IDENTIFICATION OF SPEY ENGINE DYNAMICS IN THE AUGMENTOR WING JET STOL RESEARCH AIRCRAFT FROM FLIGHT DATA

Prepared for
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<table>
<thead>
<tr>
<th>TABLE OF CONTENTS</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>I.</strong> INTRODUCTION AND SUMMARY</td>
</tr>
<tr>
<td>1.1 INTRODUCTION</td>
</tr>
<tr>
<td>1.2 SUMMARY</td>
</tr>
<tr>
<td><strong>II.</strong> MODELING THE SPEY ENGINE</td>
</tr>
<tr>
<td>2.1 INTRODUCTION</td>
</tr>
<tr>
<td>2.2 TYPES OF ENGINE MODELS</td>
</tr>
<tr>
<td>2.3 CURRENT SPEY ENGINE MODEL IN THE AWJSTOLRA SIMULATION</td>
</tr>
<tr>
<td>2.4 DETAILED MODEL OF THE ENGINE</td>
</tr>
<tr>
<td>2.5 SIMPLIFIED MODEL</td>
</tr>
<tr>
<td>2.6 SENSOR MODELS</td>
</tr>
<tr>
<td>2.7 SUMMARY</td>
</tr>
<tr>
<td><strong>III.</strong> PRELIMINARY PARAMETER ESTIMATES</td>
</tr>
<tr>
<td>3.1 SUMMARY OF FLIGHT DATA</td>
</tr>
<tr>
<td>3.2 INITIAL PARAMETER ESTIMATES</td>
</tr>
<tr>
<td>3.2.1 Steady State Operation</td>
</tr>
<tr>
<td>3.2.2 Throttle Hysteresis</td>
</tr>
<tr>
<td>3.2.3 Rate Limiting</td>
</tr>
<tr>
<td>3.2.4 Fan and Compressor Dynamics</td>
</tr>
<tr>
<td>3.2.5 Control Parameters</td>
</tr>
<tr>
<td>3.2.6 Summary of Initial Parameter Estimates</td>
</tr>
<tr>
<td><strong>VI.</strong> PARAMETER IDENTIFICATION PROGRAM</td>
</tr>
<tr>
<td>4.1 THEORETICAL BACKGROUND</td>
</tr>
<tr>
<td>4.2 STATISTICAL PROPERTIES</td>
</tr>
<tr>
<td><strong>V.</strong> RESULTS OF PARAMETER IDENTIFICATION</td>
</tr>
<tr>
<td>5.1 THROTTLE IDENTIFICATION</td>
</tr>
<tr>
<td>5.2 IDENTIFICATION OF PRESSURE DYNAMICS</td>
</tr>
<tr>
<td>5.3 IDENTIFICATION OF ENGINE PARAMETERS</td>
</tr>
<tr>
<td>5.4 IDENTIFICATION OF CONTROL PARAMETERS</td>
</tr>
<tr>
<td>5.5 SUMMARY</td>
</tr>
<tr>
<td><strong>VI.</strong> SUMMARY AND CONCLUSIONS</td>
</tr>
<tr>
<td>6.1 SUMMARY</td>
</tr>
<tr>
<td>6.2 CONCLUSIONS</td>
</tr>
<tr>
<td>REFERENCES</td>
</tr>
<tr>
<td>-----------------------------------</td>
</tr>
<tr>
<td>APPENDIX A - LISTING OF SPEY ENGINE SIMULATION AND IDENTIFICATION PROGRAM</td>
</tr>
<tr>
<td>APPENDIX B - FLIGHT PLAN: S2-E13</td>
</tr>
<tr>
<td>APPENDIX C - TIME HISTORY MATCHES FOR PARAMETER IDENTIFICATION RUNS</td>
</tr>
</tbody>
</table>
CHAPTER I
INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

The Augmentor Wing Jet STOL Research Aircraft (AWJSTOLRA) is an advanced test vehicle for STOL powered-lift flight testing. An initial area of interest is the control of the AWJSTOLRA to closely track an altitude profile during approach and flare maneuvers. For a powered lift vehicle during landing approach, the throttle may be used as the primary control of normal acceleration. The accurate digital simulation of aircraft maneuvers involving throttle changes and the design of autopilot logic using feedback to the throttle require an accurate model of the dynamic response of engine thrust to changes in throttle commands. This model represents a complex interaction of nonlinear static and dynamic behavior in the engine control and aerothermodynamic and mechanical couplings in the engine and duct. It has been the goal of Systems Control, Inc. (SCI), in close coordination with NASA Ames personnel, to determine experimental mathematical models of the AWJSTOLRA Spey engine dynamics by the application of system identification techniques to flight test data.

