>
<tr>
<td>Total Dist.</td>
<td>6978</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>POINT MASSES:</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Main Landing Gear</td>
<td>33681</td>
<td>1292, 264, 81</td>
</tr>
<tr>
<td>Nose Landing Gear</td>
<td>4407</td>
<td>418, 0, 86</td>
</tr>
<tr>
<td>Auxiliary Power Unit</td>
<td>933</td>
<td>1485, 264, 141</td>
</tr>
<tr>
<td>Air Conditioning</td>
<td>3411</td>
<td>964, 0, 294</td>
</tr>
<tr>
<td>Auxiliary Gear</td>
<td>39</td>
<td>2025, 0, 308</td>
</tr>
<tr>
<td>Crew</td>
<td>1290</td>
<td>318, 0, 332</td>
</tr>
</tbody>
</table>
Table 9 (cont'd)

<table>
<thead>
<tr>
<th>Category</th>
<th>Weight</th>
<th>X, Y, Z</th>
</tr>
</thead>
<tbody>
<tr>
<td>Radar</td>
<td>376</td>
<td>80, 0, 260</td>
</tr>
<tr>
<td>Surface Controls</td>
<td>271</td>
<td>290, 0, 332</td>
</tr>
<tr>
<td>Instruments &amp; Navigation</td>
<td>657</td>
<td>290, 0, 332</td>
</tr>
<tr>
<td>Other Equipment</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>Tiedown Devices</td>
<td>1750</td>
<td>694, 0, 165</td>
</tr>
<tr>
<td>Life Rafts</td>
<td>200</td>
<td>698, 0, 334</td>
</tr>
<tr>
<td>Food</td>
<td>17</td>
<td>690, 0, 335</td>
</tr>
<tr>
<td>Water</td>
<td>43</td>
<td>651, 0, 365</td>
</tr>
<tr>
<td>Liquid O₂</td>
<td>63</td>
<td>1280, 0, 153</td>
</tr>
<tr>
<td><strong>Total Pt. Mass</strong></td>
<td>47138</td>
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</tr>
<tr>
<td><strong>Volumes:</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Avionics</td>
<td>3514</td>
<td>707, 0, 316</td>
</tr>
<tr>
<td>Furnishings</td>
<td>6836</td>
<td>763, 0, 281</td>
</tr>
<tr>
<td><strong>Total Vol</strong></td>
<td>10350</td>
<td></td>
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</tbody>
</table>

**HORIZONTAL TAIL GROUP**

<table>
<thead>
<tr>
<th>Category</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>6793</td>
</tr>
<tr>
<td>Surface Controls</td>
<td>913</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td>7706</td>
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</table>

**VERTICAL TAIL GROUP**

<table>
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<tr>
<th>Category</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>5603</td>
</tr>
<tr>
<td>Surface Control</td>
<td>885</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td>6488</td>
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</table>

**PROPULSION GROUP**

<table>
<thead>
<tr>
<th>Category</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Engines &amp; System</td>
<td>33804</td>
</tr>
<tr>
<td>Hydraulic</td>
<td>1313</td>
</tr>
</tbody>
</table>

30
Table 9. Continued

<table>
<thead>
<tr>
<th>Electrical</th>
<th>690</th>
</tr>
</thead>
<tbody>
<tr>
<td>Oil</td>
<td>264</td>
</tr>
<tr>
<td>Nacelles &amp; Pylons</td>
<td>9586</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>45657</strong></td>
</tr>
</tbody>
</table>

Aircraft WE

Table 10. C-5A Geometry Definitions for Inertia

**GEOMETRY DATA**

<table>
<thead>
<tr>
<th>WING</th>
<th>HORIZ</th>
<th>VERT</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\Lambda_{L1}$</td>
<td>28°</td>
<td>$30^\circ$</td>
</tr>
<tr>
<td>$\Lambda_T$</td>
<td>14°</td>
<td>$9^\circ$</td>
</tr>
<tr>
<td>b</td>
<td>1336</td>
<td>412</td>
</tr>
<tr>
<td>c</td>
<td>525</td>
<td>250</td>
</tr>
<tr>
<td>$t_r$</td>
<td>72</td>
<td>26</td>
</tr>
<tr>
<td>$t_t$</td>
<td>20</td>
<td>10</td>
</tr>
<tr>
<td>XS4</td>
<td>806</td>
<td>2605</td>
</tr>
<tr>
<td>ZS1</td>
<td>370</td>
<td>780</td>
</tr>
<tr>
<td>YS4</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>$\theta$</td>
<td>$-5^\circ$</td>
<td>$-5^\circ$</td>
</tr>
</tbody>
</table>

**FUSELAGE**

<table>
<thead>
<tr>
<th>PROPULSION</th>
</tr>
</thead>
<tbody>
<tr>
<td>$l_h$</td>
</tr>
<tr>
<td>$l_c$</td>
</tr>
<tr>
<td>$l_t$</td>
</tr>
<tr>
<td>R</td>
</tr>
<tr>
<td>$Z_b$</td>
</tr>
</tbody>
</table>

**ADDITIONAL ITEMS**

| $\Lambda_{L3}$ | 27° |
| $t_f$ | 14 |
| $l_p$ | 1600 |

$31$
A. Centroids

**WING**

\[ C_a = 525 \quad C_b = 710 \quad C_c = 858 \]

