DESCRIPTION OF A COMPUTER PROGRAM WRITTEN FOR APPROACH AND LANDING TEST POST FLIGHT DATA EXTRACTION OF PROXIMITY SEPARATION AERODYNAMIC COEFFICIENTS AND AERODYNAMIC DATA BASE VERIFICATION

National Aeronautics and Space Administration
LYNDON B. JOHNSON SPACE CENTER
Houston, Texas
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SUMMARY

This report presents a description of a computer program written to calculate the proximity aerodynamic force and moment coefficients of the Orbiter/Shuttle Carrier Aircraft (SCA) vehicles based on flight instrumentation.

The Ground Reduced Aerodynamic Coefficients and Instrumentation Errors (GRACIE) program was developed as a tool to aid in flight test verification of the Orbiter/SCA separation aerodynamic data base. The program calculates the force and moment coefficients of each vehicle in proximity to the other, using the load measurement system (LMS) data, the flight instrumentation data (α, β, body rates, accelerations, etc.), and the vehicle mass properties. The uncertainty in each coefficient is determined, based on the quoted instrumentation accuracies. A subroutine, provided by McDonnell Douglas - Houston, manipulates the "Orbiter/747 Carrier Separation Aerodynamic Data Book" (reference 1) to calculate a comparable set of predicted coefficients for comparison to the calculated flight test data.

INTRODUCTION

In the Approach and Landing Test (ALT) phase of the shuttle program, one of the major problems to be considered is the separation of the Orbiter from the SCA. During mated flight, and for the first 3 sec after separation, the aerodynamics of each vehicle are influenced by the presence of the other vehicle. From past programs, it has been shown that such proximity effects are predicted with very little confidence from wind tunnel testing. Therefore, good flight test data are required to verify the adequacy of predicted separation windows and vehicle trajectories. The SCA is equipped with typical flight test instrumentation to record its absolute motion during flight. Load cells to measure the relative forces between the Orbiter and SCA are located on each of the three attach struts holding the Orbiter to the SCA during mated flight (fig. 1). Once the absolute motion of the SCA center of gravity

Figure 1. - Orbiter attach struts
(measured body attitudes, rates, and accelerations) and the externally applied
forces (attach forces and SCA engine thrust) are known, the aerodynamic forces
and moments of each vehicle can be determined by the relationships described
in the following sections.

SYMBOLS

[A] Transformation matrix to change from SCA body axis to orbiter
body axis coordinate system,
\[
\begin{bmatrix}
\cos i_o & 0 & -\sin i_o \\
0 & 1 & 0 \\
\sin i_o & 0 & \cos i_o
\end{bmatrix}
\]

[C] Vehicle aerodynamic coefficients

[F] Vehicle forces

[G] Transformation matrix to change from body axis to stability
axis,
\[
\begin{bmatrix}
-\cos \alpha_0 & 0 & -\sin \alpha_0 \\
0 & 1 & 0 \\
\sin \alpha_0 & 0 & \cos \alpha_0
\end{bmatrix}
\]

[I] Vehicle inertia matrix,
\[
\begin{bmatrix}
I_{xx} & 0 & -I_{xz} \\
0 & I_{yy} & 0 \\
-I_{xz} & 0 & I_{zz}
\end{bmatrix}
\]

\(i_o\) Orbiter incidence angle, deg

\(L\) Attach strut forces as measured by the load measurement
system, lb

\(\ell\) Vehicle reference length used for calculating vehicle moment
coefficients, ft

[M] Vehicle moments

\(m\) Vehicle mass, slugs

\(N_x, N_y, N_z\) Linear acceleration at vehicle center of gravity, g

\(\rho\) Vehicle roll rate, deg/sec

\(\dot{\rho}\) Vehicle roll acceleration, deg/sec^2
\( q \) \hspace{1cm} \text{Vehicle pitch rate, deg/sec}
\( \bar{q} \) \hspace{1cm} \text{Dynamic pressure, lb/ft}^2
\( \dot{q} \) \hspace{1cm} \text{Vehicle pitch acceleration, deg/sec}^2
\( R \) \hspace{1cm} \text{Vehicle position vector}
\( r \) \hspace{1cm} \text{Vehicle yaw rate, deg/sec}
\( \dot{r} \) \hspace{1cm} \text{Vehicle yaw acceleration, deg/sec}^2
\( S \) \hspace{1cm} \text{Vehicle reference area, ft}^2
\( T \) \hspace{1cm} \text{SCA thrust, lb}
\( V \) \hspace{1cm} \text{Velocity, ft/sec}
\( W \) \hspace{1cm} \text{Vehicle weight, lb}
\( \alpha \) \hspace{1cm} \text{Vehicle angle of attack, deg}
\( \beta \) \hspace{1cm} \text{Vehicle angle of sideslip, deg}
\( \gamma \) \hspace{1cm} \text{Vehicle flight path angle, deg}
\( \delta_{a_{I/O}} \) \hspace{1cm} \text{SCA aileron (inboard/outboard) position, deg}
\( \delta_{c_w} \) \hspace{1cm} \text{SCA control wheel position, deg}
\( \delta_{e_{I/O}} \) \hspace{1cm} \text{SCA elevator (inboard/outboard) position, deg}
\( \delta_{r_{U/L}} \) \hspace{1cm} \text{SCA rudder (upper/lower) position, deg}
\( \delta_{s} \) \hspace{1cm} \text{SCA stabilizer position, deg}
\( \delta_{SP} \) \hspace{1cm} \text{SCA spoiler panel (outboard/middle/inboard) position, deg}
\( \delta_{a_{o}} \) \hspace{1cm} \text{Orbiter aileron position, deg}
\( \delta_{e_{o_{L/R}}} \) \hspace{1cm} \text{Orbiter elevon (left/right) position, deg}
\( \delta_{r_{o}} \) \hspace{1cm} \text{Orbiter rudder position, deg}
\( \Delta N_x, \Delta N_y, \Delta N_z \) Relative load factors, g