The AWSTOLRA is a modified de Havilland Buffalo C8-A aircraft using two Rolls Royce Spey engines [1]. The program is sponsored jointly by the National Aeronautics and Space Administration and the Department of Industry, Trade and Commerce of Canada.

Modifications to this C8-A were made to improve lift characteristics at low speed for the short landing requirement. A transverse duct is used to channel fan airflow to ejector nozzles on the opposite wing blowing over an augmentor flap. Direct lift is provided by vectoring core stream flow.
through an adjustable nozzle. The modifications have a significant effect on the dynamical properties of the propulsion system and lead to the consideration of experimental model development for this aircraft.

The Spey engine is a low bypass, twin spool turbofan built by Rolls Royce of Canada. The normal gas path has been significantly modified for AWJSTOLRA operation. The new configuration is shown in Figure 1.1. The engine fuel control is hydromechanical with a mechanical cable linkage used to command throttle lever inputs from the cockpit electromechanical servo.

The linkage/control/engine system is nonlinear in both static and dynamic operation. Simple models previously utilized to describe engine behavior were not sufficient to explain observed response to closed-loop throttle inputs during landing. The SCI (Vt) program was undertaken to utilize system identification software to develop flight verified models of propulsion system dynamics.

1.2 SUMMARY

The report contains six chapters describing the development and validation of the Spey engine model. Chapter II describes the analysis of the dynamical interactions involved in the propulsion unit and the reduction of the model to contain only significant effects. The reduced model was used in conjunction with the flight data to develop initial estimates of parameters in the system using classical approaches. This analysis is presented in Chapter III. Chapter IV describes the theoretical background used in estimating the parameters and the software package developed to accomplish the processing. Flight data was processed to estimate and validate a set of parameters and these results are presented in Chapter V. The work and results are summarized in Chapter VI.
Figure 1.1 Gas Path Configuration of Spey Engines on AWJSTOLRA
CHAPTER II
MODELING THE SPEY ENGINE

2.1 INTRODUCTION

A simulation of the AWJSTOLRA was required to assess its handling and control characteristics prior to flight test [1,2]. In addition, pilots were to receive training in various flight regimes including system failures and abnormal operation. For this reason, a detailed digital simulation of the AWJSTOLRA aerodynamic and kinematics was developed by NASA Ames, including a model of the characteristics of the propulsion system. In this chapter, models for the propulsion system of varying levels of complexity are examined and a model is proposed for system identification.

2.2 TYPES OF ENGINE MODELS

The propulsion system can be described in mathematical terms in several ways. The choice of model form has an important impact on the analysis tools, the type of data taken, and the regimes of application of the expressions. Another consideration in the model is the interface between the engine and the input/output boundaries. Models that describe the throttle linkage and fuel control as well as the engine may be significantly different from "open-loop" models of the engine. Also, the estimation capabilities from experimental data will be directly influenced by the model interface. This partition of the physical system into model and "outside" is determined by the objectives of the descriptions.

In the Spey engine, a model is desired which describes the engine thrust and airflow response to power lever inputs from the cockpit. Dynamics in the engine or control which are not
excited during this operation are not important to the modeling task. The model will be used to describe large power excursions approximately and model dynamics in the high power region accurately. Altitude/speed regions of validity will be limited to low altitude/low speed and nonstandard temperature conditions.

The value of the model will be in its description of important aspects of performance. Steady-state values of thrust and airflow should match the values measured in static tests corrected to nonstandard conditions. Dynamic response to small perturbations should accurately reflect engine behavior. Match of intermediate variables describing engine operation is not critical.