USING Eq (8)*, \( X_{S1} = 422 \)

USING Eq (9), \( Y_{S1} = 441 \)

\[ Y_{S1} = Y_{S1} \cos (-5^\circ) = 440 \]

\[ Z_{S3} = Y_{S1} \sin (-5^\circ) = -38 \]

**HORIZONTAL**

\[ C_a = 238 \quad C_b = 250 \quad C_c = 315 \]

USING Eq (8), \( X_{S1} = 165 \)

USING Eq (9), \( Y_{S1} = 144 \)

\[ Y_{S1} = Y_{S1} \cos (-5^\circ) = 143.5 \]

\[ Z_{S3} = Y_{S1} \sin (-5^\circ) = -13 \]

**VERTICAL**

\[ C_a = 305 \quad C_b = 371 \quad C_c = 605 \]

USING Eq (8), \( X_{S1} = 277 \)

USING Eq (9), \( Y_{S1} = 188 \)

\[ Y_{S1} = Y_{S1} \cos 90^\circ = 0 \]

\[ Z_{S3} = Y_{S1} \sin 90^\circ = 188 \]

**WING FUEL TANK**

USING Eq (59) \( X_{F1} = 254 \)

USING Eq (60) \( Y_{F1} = 323 \)

\[ Y_{F1} = Y_{F1} \cos (-5) = 321.6 \]

\[ Z_{F2} = Y_{F1} \sin (-5) = -28 \]

B. Wing Group Inertia

USING Eq (1), \( I_{1x} = 2.7028033 \times 10^{10} \)

USING Eq (2), \( I_{1y} = 1.9574734 \times 10^{10} \)

USING Eq (3), \( I_{1z} = 4.6602767 \times 10^{10} \)

TRUE \( I_{1y} = I_{1y} \cos (-5) + I_{1z} \sin (-5) = 1.5438548 \times 10^{10} \)
TRUE $I_{lx} = I_{ly} \sin(-5) + I_{lz} \cos(-5) = 4.471938 \times 10^{10}$

USING (7), $I_{lxz} = -1.819266687 \times 10^{9}$

$I_x = I_{lx} - 89090 (440)^2 - 89090 (-38)^2 + 89090 (440 + 0)^2$
+ 89090 (-38 + 370)$^2 = 3.67192431 \times 10^{10}$

$I_y = I_{ly} - 89090 (422)^2 - 89090 (-38)^2 + 89090 (422 + 806)^2$
+ 89090 (-38 + 370)$^2 = 1.4361055 \times 10^{11}$

$I_z = I_{lz} - 89090 [(422)^2 + (440)^2] + 89090 (422 + 806)^2$
+ 89090 (440 + 0)$^2 = 1.63200171 \times 10^{11}$

$I_{xz} = I_{lxz} - 89090 (422)(-38) + 89090(422 + 806)(-38 + 370) = 3.5931017 \times 10^{10}$

C. HORIZONTAL TAIL GROUP INERTIA

USING (1), $I_{lx} = 2.45844912 \times 10^{8}$

USING (2), $I_{ly} = 2.80151979 \times 10^{8}$

USING (3), $I_{lz} = 5.2599689 \times 10^{8}$

TRUE $I_{ly} = I_{ly} \cos(-5) + I_{lz} \sin(-5) = 2.3324227 \times 10^{8}$

TRUE $I_{lz} = I_{ly} \sin(-5) + I_{lz} \cos(-5) = 4.9957846 \times 10^{8}$

USING (7), $I_{lxz} = -1.941127 \times 10^{7}$

$I_x = I_{lx} - 7706 (143.5)^2 - 7706 (-13)^2 + 7706 (143.5 + 0)^2$
+ 7706 (-13 + 780)$^2 = 4.77784763 \times 10^{9}$

$I_y = I_{ly} - 7706 (165)^2 - 7706 (-13)^2 + 7706 (165 + 2605)^2$
+ 7706 (-13 + 780)$^2 = 6.3682867 \times 10^{10}$

$I_z = I_{lz} - 7706 (165)^2 + 143.5^2) + 7706 (165 + 2605)^2 + 7706 (143.5 + 0)^2$
= 5.82546306 \times 10^{10}$

$I_{xz} = I_{lxz} - 7706(165) (-13) + 7706(165+2605) (-13 + 780) = 1.636920864 \times 10^{10}$

D. VERTICAL TAIL GROUP INERTIA

USING (1), $I_{lx} = 3.17905191 \times 10^{8}$

USING (2), $I_{ly} = 7.2770685 \times 10^{8}$
USING (3), $I_{lx} = 1.045612041 \times 10^9$

TRUE $I_{ly} = I_{ly} \cos 90 + I_{lz} \sin 90 = 1.045612041 \times 10^9$

TRUE $I_{lz} = I_{ly} \sin 90 + I_{lz} \cos 90 = 7.2770685 \times 10^8$

USING (7), $I_{lxz} = 2.58157 \times 10^8$

$I_x = I_{lx} - 6488 (0)^2 - 6488 (188)^2 + 6488 (0 + 0)^2 + 6488 (188 + 365)^2$
\[= 2.07268211 \times 10^9\]

$I_y = I_{ly} - 6488 (277)^2 - 6488 (188)^2 + 6488 (277 + 2425)^2 + 6488 (188 + 365)^2$
\[= 4.967018756 \times 10^{10}\]