\( \theta \) Vehicle pitch angle, deg

\( \ddot{\theta} \) Orbiter instantaneous pitch acceleration, deg/sec^2

\( \lambda \) Tilt angle of forward strut

\( \rho \) C.G. relative position vector \((\Delta x, \Delta y, \Delta z)\), ft

\( \rho \) C.G. to attach strut moment arm

\( \phi \) Vehicle roll angle, deg

\( \psi \) Vehicle yaw angle, deg

\( \omega \) Vehicle angular velocity vector \(-p, q, r\)

\( \ddot{\omega} \) Vehicle angular acceleration vector \(-\ddot{p}, \ddot{q}, \ddot{r}\)

**SUBSCRIPTS**

\( c \) SCA vehicle

\( F \) Forward

\( L \) Left

\( M \) Mated vehicle

\( O \) Orbiter vehicle

\( R \) Right

**PROGRAM DESCRIPTION AND ASSUMPTIONS**

GRACIE is a program that uses flight test data to determine aerodynamic coefficients and their corresponding uncertainties for comparison with wind tunnel predicted values. The program manipulates LMS forces, SCA body motions, vehicle configurations, and vehicle mass properties to output tabulated and plotted time histories of Orbiter proximity, SCA proximity, and mated vehicle aerodynamic force and moment coefficients, as well as, relative normal load factor \( \Delta N_z \) and Orbiter instantaneous pitch acceleration \( \ddot{\theta}_O \). The LMS data, the SCA body motion data, and the vehicle configuration are obtained from a ground recorded telemetry data tape to which all instrumentation calibrations have been applied. The vehicle mass properties and the SCA thrust data time
histories are input through subroutines, since they require post flight calculations and are not recorded on the data tape.

The program performs three basic operations utilizing the flight test data. The equations of motion and the aerodynamic uncertainty calculations are made using data retrieved from flight test instrumentation, while the predicted values of the coefficients are determined using the algorithms and data presented in reference 1. The following sections describe these operations.

Equations of Motion

As a basis for this evaluation, the mated vehicle is assumed to be a rigid body in motion with respect to a fixed coordinate system XYZ (fig. 2).

Figure 2. - Vehicle coordinate system

Affixing a second set of axes to the carrier aircraft, with the origin (C) located at the carrier c.g. and observing its motion, allows evaluation of the motion of any other point in the mated configuration, namely the Orbiter c.g., as well as the mated c.g. For example, the acceleration of the Orbiter c.g. (0) can be determined by knowing the relative position (0), the linear acceleration (\(\ddot{R}\)), angular rates (\(\omega\)), and angular accelerations (\(\dot{\omega}\)) of the carrier c.g. (C),

\[
\ddot{R}_o = \ddot{R}_c + \dot{\omega}_c \times \omega_c + \omega_c \times (\omega_c \times 0) + \alpha_{o/xyz} + 2\dot{\omega}_c \times \nu_{o/xyz}.
\]
but since the mated vehicle is assumed to be a rigid body

\[ \alpha_{o/xyz} = v_{o/xyz} = 0 \]

Therefore,

\[ \ddot{R}_o = \ddot{R}_c + \dot{\omega}_c \times \omega_c + \omega_c \times (\omega_c + \dot{\omega}_c). \]

The total resultant or applied forces on either vehicle are then:

\[ F_{c\text{TOTAL}}^{\text{APPLIED}} = m_c \ddot{R}_c \]

\[ F_{o\text{TOTAL}}^{\text{APPLIED}} = m_o \ddot{R}_o \]

and

\[ F_{m\text{TOTAL}}^{\text{APPLIED}} = m_m \ddot{R}_m \]

Similar use of kinematics provides the equations for calculating the resultant moments (M) on each vehicle, i.e.

\[ M_{c\text{TOTAL}}^{\text{APPLIED}} = [I]_c \dot{\omega}_c + \omega_c \times [I]_c \omega_c, \]

\[ M_{o\text{TOTAL}}^{\text{APPLIED}} = [I]_o \dot{\omega}_o + \omega_o \times [I]_o \omega_o, \]

and

\[ M_{m\text{TOTAL}}^{\text{APPLIED}} = [I]_m \dot{\omega}_m + \omega_m \times [I]_m \omega_m. \]

where,

\[ \omega_o = [A] \omega_c, \quad \dot{\omega}_o = [A]_c \dot{\omega} \]
and \( \omega_m = \omega_c, \dot{\omega}_m = \dot{\omega}_c \)