Models used to describe typical engine behavior vary widely in complexity and accuracy [3]. The simplest description of engine response is to plot corrected engine thrust versus power lever angle. This characteristic is shown in Figure 2.1. The plot can be used as a dynamic model if an appropriate time lag is associated between the actual throttle position and a lagged or virtual position. The engine is observed to accelerate and decelerate at rates dependent on the power level. This effect can be represented in the model as a variable rate limit. The experimental model is shown in Figure 2.2. No attempt is made at a phenomenological explanation of the behavior and parameters are adjusted from observations. Since the model does not reflect the internal derivative agents of the thrust response, the match between the system behavior for various size inputs and for different starting and ending values of thrust will be poor. A more significant drawback of these models is the poor closed-loop description of the behavior. This mismatch is typically manifested by an incorrect prediction of the closed-loop stability of the system. This is due primarily to matching the step response in estimating parameters rather than matching
Figure 2.1 Thrust Response to Throttle Input

Figure 2.2 Experimental Model of Engine Dynamics
frequency response over a suitable bandwidth. Generally, frequency matching requires more complex model forms. In order to accommodate this lack of precision, simple models tend to be "improved" by ad hoc additions which attempt to correct the fundamental nonphenomenological characteristics of the formulation.

Far more complex digital simulations exist which can model component characteristics measured from rig tests and aerothermodynamic phenomena occurring in the gas path. Such a simulation is shown schematically in Figure 2.3. These simulations include basic physical laws relating energy, work, massflow and acceleration as they dynamically interact in the engine. Various "adjustable" parameters such as lumped isentropic efficiencies and areas are adjusted to match the observed steady-state relationships between input and output variables. A detailed representation of the governor is included and the overall response can be generally tuned to match observed dynamics. While this type of simulation recreates steady-state performance accurately, the match of transient response is poorer due to modeling uncertainties in the complex equations. Predicted stability characteristics in a closed-loop control are quite good. Thus, these programs can be used as a durable test bed for control and autothrottle design and evaluation.

The drawbacks of detailed analytical simulations are that: (a) a large computer capability is necessary to execute them (often slower than real time), (b) detailed internal engine and control information is required which is only available (if at all) to the engine manufacturer, and (c) the added complexity cannot be justified or validated from input/output performance observed in operation. Clearly, a middle ground must exist.

The development of an experimentally validated analytical description of the propulsion system can be accomplished by
Figure 2.3 Typical Detailed Nonlinear Simulation Computational Flowpath
considering the available measurement data and the type of inputs provided. The development starts with a complete analysis of possible dynamic effects. Data is examined and unimportant or unexcited phenomena are removed. The remaining model terms are examined for bandwidth of response relative to the simulation requirement. High frequency effects can be neglected. The resulting system represents a description of the response including both steady-state and dynamic performance whose parameterization is available from recorded measurements. This model will provide an excellent component module in an overall aircraft simulation used as a preliminary tool in the design of outer loop integrated control functions such as flight direction, autothrottle and SAS.

2.3 CURRENT SPEY ENGINE MODEL IN THE AWJSTOLRA SIMULATION

The engine model used on the AWJSTOLRA simulator is shown in Figure 2.4 (a detailed listing is included in Appendix B). The throttle setting at the input to engine control, $\delta_{TH}$, differs from the commanded value by a hysteresis function (not shown) which models the mechanical backlash. Two forward paths model accelerations and decelerations. This form is used because the engine control accelerates the turbine using a different control law than the one used during decelerations as is discussed in Section 2.4. The forward paths modulate variable rate limits for the first order lag system which produces the lagged or virtual throttle position, $\delta_{VTH}$, as the output. The acceleration rate limit is low at low power and increases at higher power levels. This increased rate limit is lagged by the fuel metering time constant, $\tau_1$, which is discussed in Section 2.4. The deceleration path has a small signal time constant which is 60% of the acceleration time constant. This represents a parameter used to match observed full throttle transients. It is not internal to the engine, since for small inputs, the engine acceleration and deceleration
Figure 2.4 Current Spey Engine Model in AWJSTOLR Aircraft Simulation
time constants are the same. Stated another way, the engine can be modeled as a linear dynamic system for reasonably small inputs. The deceleration rate limit decreases at lower powers. Figures 2.5 and 2.6 show typical sea level static acceleration/deceleration time responses. Note that the form of the model in Figure 2.4 is dictated by the response rather than fundamental system phenomenology describing the behavior.

It can be observed from the model that, at power levels above about 65% thrust, the engine is operating away from the rate limits for most inputs. Removing these blocks from the figure produces a simpler view of the equivalent high power representation of the system. This is shown in Figure 2.7. The system is quite simply represented, but is fundamentally nonlinear since the loop gain and time constant depend on the sign of the power transition.