$I_z = I_{lz} - 6488 [(277)^2 + 0^2] + 6488 (277 + 2425)^2 + 6488 (0 + 0)^2$
\[= 4.75975055 \times 10^{10}\]

$I_{xz} = I_{lxz} - 6488 (277)(188) + 6488(277 + 2425)(188 + 365) = 9.69440853 \times 10^9$

E. Fuselage Group Inertia

STRUCTURE:

$S_n = \pi (138) \sqrt{138^2 + 440^2} = 199,920$

$S_c = 2 \pi (138)(1300) = 1,127,203$

$S_t = \pi (138) \sqrt{138^2 + 1027} = 449,247$

$S_f = 1,776,370$

$W_n = 199,920 \ (116,048) = 13,061$

$1,776,370$

$W_c = 1,127,203 \ (116,048) = 73,639$

$1,776,370$

$W_t = 449,247 \ (116,048) = 29,349$

$1,776,370$

USING (25), $I_x = 9.651054 \times 10^9$

USING (26), $I_y = 2.36853866 \times 10^{11}$

USING (27), $I_z = 2.290090211 \times 10^{11}$

USING (28), $I_{xz} = 3.722851418 \times 10^{10}$
DISTRIBUTED CONTENTS:

USING (29), \( I_x = 6.04602 \times 10^8 \)

USING (30), \( I_y = 1.79106443 \times 10^{10} \)

USING (31), \( I_z = 1.743893 \times 10^{10} \)

USING (32), \( I_{xz} = 2.6135712 \times 10^9 \)

VOLUMES:

AVIONICS: RECTANGULAR SOLID

USING (37), \( I_x = \frac{3514}{12} (250^2 + 250^2) + 3514 (316)^2 \)

\[ = 3.8749815 \times 10^8 \]

USING (38), \( I_y = \frac{3514}{12} (250^2 + 1315^2) + 3514 (707^2 + 316^2) \)

\[ = 2.6320402 \times 10^9 \]

USING (39), \( I_z = \frac{3514}{12} (1315^2 + 250^2) + 3514 (707)^2 \)

\[ = 2.2811462 \times 10^9 \]

USING (40), \( I_{xz} = 3514 (707)(316) = 7.850698 \times 10^8 \)

FURNISHINGS: RECTANGULAR SOLID

USING (37), \( I_x = \frac{6836}{12} (250^2 + 250^2) + 6836 (281^2) \)

USING (38), \( I_y = \frac{6836}{12} (250^2 + 1100^2) + 6836 (763^2 + 281^2) \)

USING (39), \( I_z = \frac{6836}{12} (1100^2 + 250^2) + 6836 (763^2) \)

USING (40), \( I_{xz} = 6836 (763)(281) = 1.4656589 \times 10^9 \)

POINT MASSES: for \( I_y \)

33681 \((1292^2 + 81^2)\)

4407 \((418^2 + 86^2)\)

933 \((1485^2 + 141^2)\)

3411 \((964^2 + 294^2)\)
\[
39 \left(2025^2 + 308^2\right)
\]
\[
1290 \left(318^2 + 332^2\right)
\]
\[
376 \left(80^2 + 260^2\right)
\]
\[
271 \left(290^2 + 332^2\right)
\]
\[
657 \left(290^2 + 332^2\right)
\]
\[
1750 \left(694^2 + 165^2\right)
\]
\[
200 \left(698^2 + 334^2\right)
\]
\[
17 \left(690^2 + 335^2\right)
\]
\[
43 \left(651^2 + 365^2\right)
\]
\[
63 \left(1280^2 + 153^2\right)
\]

Using (42), \( I_y = 6.45800764 \times 10^{10} \)

\( w_{pc} = 0 \)

\( W_{pc} = 1989 \)

Using (41), \( I_x = \frac{1989 \left(138\right)^2 + 3 \left(0\right) \left(138\right)^2 + \Sigma W \left(y^2 + z^2\right)}{2} = 3.41181608 \times 10^9 \)

Using (43), \( I_z = 6.36604587 \times 10^{10} \)

Using (44), \( I_{xz} = 5.376484824 \times 10^9 \)

**F. PROPULSION GROUP INERTIA**

Using (45), \( I_x = 2 \left[ \frac{11414.3}{12} \left(80\right)^2 + \frac{.414.3}{12} \left(476^2 + 222^2\right) \right] \)

\[+ 2 \left[ \frac{11414.3}{12} \left(80\right)^2 + 11414.3 \left(743^2 + 198^2\right) \right] \]

\[= 1.994107887 \times 10^{10} \]

Using (46), \( I_y = 2 \left[ \frac{11414.3}{12} \left(3\left(80\right)^2 + 312^2\right) + 11414.3 \left(1020^2 + 222^2\right) \right] \)

\[+ 2 \left[ \frac{11414.3}{12} \left(3\left(80\right)^2 + 312\right) + 11414.3 \left(1165^2 + 198^2\right) \right] \]

\[= 5.719790196 \times 10^{10} \]

Using (47), \( I_z = 2 \left[ \frac{11414.3}{12} \left(3\left(80\right)^2 + 312^2\right) + 11414.3 \left(1020^2 + 476^2\right) \right] \)

\[+ 2 \left[ \frac{11414.3}{12} \left(3\left(80\right)^2 + 312\right) + 11414.3 \left(1165^2 + 743^2\right) \right] \]

\[= 7.29527635 \times 10^{10} \]