From figure 3, the load cell outputs, expressed in the carrier body axis coordinate system, are as follows:

\[
\begin{align*}
F_F^y & = \text{Forward side force} \\
F_F^z & = \text{Forward vertical force (parallel to strut axis)} \\
F_F^x, F_F^z & = \text{Drag and vertical components of forward vertical strut force (carrier body axis coordinate system)} \\
F_L^x & = \text{Left aft drag force} \\
F_L^z & = \text{Left aft vertical force} \\
F_R^x & = \text{Right aft drag force} \\
F_R^y & = \text{Right aft side force} \\
F_R^z & = \text{Right aft vertical force}
\end{align*}
\]

where, \( F_F^x = F_F^z \sin \lambda, F_F^z = F_F^z \cos \lambda \)

\[
\lambda = 88.27^\circ - \sin^{-1} \left[ \frac{929.098 \sin(i_o + 2.734^\circ)}{1723336.5 - 1723333.7 \cos(i_o + 2.734^\circ)} \right]
\]
Also shown in figure 3 are the moment arms from the orbiter c.g. to each load cell attach point based on the carrier body axis coordinate system. Using figure 3 in conjunction with figure 4, the moment arms are determined from the following relations, making note of the fact that the attach point locations are in the orbiter body coordinate system:

\[ \phi_F = \tan^{-1}\left(\frac{Z_{cg_o} - Z_F}{X_{cg_o} - X_F}\right) \]

\[ \phi_A = \tan^{-1}\left(\frac{Z_{cg_o} - Z_R}{X_R - X_{cg_o}}\right) \]

\[ L_F = \frac{Z_{cg_o} - Z_R}{\sin \phi_F} \]
\[ L_A = \frac{Z_{CGO} - Z_R}{\sin \phi_A} \]

\[ l_1 = L_A \sin(\phi_A + i_0)/12 \]

\[ l_2 = L_A \cos(\phi_A + i_0)/12 \]

\[ l_3 = L_F \sin(\phi_F - i_0)/12 \]

\[ l_4 = L_F \cos(\phi_F - i_0)/12 \]

\[ l_5 = -(Y_L + Y_{CGO})/12 \]

\[ l_6 = (Y_R - Y_{CGO})/12 \]

From figure 5, and using the Orbiter moment arms previously calculated, the

\textbf{Figure 5. - Relative c.g. locations}
position vector is:

\[
\mathbf{\phi} = \begin{bmatrix}
\frac{1}{2} - \frac{(x_s - x_{cg747})}{12} \\
\frac{(y_{cg0} - y_{cg747})}{12} \\
\frac{(z_{cg747} - z_s)}{12} - 1
\end{bmatrix}
\]

Note that in figure 5, the attach point locations are in the carrier body axis coordinate system.

The following free body diagrams and corresponding equations of motion are used in calculating the aerodynamic force and moment coefficients of the mated vehicle, the SCA in proximity of the orbiter, and the orbiter in proximity of the SCA.

**Mated vehicle aerodynamic coefficients.** - Force coefficients (drag, side force, lift)

\[
F_{\text{TOTAL APPLIED}} = F_{\text{AERO}} + F_{\text{THRUST}}
\]

\[
F_{\text{TOTAL APPLIED}} = m_m \ddot{R}_m = m_m \left[ \ddot{R}_c + \omega_c \times \dot{\omega}_m + \omega_c \times (\omega_c \times \omega_m) \right]
\]
\[ \mathbf{F}_{\text{AERO}} - m \ddot{\mathbf{r}}_m - \mathbf{F}_{\text{THRUST}} \]

\[ C_{\text{BODY AXIS}} = \frac{F_{\text{AERO}}}{\frac{q}{2} S_c} \]

\[ C_{\text{STABILITY AXIS}} = \begin{bmatrix} \mathbf{G} \end{bmatrix} C_{\text{BODY AXIS}} \]

\[ \mathbf{G} = \text{Transformation from Carrier BODY AXIS TO Carrier STABILITY AXIS.} \]

**Mated vehicle aerodynamic coefficients.** - Moment coefficients (rolling moment, pitching moment, yawing moment)

\[ M_{\text{TOTAL APPLIED}} = M_{\text{AERO APPLIED}} + M_{\text{THRUST APPLIED}} \]

\[ M_{\text{TOTAL APPLIED}} = \begin{bmatrix} I \end{bmatrix}_m \dot{\omega}_c + \omega_c \begin{bmatrix} I \end{bmatrix}_m \omega_c \]

\[ M_{\text{AERO APPLIED}} = M_{\text{TOTAL APPLIED}} - M_{\text{THRUST APPLIED}} \]

\[ C_{\text{MOMENT}} = \frac{M_{\text{AERO APPLIED}}}{\frac{q}{2} S_c \omega_c} \]
Carrier aerodynamic coefficients (proximity). - Force coefficients (drag, side force, lift)

\[ F_{\text{TOTAL APPLIED}} = F_{\text{AERO}} + F_{\text{LOAD CELL}} + F_{\text{THRUST}} \]

\[ F_{\text{TOTAL APPLIED}} = m_c \ddot{R}_c \]