The model in Figure 2.7 uses steady-state representations to calculate the mass flow and thrust. This is equivalent to the assumption that the engine maintains its steady-state rotor speed match during nonsteady operation. This assumption is not correct. Cold and hot thrust are determined by the fan airflow and core airflow which have independent degrees of freedom.

In summary, the possible problems in utilizing a model of the form of Figures 2.4 and 2.7 are as follows:

(1) A nonphenomenological representation of behavior leads to false conclusions such as a fundamental nonlinearity of response.

(2) A small number of degrees of freedom produce poor transient matches.

(3) Matches of engine response between arbitrary endpoints are uncertain and the number of parameters in the model is insufficient for adjustment.
Figure 2.5 Engine Test Stand Acceleration Characteristics
Figure 2.6 Engine Test Stand and Flight Test Decelerations to Idle
Figure 2.7 Simplified High Power Model
The Spey engine is characterized by a group of nonlinear dynamical elements: (a) the engine, (b) the fuel control, and (c) the throttle linkage. Models representing each of these elements can be derived for the individual physical members in the system. Simple analysis of each component produces approximate parameter values. The model can be simplified for identification to contain only those elements which can be estimated from the flight data. The development of the detailed engine model will be described in this section.

The dynamics of the engine in the region near full power are nearly linear. Small perturbations in the state of the engine are characterized by the two rotor speeds. Previous analysis of turbofan engine dynamics has verified the linearity and the structure of the linear system representing this behavior [3,4].

Since the engine has no variable geometry in the gas path, the primary dynamic exciter is the fuel flow which is metered by the hydromechanical governor. Fuel is chemically converted to heat in the combustor. This provides excess energy at the turbine entrance. The combustor lag is typically on the order of 0.5 to 2.0 msec and is therefore much faster than the dominant system response. The gas temperature at the entrance to the turbines is converted into work by the expansion through the rotating turbines. This energy is used to drive the compressor on the high rotor shaft and the fan on the low rotor shaft. The dominant engine dynamical states can be associated with the rotating inertia of these elements. Newton's law for the shafts can be written in terms of the speeds and torques as follows:
\[ \dot{N}_H = \frac{1}{T_H} \left( Q_{HTURB} - Q_{COMP} \right) \]  
\[ \dot{N}_L = \frac{1}{T_L} \left( Q_{LTURB} - Q_{FAN} \right) \]

where \( (\quad)_H, (\quad)_L \) refer to the high pressure and low pressure component shafts. The torques are determined from the mass flow and pressure change across the component. The form of the expression is given below (for a compressor)

\[ Q = \frac{T_{IN} \dot{m}(P_R\gamma - 1/\gamma - 1)}{\eta C_N} \]

where the compression ratio, \( P_R \), is determined by flow equilibrium through the engine and duct.

These equations can be used as the basis of a global nonlinear model of the engine. Nonlinear maps and detailed dynamical representation of nonequilibrium flow can model the continuity, energy and mixing phenomena occurring in the gas path. These terms can be considered negligible if one is interested in system bandwidths of flight control, engine and autothrottle function.

For modeling the engine behavior during moderate speed excursions, a linear set of dynamical equations provides an accurate representation. The form of the system can also be simplified to second order for most turbofans by considering the physical character of the motion. For perturbation of fuel flow with bandlimited content, the two engine spools will accelerate in a collective fashion. For perturbation in fan loading due to variations in duct pressure or inlet distortion, the fan speed will tend to "relax" toward equilibrium independently of the compressor. Using these observations, or equivalently calculating the relative magnitudes of the component map slope terms [5] such as

\[ \frac{\partial Q}{\partial N} \]

the qualitative analysis can be verified.
The linear system of constant-coefficient equations which is constructed for the engine is shown in a Laplace transform block diagram in Figure 2.8 and collected in state variable form as follows:

\[
\begin{bmatrix}
\dot{\delta N_L} \\
\dot{\delta N_H}
\end{bmatrix} =
\begin{bmatrix}
-1/\tau_L & C_{HL}(1-\alpha) \\
0 & -1/\tau_H
\end{bmatrix}
\begin{bmatrix}
\delta N_L \\
\delta N_H
\end{bmatrix} +
\begin{bmatrix}
\alpha C_{FH}C_{HL} \\
C_{FH}
\end{bmatrix}
\begin{bmatrix}
\delta W_f
\end{bmatrix}
\]

(2.4)

where the five linear parameters, \(\tau_L\), \(\tau_H\), \(C_{HL}\), \(C_{FH}\), and \(\alpha\) represent the dynamical behavior of the engine.