36
USING (48), \( I_{xz} = 22829(1020)(222) + 22829(1165)(198) = 1.043536419 \times 10^{10} \)

G. **SUMMATION OF GROUP I** \( x,y,z \)

Total \( I_x = 7.817685768 \times 10^{10} \)
Total \( I_y = 6.5382519 \times 10^{11} \)
Total \( I_z = 6.59099235 \times 10^{11} \)
Total \( I_{xz} = 1.198992973 \times 10^{11} \)

H. **AIRCRAFT CENTER OF GRAVITY**

\( \bar{x}: \ W \ (x) \)

**Wing**
89,090 (1228)

**Fuselage**
13,061 (293)
73,639 (1090)
29,349 (2082)
6978 (1423)

**Horizontal Tail**
7706 (2770)

**Vertical Tail**
6488 (2702)

**Propulsion**
22,829 (1020)
22,829 (1165)

**Point Masses**
33681 (1292)
4407 (418)
933 (1485)
3411 (964)
39 (2025)
1290 (318)
376 (80)
271 (290)
657 (290)
1750 (694)
200 (698)
17 (690)
43 (651)
63 (1280)

Volumes
6836 (763)
3514 (707)

\[ WX = 4.13282587 \times 10^8 \]
\[ W = 329,456 \]
\[ X = \frac{WX}{W} = 1254.4 \]

Fuselage
116,048 (260)

6.978 (260)

Wing
89,090 (332)

Horizontal Tail
7706 (767)

Vertical Tail
6488 (553)

Propulsion
22,829 (222)
22,829 (198)
### Point Masses

<table>
<thead>
<tr>
<th>Mass</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>33,681</td>
<td>(81)</td>
</tr>
<tr>
<td>4,407</td>
<td>(86)</td>
</tr>
<tr>
<td>933</td>
<td>(141)</td>
</tr>
<tr>
<td>3411</td>
<td>(294)</td>
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<tr>
<td>39</td>
<td>(308)</td>
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<td>(332)</td>
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<td>271</td>
<td>(332)</td>
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<tr>
<td>657</td>
<td>(332)</td>
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<td>200</td>
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<tr>
<td>43</td>
<td>(365)</td>
</tr>
<tr>
<td>63</td>
<td>(153)</td>
</tr>
</tbody>
</table>

### Volumes

<table>
<thead>
<tr>
<th>Volume</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>3514</td>
<td>(316)</td>
</tr>
<tr>
<td>6836</td>
<td>(281)</td>
</tr>
</tbody>
</table>

\[ \Sigma WZ = 8.9156803 \times 10^7 \]

\[ \Sigma W = 329,456 \]

\[ Z = \frac{\Sigma WZ}{\Sigma W} = 270.6 \]

### I. Moments of Inertia About Aircraft Center of Gravity (Operating Weight Empty)

\[ I_{cgx} = I_x - (329,456)(270.6)^2 \]

\[ = 5.40461473 \times 10^{10} \]
I_{cg} = I - 329,456 (270.6^2 + 1254.4)^2
= 9.88529;998 \times 10^{10}

I_{cz} = I - 329,456 (1254.4)^2
= 1.406938403 \times 10^{11}

I_{cgy} = 8.06854177 \times 10^9

J. CALCULATIONS FOR BASIC FLIGHT DESIGN WEIGHT WITH MAX FUEL:

INTERNAL WING FUEL:

USING (53), I_{4x} = 5.24631042 \times 10^{10}

USING (54), I_{4y} = 2.467113514 \times 10^{10}

USING (55), I_{4z} = 7.71342393 \times 10^{10}

USING (58), I_{4xz} = -2.95088621 \times 10^9

TRUE I_{4y} = I_{4y} \cos (-5) + I_{4z} \sin (-5) = 1.7854562 \times 10^{10}

TRUE I_{4z} = I_{4y} \sin (-5) + I_{4z} \cos (-5) = 7.690489 \times 10^{10}

I_x = I_{4x} - 318500 (321.6)^2 - 318500 (-28)^2 + 318500 (321.6 + 190)^2
+ 318500 (-28 + 360)^2 = 1.3774084 \times 10^{11}

I_y = I_{4y} - 318500 (254)^2 - 318500 (-28)^2 + 318500 (254 + 941)^2
= 4.869888185 \times 10^{11}

I_z = I_{4z} - 318500 (254^2 + 321.6^2) + 381500 (254 + 941)^2 + 318500 (321.6 + 190)^2
= 5.618329538 \times 10^{11}

I_{xz} = I_{4xz} - 318500 (254) (-28) + 318500 (254 + 941) (-28 + 360)
= 1.25665746 \times 10^{11}
PAYLOAD

USING (69), \( I_x = \frac{71787}{(12)(170^2 + 170^2) + 71787 (192)^2} = 2.99213002 \times 10^9 \)

USING (70), \( I_y = \frac{71787}{12 (1600^2 + 170^2) + 71787 (1084^2 + 192^2)} \)
\[= 1.0248755 \]

USING (71), \( I_z = \frac{71787}{12 (1600^2 + 170^2) + 71787 (1084)^2} = 9.98411921 \times 10^{10} \)

USING (72), \( I_{xz} = 71787 (1084) (192) = 1.494088474 \times 10^{10} \)

POINT MASSES:

\( 680 (440^2 + 319^2) \)
\( 7581 (1619^2 + 341^2) \)
\( I_y = 2.09534 \times 10^{10} \)
\( I_z = 2.000267 \times 10^{10} \)

\( W_{pc} = 8261 \)
\( I_x = \frac{8261 (180)^2}{2} + 8261 (335)^2 = 1.0609189 \times 10^9 \)
\( I_{xz} = 4.2807557 \times 10^9 \)

Total \( I_x = 2.199674956 \times 10^{11} \)
Total \( I_y = 1.25181229 \times 10^{12} \)
Total \( I_z = 1.340776051 \times 10^{12} \)

Total \( I_{xz} = 2.647866836 \times 10^{11} \)

NEW CENTER OF GRAVITY DUE TO ADDITIONAL WEIGHT:

\[
\bar{x}: \quad 318500 \quad (1195)
\]

\[
71787 \quad (1084)
\]

\[
680 \quad (440)
\]

\[
7581 \quad (1619)
\]

\( \Sigma W (INCLUDING\ OPERATING\ WEIGHT) = 8.84280034 \times 10^8 \)

\( \Sigma W (INCLUDING\ OPERATING\ WEIGHT) = 728,025 \)

\( \bar{x} = 1214.6 \)

\[
\bar{z}: \quad 318500 \quad (332)
\]

\[
71787 \quad (192)
\]

\[
680 \quad (319)
\]

\[
7581 \quad (341)
\]

\( \Sigma WZ (INCLUDING\ OPERATING\ WEIGHT) = 2.114839 \times 10^8 \)

\( \Sigma W (INCLUDING\ OPERATING\ WEIGHT) = 728,025 \)

\( \bar{z} = 290 \)

MOMENTS OF INERTIA ABOUT AIRCRAFT CENTER OF GRAVITY:

\[
I_{cgx} = I_x - 728025 \times 290^2 = 1.5874059 \times 10^{11}
\]

\[
I_{cgy} = I_y - 728025 \times (290.2 + 1214.6^2) = 1.16564203 \times 10^{11}
\]

\[
I_{cgz} = I_z - 728025 \times (1214.6)^2 = 2.66754869 \times 10^{11}
\]

\[
I_{cgxz} = I_{xz} - 728025 \times (290 \times 1214.6) = 8.3515259 \times 10^9
\]
### SUMMARY OF ACTUAL VERSUS CALCULATED INERTIAS FOR C-5A

(MOMENTS OF INERTIA $\times 10^{-6}$)

<table>
<thead>
<tr>
<th>OPERATING WEIGHT EMPTY</th>
<th>$I_x$ (ROLL)</th>
<th>$I_y$ (PITCH)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>ACTUAL</td>
<td>CALCULATED</td>
</tr>
<tr>
<td>WEIGHT EMPTY</td>
<td>57,909</td>
<td>54,246</td>
</tr>
<tr>
<td>WITH MAX FUEL</td>
<td>170,867</td>
<td>158,941</td>
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</table>

<table>
<thead>
<tr>
<th>OPERATING WEIGHT EMPTY</th>
<th>$I_z$ (YAW)</th>
<th>$I_{xz}$</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>ACTUAL</td>
<td>CALCULATED</td>
</tr>
<tr>
<td>WEIGHT EMPTY</td>
<td>146,944</td>
<td>140,694</td>
</tr>
<tr>
<td>WITH MAX FUEL</td>
<td>279,748</td>
<td>266,755</td>
</tr>
</tbody>
</table>
References


1. Interactive Computer-Aided Design Aircraft Flying Qualities Program, ASD/XR 74-17, August 1974.

3. C-5A Actual Weight and Balance Report, Lockheed


5. A-10A Actual Weight and Balance Report, Fairchild SW160 R0070

Appendix - Derivation of Equations

1. Surface Inertia and Volume

(Surface Diagram)

\[ I = \int r^2 \, dm, \quad dm = \rho \, dV, \quad I = \rho \int r^2 \, dV, \quad dV = tdxdy \]

\[ I_{ix} \text{ (ROLL)} = \rho \int_0^b \int_0^{y\tan \Lambda_T} \left( t_r - \frac{t_r - t_t}{b} y \right) dx dy \]

\[ = \rho \int_0^b \left( y^2e + y^3\tan \Lambda_T - y^3\tan \Lambda_L \right) \left( t_r - \frac{t_r - t_t}{b} y \right) dy \]

\[ = \frac{Wb^3}{V} \left\{ \left( t_r - t_t \right) \left( \frac{e}{4} + \frac{b\tan \Lambda_T}{5} - \frac{b\tan \Lambda_L}{5} \right) \right\} \]

\[ + \left[ t_r \left( \frac{e}{3} + \frac{b\tan \Lambda_T}{4} - \frac{b\tan \Lambda_L}{4} \right) \right] \]

Calculating the volume \( V \):

\[ V = \int \int \int dx \, dy \, dz = \int_0^b \int_0^{y\tan \Lambda_L} \left( t_r - \frac{t_r - t_t}{b} y \right) dx dy \]