\[ F_{\text{LOAD CELL}} = L_c = L_{\text{fwd}} + L_{\text{left}} + L_{\text{right}} \]

\[ F_{\text{AERO}} = m_c \ddot{R}_c - F_{\text{LOAD CELL}} - F_{\text{THRUST}} \]

\[ C_{\text{BODY AXIS}} = \frac{F_{\text{AERO}}}{q S_c} \]

\[ C_{\text{STABILITY AXIS}} = [G]_c C_{\text{BODY AXIS}} \]
Carrier aerodynamic coefficients (proximity). - Moment coefficients (rolling moment, pitching moment, yawing moment)

\[ M_{\text{TOTAL APPLIED}} = M_{\text{AERO}} + M_{\text{LOAD CELL}} + M_{\text{THRUST APPLIED}} \]

\[ M_{\text{TOTAL}} = \sum_{s=1}^{3} \left( \rho_s x L_s \right) \]

\[ M_{\text{LOAD CELL}} = M_{\text{TOTAL APPLIED}} - M_{\text{LOAD CELL}} - M_{\text{THRUST}} \]

\[ C_{\text{MOMENT}} = \frac{M_{\text{AERO}}}{q c^2} \]

Orbiter aerodynamic coefficients (proximity). - Force coefficients (drag, side force, lift)
\[ F_{\text{TOTAL}} = F_{\text{AERO}} + F_{\text{LOAD}} \]

\[ F_{\text{TOTAL}} = m_o \ddot{R}_o = m_o \left[ \ddot{R}_c + \dot{\omega}_c \times R + \omega_c \times (\omega_c \times R) \right] \]

\[ F_{\text{LOAD}} = L_o = L_{\text{fwd}} + L_{\text{left}} + L_{\text{right}} \]

\[ F_{\text{AERO}} = [A] \left[ m_o \ddot{R}_o - L_o \right] \]

\[ C_{\text{BODY AXIS}} = \frac{F_{\text{AERO}}}{\dot{q} S_o} \]

\[ C_{\text{STABILITY}} = [\theta]_o C_{\text{BODY AXIS}} \]

\[ [A] = \text{Transformation from CARRIER to ORBITER coordinate system at INCIDENCE angle } \theta_o. \]

**Orbiter aerodynamic coefficients (proximity). - Moment coefficients**

(rolling moment, pitching moment, yawing moment)

\[ M_{\text{TOTAL APPLIED}} = M_{\text{AERO}} - M_{\text{LOAD CELL}} \]

\[ M_{\text{TOTAL APPLIED}} = [I]_o \dot{\omega}_o + \omega_o \times [I]_o \omega_o \]

\[ M_{\text{LOAD CELL}} = \sum_{s=1}^{3} (L_{s_o} \times \omega_{s_o}) \]
\[ M_{\text{AERO}} = M_{\text{TOTAL}} - M_{\text{LOAD}} \]

\[ C_{\text{MOMENT}} = \frac{M_{\text{AERO}}}{q \cdot S_0 \cdot v_0} \]

**ORBITER PITCH ACCELERATION**

\[ \ddot{\theta}_{\text{ORB}} = \frac{M_{\text{AEROy}}}{I_{yy0}} \]

**RELATIVE LOAD FACTORS**

\[ \Delta N_z = L_0 \left( \frac{W_o + W_c}{W_o} \right) \]

---

**Aerodynamic Uncertainties**

An integral part of the separation analysis is knowing the uncertainty associated with each coefficient and how that uncertainty affects the size of the separation window, as well as the vehicle trajectory. Each aerodynamic coefficient is a function of \( i \) independent measurements, \( n_i \), and the uncertainty of each measurement is \( \Delta n_i \).

\[ C = f(n_1, n_2, n_3, \ldots, n_i) \]  \hspace{1cm} (1)

The uncertainty in each calculated coefficient is obtained using the following equation:

\[ \Delta C = \left[ \left( \frac{\partial C}{\partial n_1} \right)^2 (\Delta n_1)^2 + \left( \frac{\partial C}{\partial n_2} \right)^2 (\Delta n_2)^2 + \ldots + \left( \frac{\partial C}{\partial n_i} \right)^2 (\Delta n_i)^2 \right]^{1/2}. \]  \hspace{1cm} (2)

The uncertainties in the aerodynamic coefficients are based on the quoted accuracies of the load measurement system and the flight test instrumentation.
The uncertainty in the Orbiter force and moment coefficients are calculated as follows. The Orbiter aerodynamic forces are first calculated with respect to the SCA body coordinate system from the following equations:

\[
C_X = \frac{M_o[N_x + \dot{q} \Delta z - \dot{r} \Delta y + q p \Delta y - q^2 \dot{\Delta} x - r \Delta z - r^2 \Delta x]}{\bar{q} S_o} - \sum F_X
\]

\[
C_Y = \frac{M_o[N_y - \dot{p} \Delta z + \dot{r} \Delta x - p^2 \Delta y + p q \Delta x + r q \Delta y - r^2 \Delta y]}{\bar{q} S_o} - \sum F_Y
\]

\[
C_Z = \frac{M_o[N_z + \dot{p} \Delta y - \dot{q} \Delta x - p^2 \Delta z + p r \Delta x - q^2 \Delta z + q r \Delta y]}{\bar{q} S_o} - \sum F_Z
\]