The dynamical equations are written for perturbations away from an equilibrium condition. A typical choice for this equilibrium point is 100% power.

In the Spey configuration on the AWJSTOLR aircraft, the ducting arrangement can cause the engine dynamics to be more uncoupled than the normal mixed flow path configuration. The fan flow is completely ducted to the active lift devices on
the wings. Back pressure effects on this flow will greatly affect fan response characteristics. The core stream exhaust is modified by a colander plate at the low pressure turbine exit. This plate provides choked exit conditions at all flight/power points and nozzle attitudes. The effect of nozzle motion on the flow is greatly reduced by this device.

The flow equations modeling the pressure phenomena can be written from the continuity equations as follows:

\[ \dot{\rho} = \frac{\gamma RT}{V} \Delta \dot{m} \quad (2.5) \]

where \( \Delta \dot{m} \) is the difference between inflow and outflow rates. This equation represents a flow lag effect and for small changes in power, the equation for the pressures in the duct and nozzle can be written:

\[
\begin{align*}
\delta \dot{p}_D &= -\frac{1}{\tau_D} (\delta p_D - K_{DL} \delta N_L) \\
\delta \dot{p}_E &= -\frac{1}{\tau_E} (\delta p_E - K_{EH} \delta N_H)
\end{align*} \quad (2.6)
\]

where the temperature dependence is ignored and the mass flows are modeled as linear functions of the perturbational rotor speeds which feed the volumes. Unlike the normal turbofan configuration, there is no flow coupling back through the duct to the fan from the nozzle volume. It is expected that the flow time constants will be small compared to the rotor lags, but because of the usual configuration, they will be retained in the model.

For power levels typical of the approach configuration, the Spey engines themselves do not represent the largest source of nonlinearity. The throttle linkage and hydromechanical fuel control represent the limiting elements in the linearity of the response. The components are quite complex.
and accurate representations of the internal dynamics are difficult to reconstruct from flight data.

Engine power requests from the cockpit are transferred by a mechanical linkage driven by an electromechanical force servo. The cable moves the throttle arm at the engine. About a 15° rotation of the linkage represents the full idle-to-maximum forward power excursion. The system contains a moderate amount of hysteresis which affects the small inputs typical of the autothrottle landing commands. Figure 2.9 shows the representation of the throttle linkage and the nonlinear equations are written below. This calculation is performed at each update time in the simulation.

\[ \delta_T(n) = \begin{cases} \delta_C(n) - h & \delta_C(n) > \delta_T(n) + h \\ \delta_T(n) & \delta_T(n) - h \leq \delta_C(n) \leq \delta_T(n) + h \\ \delta_C(n) + h & \delta_T(n) - h > \delta_C(n) \end{cases} \]

The hydromechanical fuel control is the most complex nonlinear element of the system. Figure 2.10 shows the speed control and high pressure (H.P.) fuel pump. These systems affect the dominant response of the governor. A description of the elements of governor operation is included to put the detailed analysis into perspective.

The fuel control meters fuel to engine in response to throttle inputs and engine demand. Instability, temperature, speed, and airflow constraints are accommodated while producing specified performance at different temperatures, altitudes and speeds.
Fuel is supplied from the tank via a low pressure pump, filter, and flow meter to the high pressure (H.P.) pump entrance. Fuel is metered through the control to the fuel spray nozzles in the combustor. The pressure drop across the fuel spray nozzles in the combustor, The pressure drop across the governor controls the fuel rate and this quantity is the primary actuated variable. Airflow limits, fan overspeed, temperature limits and emergency cutoff functions are performed by elements which are peripheral to the governor and not significant in the approach regime at near sea level conditions (the engine is not temperature limited at this condition)

The dynamical elements of the governor consist of the bellows assembly, speed governor, pressure drop governor and H.P. pump. Each element will be discussed and the appropriate model given. Figure 2.10 shows a schematic diagram of the pump and fuel control.