45
\[
= \int_0^b \left[ c + y \tan \Lambda_T - y \tan \Lambda_L \right] \left[ t_r - \frac{t_r - t_s}{b} y \right] \, dy
= b \left\{ t_r \left[ c + \frac{b}{2} \left( \tan \Lambda_T - \tan \Lambda_L \right) \right] - (t_r - t_s) \left[ \frac{c}{2} + \frac{b}{3} \left( \tan \Lambda_T - \tan \Lambda_L \right) \right] \right\}
\]

\[I_{1y} \text{ (PITCH)} = \rho \int_0^b \int_{y \tan \Lambda_L}^{c + y \tan \Lambda_T} x^2 \left( t_r - \frac{t_r - t_s}{b} y \right) \, dx \, dy\]

\[= \rho \int_0^b \left\{ \left( \frac{c + y \tan \Lambda_T}{3} \right)^3 \left( t_r - \frac{t_r - t_s}{b} y \right) - \frac{y^3 \tan^3 \Lambda_L}{3} \left( t_r - \frac{t_r - t_s}{b} y \right) \right\} \, dy\]

\[= \rho \int_0^b \left\{ \left[ \frac{c^3}{3} y^3 + c^2 y^2 \tan \Lambda_T + c y^3 \tan^2 \Lambda_T + y^4 \tan^3 \Lambda_T - y^4 \tan^3 \Lambda_L \right] + \left[ t_r \left( \frac{c^3}{3} + c^2 y \tan \Lambda_T + c y^2 \tan^2 \Lambda_T + y^3 \tan^3 \Lambda_T - y^3 \tan^3 \Lambda_L \right) \right] \right\} \, dy\]

\[= \frac{Wb}{V} \left\{ t_r \left( \frac{c^3}{3} + bc \tan \Lambda_T \left( \frac{c}{6} + \frac{b \tan \Lambda_T}{9} \right) + \frac{b^3}{12} \left( \tan^3 \Lambda_T - \tan^3 \Lambda_L \right) \right) \right\} - \left[ \left( t_r - t_s \right) \left( \frac{c^3}{6} + bc \tan \Lambda_T \left( \frac{c}{3} + \frac{b \tan \Lambda_T}{9} \right) + \frac{b^3}{15} \left( \tan^3 \Lambda_T - \tan^3 \Lambda_L \right) \right) \right\} \]

\[I_{1z} \text{ (YAW)} = \rho \int_0^b \int_{y \tan \Lambda_L}^{c + y \tan \Lambda_T} y^2 + x^2 \left( t_r - \frac{t_r - t_s}{b} y \right) \, dx \, dy\]

\[= I_{1x} + I_{1y}\]
\[
I_{ix\_z} = \rho \int_0^\circ \int_0^{\frac{cr + z\tan \lambda_r}{\sin \theta}} xzt \sin \theta \, dx \, dz +
\rho \int_0^{\frac{bcos \theta}{\sin \theta}} xzt \cos \theta \, dx \, dy
\]

\[
= trp \int_0^\circ \int_0^{\frac{cr + z\tan \lambda_r}{\sin \theta}} xz \sin \theta \, dx \, dz + trp \int_0^{\frac{bcos \theta}{\sin \theta}} xz \cos \theta \, dx \, dy
\]

\[
= \rho \left( t_r - t_e \right) \int_0^\circ \int_0^{\frac{y\tan \lambda_r}{\cos \theta}} xz \tan \theta \, dx \, dy
\]

\[
= \frac{w}{v} \cdot t_r \sin \theta \left[ \frac{cr^2 b^2}{4} + \frac{cr b^3 \tan \lambda_r}{3} + \frac{b^4}{8} (\tan^3 \lambda_r - \tan^3 \lambda_e) \right]
\]

\[
- \frac{w}{v} \left( t_r - t_e \right) \sin \theta \left[ \frac{cr^2 b^2}{6} + \frac{cr b^3 \tan \lambda_r}{4} + \frac{b^4}{10} (\tan^3 \lambda_r - \tan^3 \lambda_e) \right]
\]

**NOTE:** Equations are correct for any \( \tan \lambda_r \)
2. Longitudinal and spanwise surface center of gravity location

Longitudinal centroid:

\[
\begin{align*}
\Sigma m &= \int_0^{c_e} \rho \frac{C^2}{C_0} x^2 dx + \int_0^{c_t} \rho \frac{C^2}{C_0} x^2 dx - \int_0^{c_e} \frac{\rho C_x}{C_0} x dx + \int_0^{c_t} \frac{\rho C_x}{C_0} x dx + \frac{\rho C_x}{C_0} x dx + \frac{\rho C_x}{C_0} x dx \\
\Sigma m x &= \int_0^{c_e} \rho \frac{C^2}{C_0} x^2 dx + \int_0^{c_t} \rho \frac{C^2}{C_0} x^2 dx - \int_0^{c_e} \frac{\rho C_x}{C_0} x dx + \int_0^{c_t} \frac{\rho C_x}{C_0} x dx + \frac{\rho C_x}{C_0} x dx + \frac{\rho C_x}{C_0} x dx \\
I &= \Sigma m x^2 + \int_0^{c_e} \rho \frac{C^2}{C_0} x x dx - \int_0^{c_e} \frac{\rho C_x}{C_0} x^2 dx + \int_0^{c_t} \frac{\rho C_x}{C_0} x^2 dx + \int_0^{c_e} \frac{\rho C_x}{C_0} x^2 dx \\
I_{cm} &= K_o \left[ 1 - \frac{(\Sigma m x)^2}{2m} \right]
\end{align*}
\]

where

- \( K_o = 0.703 \) for any wing
- \( K_o = 0.771 \) for any horizontal or vertical stabilizer