From equation (2), the uncertainty in \( C_X \) is:

\[
N_X' = \left[ \frac{M_o}{\bar{q} S_o} \right]^2 \left[ \Delta N_X \right]^2
\]

\[
p' = \left[ \frac{M_o(q \Delta y + r \Delta z)}{\bar{q} S_o} \right]^2 \left[ \Delta p \right]^2
\]

\[
q' = \left[ \frac{M_o(p \Delta y - 2q \Delta x)}{\bar{q} S_o} \right]^2 \left[ \Delta q \right]^2
\]

\[
r' = \left[ \frac{M_o(p \Delta z - 2q \Delta x)}{\bar{q} S_o} \right]^2 \left[ \Delta r \right]^2
\]

\[
p' = \text{------}
\]
\[ \dot{q}' = \left( \frac{M_0 \Delta z}{q S_0} \right)^2 [\Delta \dot{q}]^2 \]

\[ \dot{r}' = \left( \frac{M_0 \Delta y}{q S_0} \right)^2 [\Delta \dot{r}]^2 \]

\[ \ddot{q}' = \left[ -M_0 (N_x + \dot{q} \Delta z - \dot{r} \Delta y - q \Delta y - q^2 \Delta x + r p \Delta z - r^2 \Delta x) + L_{XT} \right] \frac{2}{q^2 S_0} [\Delta \ddot{q}]^2 \]

\[ F_{F_X}' = \left( \frac{1}{q S_0} \right)^2 [\Delta F_{F_X}]^2 \]

\[ F_{L_X}' = \left( \frac{1}{q S_0} \right)^2 [\Delta F_{L_X}]^2 \]

\[ F_{R_X}' = \left( \frac{1}{q S_0} \right)^2 [\Delta F_{R_X}]^2 \]

\[ \Delta C_X = [N_x' + p' + q' + r' + \dot{p}' + \dot{q}' + \dot{r}' + \ddot{q}' + F_{F_X}' + F_{L_X}' + F_{R_X}']^{1/2} \]

Similarly, the uncertainty in \( C_Z \) is calculated. The coefficients are then transformed into the Orbiter body axis coordinate system.

\[ C_{A_0} = C_X \cos(i_0) - C_Z \sin(i_0) \]

\[ C_{N_0} = C_X \sin(i_0) + C_Z \cos(i_0) \]
The uncertainties in these two coefficients are:

\[ \Delta C_{A_o} = [(\cos i_o)^2(\Delta C_x)^2 + (\sin i_o)^2(\Delta C_z)^2]^{1/2}, \]

\[ \Delta C_{N_o} = [(\sin i_o)^2(\Delta C_x)^2 + (\cos i_o)^2(\Delta C_z)^2]^{1/2}. \]

Finally, these coefficients are transformed into the Orbiter stability axis coordinate system,

\[ C_{D_0} = C_{A_o} \cos(\alpha_c + i_o) + C_{N_o} \sin(\alpha_c + i_o), \]

\[ C_{L_0} = -C_{A_o} \sin(\alpha_c + i_o) + C_{N_o} \cos(\alpha_c + i_o), \]

and the uncertainties in the Orbiter coefficients of lift and drag are:

\[ \Delta C_{L_0} = [(\cos(\alpha_c + i_o))^2(\Delta C_{A_o})^2 + (\sin(\alpha_c + i_o))^2(\Delta C_{N_o})^2 \]

\[ + (C_{A_o} \sin(\alpha_c + i_o) - C_{N_o} \cos(\alpha_c + i_o))^2(\Delta \alpha)^2]^{1/2} \]

\[ \Delta C_{D_0} = [(\sin(\alpha_c + i_o))^2(\Delta C_{A_o})^2 + (\cos(\alpha_c + i_o))^2(\Delta C_{N_o})^2 \]

\[ + (C_{A_o} \cos(\alpha_c + i_o) + C_{N_o} \sin(\alpha_c + i_o))^2(\Delta \alpha)^2]^{1/2}. \]

The uncertainty in the Orbiter side force coefficient, \( \Delta C_{Y_0} \), is found in the same way.

The Orbiter moment coefficients are based on the following equations:
\[
\begin{align*}
C_{m_x} &= \frac{\dot{q}_o I_{xx} + q_o r_o (I_{zz} - I_{yy}) + I_{xz} (\dot{r}_o + q_o p_o)}{qS_o b_o} \\
   &+ \frac{F_{FZ} \rho_{FY} - F_{FY} \rho_{FZ} - F_{LZ} \rho_{LY} - F_{RZ} \rho_{FY} + F_{RY} \rho_{RZ}}{qS_o b_o} \\
C_{m_y} &= \frac{\dot{q}_o I_{yy} + p_o r_o (I_{xx} - I_{zz}) + I_{xz} (r_o^2 - p_o^2)}{qS_o b_o} \\
   &+ \frac{F_{FZ} \rho_{FX} - F_{FX} \rho_{FZ} + F_{LZ} \rho_{LX} - F_{RZ} \rho_{RX} - F_{RX} \rho_{RZ}}{qS_o b_o} \\
C_{m_z} &= \frac{\dot{r}_o I_{zz} + p_o q_o (I_{yy} - I_{xx}) - I_{xz} (q_o r_o - \dot{p}_o)}{qS_o b_o} \\
   &+ \frac{F_{FY} \rho_{FX} + F_{FX} \rho_{FY} + F_{LX} \rho_{LX} - F_{RY} \rho_{RX} + F_{RX} \rho_{RY}}{qS_o b_o}
\end{align*}
\]