The main element of the fuel system is the H.P. pump. This variable displacement device is constrained to maintain the pressure drop across the regulator by a servo piston feedback element on the pump stroke. The system is shown in Figure 2.11. The linearized equation for this system is as follows:

\[ \dot{W}_f = \frac{1}{b} [A(P_U - P_D) - k(W_f - W_{f_0})] \quad (2.9) \]

Figure 2.11 Fuel Pump Model

where the \( P_U - P_D \) is the pressure drop across the governor, and \( b, k, \) and \( W_{f_0} \) describe damping and spring restraints on the piston. Fuel is metered by a pressure drop governor through a triangular orifice. The pressure drop across an orifice in a liquid flow is proportional to the square of the flow rate. By making the pressure drop proportional to the squares of the speed, a linear control governor action is achieved. A
Figure 2.10 Engine Fuel Control and Pump
Watt's governor can be used for this purpose with the rotating masses suspended in the fuel to provide damping and fuel density compensation. The details of the mechanism are complex and a simplified diagram is used for the analysis.

The triangular orifice is drawn in Figure 2.12. The metering area is determined by two displacements, \( X_1 \) and \( X_2 \). The \( X_1 \) displacement measures motion of the throttle sleeve controlled by the governor and throttle springs. The position of the orifice is adjusted by compression ratio and altitude by the capsule assembly discussed below. The metering area can be written as follows:

\[
A_m = C_m (X_1)(X_1 + X_2)
\]  

(2.10)

Considering only motion of the throttle sleeve,

\[
\dot{X}_1 = \frac{1}{b}(f_T - f_G)
\]  

(2.11)

where \( f_T \) is determined by the spring action of the throttle linkage and \( f_G \) is the restoring force due to rotation of the counterweights. These may be written as follows:

\[
f_T = a_1 + k_T (X_C - X_1)
\]  

(2.12)

\[
f_G = k_G N^2
\]  

(2.13)

where \( X_C \) is the commanded sleeve position derived from the throttle inputs.
The pressure drop across the metering orifice is sensed by the pressure drop piston in the control. This piston moves an orifice in conjunction with a second set of fly weights. The orifice configuration is shown in Figure 2.13.

![Figure 2.13 Pressure Drop Governor Orifice](image)

Figure 2.13 Pressure Drop Governor Orifice

and the equations of motion for this system can be written as follows:

\[ \dot{x}_3 = (f_{PD} + p_D - p_U) \]

(2.14)

and the total orifice area is given by the following

\[ A_{PD} = A_0 - C_{PD} x_3^2 \]

(2.15)

and

\[ f_p = C_{PD} N^2 \]

(2.16)

The function of the governor is to provide stable, isochronous regulation at a wide range of operating conditions. Primary orifice size is limited because of dynamic range considerations. The operation can be understood by considering the static behavior.

In steady state, the pressure drop governor has the following characteristics.
The speed governor, which regulates the metering area, $A_m$, can be approximated by the following expression for small motion of $X_1$:

$$A_m = \sqrt{K_p \left( b_1 + b_2 \cdot b_3 \cdot b_4 \cdot N^2 \right)}$$

and the overall control law can be modeled as follows:

$$W_f = K_p \left( b_1 + b_2 \cdot b_3 \cdot N^2 \right) N$$

Note that for fixed metering error (Eq. (2.19), $A_m = 0$) the control law is unstable, i.e. increased speed increases the fuel. This occurs during large transients when the governor area is limited by the throttle stops. Figure 2.14 shows a typical large transient in the control plane. For a step input, fuel flow increases at the pump response rate to the constant area line $A_2$. Fuel and speed increase along this line until the required speed is reached by the selected throttle curve, $\delta_B$. The throttle sleeve moves off the stop and the cubic speed relationship dominates the control law. The system is stable as shown. Accelerations follow the reverse logic. This method provides fast and stable fuel regulation for the Spey except at low power. Altitude variations are not compensated by the components. Note that antisymmetric rate limits will cause the engine to accelerate and decelerate at different rates for large, throttle limited power excursions.

The capsule assembly and $p_3$ limiter provide altitude compensation and low power acceleration limiting for the engine. A model of this system which affects the orifice sleeve position, $X_2$, and the metering area, is not described in detail. The
effect of these components is most pronounced during accelerations from low power. At these conditions, the fuel flow increase is slowed considerably until the rotor speeds are increased. This limits the possibility of surge during acceleration. Hysteresis can be important during small motions of the bellows and this might affect the overall repeatability of the response.