Assuming that \( I_{oy} = K_o I - K_o \left( \frac{EMX}{EM} \right)^2 \) and knowing that \( x = \frac{EMX}{EM} \)
We have $I_{xy} = K_o I - EMK_o \overline{x}^2$. Since $\overline{x}^2$ is multiplied by $K_o$ we assume that $\overline{x}$ is multiplied by $\sqrt{K_o} \overline{x} = 2(\overline{c_a}^2 + \overline{c_b}^2 + \overline{c_c}^2) \sqrt{(-\overline{c_a} + \overline{c_b} + \overline{c_c})}$

\[
\overline{x} = (-\overline{c_a}^2 + \overline{c_b}^2 + \overline{c_c}^2) \sqrt{(-\overline{c_a} + \overline{c_b} + \overline{c_c})}
\]

(Where $\overline{x} = XS_1, XS_2, \text{or} XS_3$)

Spanwise centroid:

Using diagram in Section (1) we have:

\[
\overline{y} = \frac{1}{V} \int_0^b \left( c + y \tan \Lambda_T \right) g \left( t_r - \frac{t_r - t_s}{b} y \right) dx dy
\]

\[
= \frac{1}{V} \int_0^b \left[ c + y \tan \Lambda_T \right] \left[ t_r - \frac{t_r - t_s}{b} y \right] - y \left[ y \tan \Lambda_T \right] \left[ t_r - \frac{t_r - t_s}{b} y \right] dy
\]

\[
= \frac{V}{V} \left[ t_r \left( \frac{c}{2} + \frac{b}{2} \left( \tan \Lambda_T - \tan \Lambda_c \right) \right) - (t_r - t_s) \left( \frac{c}{2} + \frac{b}{2} \left( \tan \Lambda_T - \tan \Lambda_c \right) \right) \right]
\]

(Where $\overline{y} = YS_1,YS_2, \text{or} YS_3$)

(3) Fuselage Structure

$R, l_n, l_c$ and $l_t$ are chosen to best fit the fuselage geometry of the aircraft.

\[
S_n = \pi R \sqrt{R^2 + l_n^2}
\]

\[
S_c = 2 \pi R \frac{1}{c}
\]

\[
S_t = \pi R \sqrt{R^2 + l_t^2}
\]

Distributing weight according to surface area:

\[
W_n = \frac{S_n \left( W \right)}{S_n + S_c + S_t}
\]

\[
W_c = \frac{S_c \left( W \right)}{S_n + S_c + S_t}
\]

\[
W_t = \frac{S_t \left( W \right)}{S_n + S_c + S_t}
\]
(4) Fuselage distributed contents

$I_o$ for a right - cylindrical open ended shell is given in Section 9. Translating this and defining the terms in different notation:

$I_x = W_{dc} R^2 + W_{dc} (Z_b)^2$

$I_y = W_{dc} \left[ R^2 + \frac{1}{6} \left( XS4 - CREW c.g. \right)^2 \right] + W_{dc} \left( XS4 - CREW c.g. + \right. \frac{CREW c.g.}{2} \left. \right)^2 + W_{dc} \left( Z_b \right)^2$

$I_z = I_y - W_{dc} (Z_b)^2$

(5) Fuselage volumes of mass

$I_o$ for a right circular shell and solid rectangle are given in Section 9. Translating these to the remote axes and changing the notation gives:

Right circular cylindrical shell

$I_x = W_{vo} R_v^2 + W_{vo} (Z)^2$

$I_y = W_{vo} \left( R_v^2 + \frac{1}{6} \left( X_v + Z^2 \right) \right) + W_{vo} \left( X^2 + Z^2 \right)$

$I_z = W_{vo} \left( R_v^2 + \frac{1}{6} \left( X_v + Z^2 \right) \right) + W_{vo} \left( X^2 + Z^2 \right)$

Rectangular solid:

$I_x = \frac{W_{vo}}{12} \left( 2R_v^2 + 2 R_v^2 \right) + W_{vo} (Z)^2$

$I_y = \frac{W_{vo}}{2} \left( 1_v^2 + 2R_v^2 \right) + W_{vo} (X^2 + Z^2)$

$I_z = \frac{W_{vo}}{12} \left( 1_v^2 + 2R_v^2 \right) + W_{vo} (X^2 + Z^2)$

(6) Fuselage point masses

For point masses, the inertia about the center of the mass is so small that it can be neglected. For pitch and yaw we just translate the mass to each respective axis:

$I_x = W_p (Y^2 + Z^2)$

$I_y = W_p (X^2 + Z^2)$

$I_z = W_p (X)^2$
(PITCH) \[ I_y = I_y^{\text{fuselage structure}} + I_y^{\text{nose cone}} + I_y^{\text{cylinder}} + I_y^{\text{tail cone}} \]