Again using equation (2), the uncertainty in the Orbiter pitching moment is:

\[
\begin{align*}
\dot{q}^2 &= \left[\frac{4}{qS_o c_0} \Delta q \right]^2 \\
p^2 &= \left[\frac{r_o (I_{xx} - I_{zz}) + 2 p_o I_{xz} - 2 p_o I_{xz}}{qS_o c_0} \right]^2 \Delta p \right]^2 \\
r^2 &= \left[\frac{r_o (I_{xx} - I_{zz}) - 2 r_o I_{xz}}{qS_o c_0} \right]^2 \Delta r \right]^2
\end{align*}
\]
Similarly, the uncertainties in the rolling moment and yawing moment are calculated.

\[
\tilde{q'} = \frac{q_{ix} I_{yy} + p_0 r_0 (I_{xx} - I_{xz}) + I_{xz} (r_o^2 - p_o^2)}{q^2 S_o \sigma} \\
+ \frac{F_{FZ} \rho_{FX} - F_{FX} \rho_{FZ} + F_{LZ} \rho_{LX} - F_{LX} \rho_{LZ} + F_{RZ} \rho_{RX} - F_{RX} \rho_{RZ}}{q^2 S_o \sigma^2} \left[ \Delta q \right]^2
\]

\[
F_{FZ} = \left[ \frac{\rho_{FX}}{q S_o \sigma} \right]^2 [\Delta F_{FZ}]^2
\]

\[
F_{FX} = \left[ \frac{\rho_{FZ}}{q S_o \sigma} \right]^2 [\Delta F_{FX}]^2
\]

\[
F_{LZ} = \left[ \frac{\rho_{LZ}}{q S_o \sigma} \right]^2 [\Delta F_{LZ}]^2
\]

\[
F_{LX} = \left[ \frac{\rho_{LX}}{q S_o \sigma} \right]^2 [\Delta F_{LX}]^2
\]

\[
F_{RZ} = \left[ \frac{\rho_{RX}}{q S_o \sigma} \right]^2 [\Delta F_{RZ}]^2
\]

\[
F_{RX} = \left[ \frac{\rho_{RX}}{q S_o \sigma} \right]^2 [\Delta F_{RX}]^2
\]

\[
\Delta C_{m_y} = [\tilde{q'} + p' + r' + \tilde{q'} + F_{FZ} + F_{FX} + F_{LZ} + F_{LX} + F_{RZ} + F_{RX}]^{1/2}
\]
Analysis of the uncertainties in the SCA proximity and mated vehicle coefficients is performed in a like manner.

Predictions

One of the subroutines in the program uses the Algorithms presented in reference 1 and reference 2 to build-up force and moment coefficients for each vehicle using contributing elements, such as control surface deflections and proximity effects, etc. The data is a digitized version of the data presented in references 1 and 2 and stored in look-up tables in the program. Input to this subroutine comes from configuration and attitude parameters recorded on the flight data tape and orbiter elevon position time histories.

PROGRAM DECK SET-UP

Data needed to run GRACIE, other than that on the flight data tape, are input using subroutines and data cards. Two subroutines require changes for reducing data from each flight.

The first subroutine is called ORBDEF and is used to provide orbiter control surface deflection time histories.

```fortran
SUBROUTINE ORBDEF

SUBROUTINE ORBDEF(TIME,LELV,RELV,ORUD,I),DELZ,TC,DELBF
SUBROUTINE TO PROVIDE ORBITER CONTROL SURFACE DEFLECTIONS

REAL LELV,IO
TIME = TIME - 30766.
IF(TI.LT.11.6) LELV = 0.
IF(TI.GE.11.6.AND.TI.LT.16.6) LELV = 1.7
IF(TI.GE.16.6.AND.TI.LT.17.6) LELV = 0.
IF(TI.GE.17.6.AND.TI.LT.27.5) LELV = -1.3
IF(TI.GE.27.5.AND.TI.LT.3.4) LELV = 0.
IF(TI.GE.3.4.AND.TI.LT.44.4) LELV = 1.
IF(TI.GE.44.4) LELV = 0.
IF(TI.LT.11.9) RELV = 0.
IF(TI.GE.11.9.AND.TI.LT.16.8) RELV = 1.7
IF(TI.GE.16.8.AND.TI.LT.27.7) RELV = 0.
IF(TI.GE.27.7.AND.TI.LT.3.7) RELV = -1.3
IF(TI.GE.3.7.AND.TI.LT.44.6) RELV = 0.
ORUD = 0.0
IO = 0.0
DELZ = 0.0
TC = 1.0
DELBF = -9.7
RETURN
END
```
Where:

TI = reference time for deflection time histories (REAL)
TIME = time from data tape (REAL)
LELV = left elevon position in degrees (REAL)
RELV = right elevon position in degrees (REAL)
ORUD = rudder position in degrees (REAL)
IO = orbiter incidence angle in degrees (REAL)
DELZ = relative normal displacement between the orbiter and carrier in feet, 0 feet is defined as mated (REAL)
TC = tailcone designation 1. = tailcone on
0. = tailcone off

DELBF = orbiter body flap position in degrees (REAL)

The subroutine returns control surface positions for use in the main program for each time step on the flight data tape.