The various component models have been interconnected in Figure 2.15. The dynamic range of many of these elements is higher than the dominant response time constants. Thus, while the dynamical relationships can be ignored, the static feedback structure of the governor will significantly affect engine response times and linearities.
Figure 2.15 Detailed Dynamics Simulation of Spey Fuel Control
2.5 SIMPLIFIED MODEL

The system model can be simplified to include the linear model of the engine; the hysteresis model of the throttle linkage, the variable metering orifice characteristic expanded to second order and the pressure drop regulator characteristic. This system is shown in Figure 2.16.

The time constants of the electromechanical servo are between 0.05 and 0.1 seconds which is faster than the expected closed-loop engine response. Since commanded throttle and actual deflections are recorded, any relative time delay can be detected. The fuel pump response time should be faster than 0.1 seconds at high displacements. This response time is the limiting element in the overall governor response loop since fly weight motion should have response times less than 25 msec. The engine control reduces to a nonlinear feedback regulator driving a rate servo modeling the action of the pressure drop governor. Any mismatch between the forward and feedback path gains in the pressure drop system will result in a slightly stable or unstable servo loop. This gain mismatch can be lumped with the speed governor characteristic without loss of generality. The forward and feedback gains in this loop are assumed identical. The closed-loop characteristic of the pressure drop governor is a pure integration (i.e. a rate servo on the speed governor output). The nonlinearity of the metering orifice is reflected by a slightly nonsymmetric transfer characteristic. The overall hysteresis of the control after the throttle is represented by a second hysteresis block.

A further simplification of the control model is possible assuming that the transfer nonlinearity and control hysteresis are small. This is discussed further in Chapter V.
Figure 2.16  Proposed Spey Engine Identification Model
2.6 SENSOR MODELS

Data on Spey performance was acquired in flight. Instrumentation was used as listed in Table 2.1. Speed, position, and pressure transducers were assumed to respond faster than the dynamics being estimated. The measurement errors associated with these inputs can be considered uncorrelated random processes with RMS values shown in the table.

Table 2.1
Sensor List

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<th>VARIABLE</th>
<th>SYMBOL</th>
<th>RMS ERROR</th>
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<tr>
<td>Low Rotor Speed</td>
<td>$N_L$</td>
<td>0.5%</td>
</tr>
<tr>
<td>High Rotor Speed</td>
<td>$N_H$</td>
<td>0.5%</td>
</tr>
<tr>
<td>Throttle Command</td>
<td>$\delta_c$</td>
<td>0.2 Degrees</td>
</tr>
<tr>
<td>Throttle Angle</td>
<td>$\delta_T$</td>
<td>1.0%</td>
</tr>
<tr>
<td>Duct Pressure</td>
<td>$P_D$</td>
<td>2.0%</td>
</tr>
<tr>
<td>Exhaust Pressure</td>
<td>$P_E$</td>
<td>2.0%</td>
</tr>
<tr>
<td>Fuel Flow</td>
<td>$W_{fm}$</td>
<td>2.0%</td>
</tr>
</tbody>
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The fuel flow was measured at the exit of the low pressure fuel pump. The sensor is a restrained flow turbine whose angular deflection is proportional to the turbine torque and mass flow.

The system response is lightly damped because of the small value of flow damping supplied by the turbine. The model for the flowmeter is shown in Figure 2.17.
The values $\zeta = 0.2$ and $\omega_n = 13.0 \text{ sec}^{-1}$ were estimated from the flight data.

2.7 SUMMARY

A group of mathematical models has been developed for the dynamical behavior of the Spey engine. Simple non-phenomenological forms are used to match specific transients. Detailed representations of the intercomponent responses can be used to accurately predict the system response if dimensional and rig data is available. A simplified version of the detailed model is derived which contains important linear and nonlinear elements of the engine, control and throttle response for the landing approach (high power) configuration of the Spey.
CHAPTER III
PRELIMINARY PARAMETER ESTIMATES

The AWSTOLR aircraft was flown at Crow's Landing, California to acquire engine response data for parameter estimation. The data was initially evaluated to determine initial parameter estimates in the dynamic model developed in Chapter II. Important terms were identified here and data consistency was validated. A nonlinear system identification routine was incorporated with a digital simulation of the engine. This program is described in Chapter IV. The results of the processing of the flight and the parameter estimates are compiled in Chapter V.