(See Section 9)

\[ I_y^{\text{nose}} = \frac{W_n}{4} (R^2 + \frac{2}{9} l_n^2) + W_n \left( \frac{2}{3} 1_n l_n \right)^2 = \frac{W_n}{4} (R^2 + 2l_n^2) \]

\[ I_y^{\text{cylinder}} = \frac{W_c}{2} (R^2 + \frac{1}{2} l_c^2) + W_c \left( \frac{1}{2} 1_c 1_n \right)^2 = \frac{W_c}{2} (R^2 + \frac{1}{2} l_c^2) + \frac{1}{6} l_c^2 + \frac{1}{3} l_c l_n^2 \]

\[ I_y^{\text{tail}} = \frac{W_t}{4} (R^2 + \frac{2}{9} l_t^2) + W_t \left( \frac{1}{3} 1_t 1_n \right)^2 = \frac{W_t}{4} (R^2 + \frac{2}{9} l_t^2) + \frac{2}{3} l_t l_n \]

Adding these three together:

\[ I_y = \frac{R^2}{4} (W_n + 2W_c + W_t) + l_n^2 (\frac{2}{3} W_n + W_c + W_t) + l_c^2 (\frac{1}{3} W_c + W_t) + \frac{1}{6} l_t^2 \]

(Roll)

\[ I_x^{\text{nose}} = \frac{W_n R^2}{2}, \quad I_x^{\text{cylinder}} = W_c R^2, \quad I_x^{\text{tail}} = \frac{W_t R^2}{2} \]

Adding these three together:

\[ I_x = \frac{R^2}{2} (W_n + 2W_c + W_t) + W_s (Z_b)^2 \]

(YAW)

\[ I_z = I_c b - W_s (Z_b)^2 \]

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Using the I_{ox} equations for a solid cone and a solid right-cylinder and
translating them to the remote axes gives an alternate approach to I_{x}:

\[ I_x = \frac{W_{pc}}{2} x_c^2 + \frac{3}{10} \frac{W_{pnc}}{x_c^2} + (W_{pc} + W_{pb} (Z_b))^2 \]

7) Internal wing fuel tank centroid

Using the diagram in Section 1

\[
\chi F l = \frac{1}{V} \int_{0}^{b} \int_{y}^{y_{tan}} x (t_r - \frac{tr - tc + y}{b}) dy dx
\]

\[
= \frac{1}{V} \int_{0}^{b} (c + y \tan \Lambda_{l})^2 (t_r - \frac{tr - tc + y}{b}) dy - \frac{y^2 \tan \Lambda_{l}}{a} (tr - \frac{tr - tc + y}{b})
\]

\[
= \frac{b}{V} \left\{ \left[ \left( \frac{c^2}{2} + \frac{b c \tan \Lambda_{l}}{2} + \frac{b^2}{6} (\tan^2 \Lambda_{l} - \tan^2 \Lambda_{l}) \right) \right] - \left[ \left( tr - tc + \frac{c \tan \Lambda_{l}}{a} + \frac{b^2}{6} (\tan^2 \Lambda_{l} - \tan^2 \Lambda_{l}) \right) \right] \right\}
\]

YFL is the same as the spanwise centroid for surfaces derived in

section 2. V(volume) was derived in section 2.

(8) Internal fuselage fuel inertia

\[
V = W_{ff}
\]

\[
\rho \frac{V}{\Pi f}
\]

\[
R_f = \frac{W_{ff}}{\Pi f}
\]

See Section 9 for solid cylinder equation

I_{ox} is assumed equal to 0.

\[
I_{oz} = I_{oy} = \frac{W_{ff}}{12} (3(R_f)^2 + l_f^2) = \frac{W_{ff}}{12} \left( \frac{3W_{ff}}{\Pi f} + l_f^2 \right)
\]

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(9) Center of gravity, inertia and surface area of various geometric shapes.

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<td>Surface Area = ( 2\pi RH )</td>
<td>( I_x = I_y = \frac{W}{2} \left( R^2 + \frac{H^2}{6} \right) )</td>
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<td>Centroid = ( \bar{z} = \frac{H}{2} )</td>
<td>( I_z = \frac{R^2}{2} )</td>
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<td>( I_x = I_y = \frac{W}{6} \left( 3R^2 + 2H^2 \right) )</td>
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<td>Lateral Surface of a Circular Cone</td>
<td>Surface Area = ( \pi R^2 H )</td>
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<td>Right Circular Cylinder</td>
<td>Volume = ( \pi R^2 H )</td>
<td>( I_x = I_y = \frac{W}{12} \left( 2R^2 + \frac{H^2}{4} \right) )</td>
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<tr>
<td></td>
<td>Centroid = ( \bar{z} = \frac{H}{2} )</td>
<td>( I_z = \frac{R^2}{2} )</td>
</tr>
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<td>( I_x = I_y = \frac{W}{12} \left( 3R^2 + 2H^2 \right) )</td>
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<tr>
<td>Right Circular Cone</td>
<td>Volume = ( \frac{1}{3} \pi R^2 H )</td>
<td>( I_x = I_y = \frac{W}{12} \left( R^2 + \frac{H^2}{4} \right) )</td>
</tr>
<tr>
<td></td>
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<td>( I_z = \frac{R^2}{2} )</td>
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<td>Rectangular Prism</td>
<td>Volume = ( ABH )</td>
<td>( I_x = \frac{W}{12} \left( B^2 + H^2 \right) )</td>
</tr>
<tr>
<td></td>
<td>Centroid = ( \bar{z} = \frac{A}{2} )</td>
<td>( I_y = \frac{W}{12} \left( A^2 + H^2 \right) )</td>
</tr>
<tr>
<td></td>
<td></td>
<td>( I_z = \frac{W}{12} \left( A^2 + B^2 \right) )</td>
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