The second subroutine is called TRUST and provides a time history of the carrier thrust for portions of the flight of interest.

SUBROUTINE TRUST 73/74 OPT=1

SUBROUTINE TRUST(TIME,THRUST,TZ)
C
C ***** SUBROUTINE TRUST PROVIDE CARRIER THRUST *****
C ***** THRUST FOR CA-1 SEP DATA RUN
5
TZ = 0.
THRUST = -1108.
RETURN
END

Where:

TZ = The normal component of thrust which is always 0.
THRUST = Axial component of thrust for all four carrier engines in pounds. (REAL)
Three data cards are also required to run GRACIE. They are used to input mated vehicle gross weight, data locations on the data tape, print-plot options, and plot titles.

The first data card contains the mated vehicle gross weight in thousand pounds. The example card represents 523,000 lb.

The second data card contains the first point location on the data tape, the final point location on the data tape, and the print plot option. For the print plot option:

1 = print tabulated listing and plot results

2 = print tabulated data only

3 = plot results only
The example card asks for data from file 1 on the data tape to file 220, and for both tabulated listings and plotted results. Examples of the tabulated listings and plotted data are found in appendix A and appendix B, respectively.

The third and final data card is used for titling the plotted results. The titles are plotted in blocks of 30 figures. The example card below prints the title as shown on the example plots in appendix B.
APPENDIX A

TAB DATA
TIME
Time of day (pacific time) - hours minutes seconds hundredths

COEF.
Vehicle aerodynamic coefficients

  O  - Orbiter
  C  - Carrier (SCA)
  M  - Mated

CMR - Vehicle pitching moment about the vehicle moment reference center

NOM.
Coefficients as calculated from GRACIE

I+L
Absolute value of the uncertainty due to instrumentation inaccuracies plus the uncertainty due to LMS inaccuracies

I-L
Absolute value of the uncertainty due to instrumentation inaccuracies minus the uncertainty due to LMS inaccuracies

DATA BOOK
Predicted coefficients based on the "Orbiter/747 Carrier Separation Aerodynamic Data Book".

RT. ELV
Right hand inboard Orbiter elevon position (deg)

LT. ELV
Left hand inboard Orbiter elevon position (deg)

DELNZ
Relative normal acceleration ± uncertainty, predicted (g)

QDOTO
Instantaneous Orbiter pitch acceleration ± uncertainty, predicted (deg/sec^2)

INCIDENCE
Orbiter incidence angle (deg)

ALPHAC
SCA angle of attack (deg)

ALPHAO
Orbiter angle of attack (deg)

BETA
SCA angle of sideslip (deg)

GAMMA
SCA flight path angle (deg)

THETA
SCA pitch angle (deg)

ROLL
SCA roll angle (deg)

YAW
SCA yaw angle (deg)

ORBAIL
Orbiter aileron angle (deg)
ORBRUD  Orbiter rudder position (deg)
QBAR    Dynamic pressure (lb/ft²)
ALT     Altitude (MSL ft)
VKEAS   Velocity (knots equivalent airspeed)
VFPS    True airspeed (ft per sec)
NLF     Mated vehicle load factor (g)
XCGC    SCA cg location in body coordinate system (in.)
YCGC    
ZCGC    
NX      SCA cg linear acceleration (ft/sec²)
NY      
NZ      
P       SCA body angular rates (deg/sec)
Q       
R       
PD      SCA body angular accelerations (deg/sec²)
QD      
RD      
FFY     
FFZ     
FLX     
FLZ     LMS attach forces (lb)
FRX     
FRY     
FRZ     
HSTAB   SCA horizontal stabilizer position (deg)
ELVI/O  SCA elevator (inboard/outboard) position (deg)
RUDU/L  SCA rudder (upper/lower) position (deg)
SPO/M/I SCA spoiler panel (outboard/middle/inboard) position (deg)
AILI/O  SCA aileron (inboard/outboard) position (deg)
CW      SCA control wheel position (deg)
<table>
<thead>
<tr>
<th>Variable</th>
<th>Description</th>
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</thead>
<tbody>
<tr>
<td>WTC</td>
<td>SCA weight (lb) and inertias (slug-ft²)</td>
</tr>
<tr>
<td>IIIIC</td>
<td></td>
</tr>
<tr>
<td>IYYC</td>
<td></td>
</tr>
<tr>
<td>IIIIC</td>
<td></td>
</tr>
<tr>
<td>IYXZ</td>
<td>Orbiter weight (lb) and inertias (slug-ft²)</td>
</tr>
<tr>
<td>IIIIX</td>
<td></td>
</tr>
<tr>
<td>IIIYO</td>
<td></td>
</tr>
<tr>
<td>IIIIZ</td>
<td></td>
</tr>
<tr>
<td>IXZI</td>
<td>SCA total engine thrust (lb)</td>
</tr>
</tbody>
</table>

Average coefficient values for previous twenty samples are tabulated after twentieth time step in the following format:

\[
\text{COEFFICIENT} = \frac{\text{AVERAGE NAME VALUE}}{\text{AVERAGE DATA BOOK VALUE}}
\]
APPENDIX B

PLOTS
PROGRAM PLOTTING CAPABILITY

GRACIE generates time history plots of the aerodynamic coefficients for each vehicle, the angle of attack of each vehicle, the relative normal load factor ($\Delta N_z$), and the instantaneous pitch acceleration of the Orbiter ($\dot{\alpha}$). With minor modifications, the program has the capability to plot any input or calculated parameter. As an example, GRACIE generated the following set of plots.
GRACIE (D.J. HOMAN)

CAPTIVE ACTIVE FLIGHT No. II
ELEVON BIAS = -1.0
BODY FLAP = -9.7
TAIL CONE ON
Cp0 = .638
INCIDENCE = .6
GRACIE (D.L. HOMAN)

<table>
<thead>
<tr>
<th>CAPTIVE ACTIVE FLIGHT NO.</th>
<th>ELEVON BIAS</th>
<th>BODY FLAP</th>
<th>TAIL CONE ON</th>
<th>C.G.</th>
<th>INCIDENCE</th>
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<tr>
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<table>
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<tr>
<td>30.774.50</td>
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<tr>
<td>30.775.00</td>
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</table>

---

The diagram shows various line graphs representing different parameters such as time and other flight data. The specific details of the graphs are not clearly visible due to the image quality.
GRACIE (JOHN HUMAN)
CAPTIVE ACTIVE FLIGHT NO. 1
ELEVON BIAS = -1.0
BODY FLAP = -9.7
TAIL CONE ON
C.G. = 63.8
INCIDENCE = "6.
**GRACIE (O.J. HOMAN)**

**Captive Active Flight No. 1**

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<th>C.G.</th>
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<td>-1.0</td>
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**Graph and Data**

Time: 0 - 3074.00 seconds

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<tr>
<td>TIME</td>
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<table>
<thead>
<tr>
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<th>CARRIER</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Captive Active Flight No. 1
Elevon Bias = -1.0
Body Flap = -9.7
Tail Cone On
C.G. = 63.9
Incidence = 6
**GRACIE (D. J. HOMAN)**

| Time (sec) | 0.00 | 0.02 | 0.04 | 0.06 | 0.08 | 0.10 | 0.12 | 0.14 | 0.16 | 0.18 | 0.20 | 0.22 | 0.24 | 0.26 | 0.28 | 0.30 | 0.32 | 0.34 | 0.36 | 0.38 | 0.40 | 0.42 | 0.44 | 0.46 | 0.48 | 0.50 | 0.52 | 0.54 | 0.56 | 0.58 | 0.60 | 0.62 | 0.64 | 0.66 | 0.68 | 0.70 | 0.72 | 0.74 | 0.76 | 0.78 | 0.80 | 0.82 | 0.84 | 0.86 | 0.88 | 0.90 | 0.92 | 0.94 | 0.96 | 0.98 | 1.00 |
|------------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|------|

**Captive Active Flight No.**

- **Elevator Bias:** -1.0
- **Body Flap:** -9.7
- **Tail Cone On:**
- **C.G.:** .648
- **Incidence:** .

**Carrier**

- **TIME:** .SEC
GRACIE (D. J. HOMAN)

CAPTIVE ACTIVE FLIGHT NO. 1

ELEVON BIAS = ±1.0
BODY FLAP = ±9.7
TAIL CONE ON
C.G. = 63.8
INCIDENCE = 6

TIME: SEC
GRACIE (D.J. HOMAN)

CAPTIVE ACTIVE FLIGHT NO. 1
ELEVON BIAS = -1.0
BODY FLAP = -9.7
TAIL CONE ON
C.O. = 63.8
INCIDENCE = 6.
GRACIE (D.J. HOMAN)

CAPTIVE ACTIVE FLIGHT NO. 1

ELEVON BIAS = -1.0

BODY FLAP = -9.7

TAIL CONE ON

C/G = 638

INCIDENCE =

TIME: SEC.
GRACIE (D.J. HOMON)

CAPTIVE ACTIVE FLIGHT NO. 1
ELEVON BIAS = 1.0
BODY FLAP = 4.7
TAIL CONE ON
C.O.G. = 63.8
INCIDENCE = 6
**Gracie (D.J. Homan)**

- **Captive Active Flight No.**
- **Elevator Bias = -1.0**
- **Body Flap = -9.2**
- **Tail Cone On**
- **C.G. = 53.6**
- **Incidence = 6°**

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Note: The data above are typical of a flight test scenario showing various aerodynamic derivatives over time.
GRACIE (D.J. HUMAN)

Captive active flight no. 1

Elevator bias = -1.0

Body flap = -9.7

Tail cone on

C.G. = 63.8

Incidence = 6

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