3.1 SUMMARY OF FLIGHT DATA

Flight data was taken in the approach configuration in a powered descent from 7500 ft. The nozzles in this configuration are rotated downward and the engines are operating near full power. The flight computer was used to generate throttle rate commands to the electromechanical servo. The engine throttles were modulated in unison. Appendix B contains a description of the flight log for the two data acquisition runs. Flight time for the two runs was 124 minutes from takeoff to final landing. Data was recorded during this time on 84 instrumentation channels comprising engine, aircraft and avionics equipment sampled every 20 msec. Data records were chosen from the complete flight data for the parameter estimation. These data records are listed in Table 3.1. A set of instrumentation inputs was chosen to reflect the variables incorporated in the model. These measurements are listed in Table 3.2.
<table>
<thead>
<tr>
<th>RECORD</th>
<th>START TIME (d-h:m:s:ms)</th>
<th>END TIME (d-h:m:s:ms)</th>
<th>DURATION (SEC)</th>
<th>NO. OF POINTS</th>
<th>FLIGHT NO.</th>
<th>INPUT DESCRIPTION</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>295-17:59:30</td>
<td>295-18:02:31</td>
<td>181</td>
<td>9050</td>
<td>323</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>RAMP RATE / PERIOD / AMPLITUDE (DEG/SEC) / (SEC) / (DEG)</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>0.5 / 5 / 1.25</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>1.0 / 5 / 2.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>2.0 / 5 / 5.0</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>3.0 / 5 / 7.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>0.5 / 10 / 2.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>1.5 / 10 / 7.5</td>
</tr>
<tr>
<td>4</td>
<td>295-20:21:01</td>
<td>295-20:22:05</td>
<td>64</td>
<td>3200</td>
<td>324</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>1.5 / 5 / 3.75</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>2.0 / 5 / 5.0</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>0.25 / 50 / 6.25</td>
</tr>
<tr>
<td>6</td>
<td>295:20:24:00</td>
<td>295:20:25</td>
<td>60</td>
<td>3000</td>
<td>324</td>
<td>Step Inputs</td>
</tr>
</tbody>
</table>
Table 3.2
Sensor List

<table>
<thead>
<tr>
<th>SENSOR</th>
<th>ENGINE</th>
<th>UNITS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Throttle Command</td>
<td>Left</td>
<td>Deg</td>
</tr>
<tr>
<td>Throttle Pickoff</td>
<td>Left</td>
<td>Counts</td>
</tr>
<tr>
<td>Fuel Flow</td>
<td>Left</td>
<td>PPH</td>
</tr>
<tr>
<td>Compressor Speed</td>
<td>Left</td>
<td>RPM</td>
</tr>
<tr>
<td>Fan Speed</td>
<td>Left</td>
<td>RPM</td>
</tr>
<tr>
<td>Exhaust Pressure</td>
<td>Left</td>
<td>Counts</td>
</tr>
<tr>
<td>Duct Pressure</td>
<td>Left</td>
<td>Counts</td>
</tr>
<tr>
<td>Throttle Command</td>
<td>Right</td>
<td>Deg</td>
</tr>
<tr>
<td>Throttle Pickoff</td>
<td>Right</td>
<td>Counts</td>
</tr>
<tr>
<td>Fuel Flow</td>
<td>Right</td>
<td>PPH</td>
</tr>
<tr>
<td>Compressor Speed</td>
<td>Right</td>
<td>RPM</td>
</tr>
<tr>
<td>Fan Speed</td>
<td>Right</td>
<td>RPM</td>
</tr>
<tr>
<td>Exhaust Pressure</td>
<td>Right</td>
<td>Counts</td>
</tr>
<tr>
<td>Duct Pressure</td>
<td>Right</td>
<td>Counts</td>
</tr>
</tbody>
</table>
The flight data was recorded on an in-flight data acquisition unit and the digital tape processed to provide scaling, units and conversion for use on the CDC 6600 at NASA Ames Research Center.

3.2 INITIAL PARAMETER ESTIMATES

Classical techniques were used to calculate initial parameter estimates using plots obtained from the data. Four areas were investigated using left engine data: (a) steady state characteristics, (b) hysteresis, (c) rate limiting and symmetry, and (d) fan/compressor dynamics. These are discussed in the subsections below.

3.2.1 Steady State Operation

An accurate model of Spey operation must match engine static behavior in this configuration. The engine model described in Chapter II provides a linear operating line for the state and control variables. Figure 3.1 is a compilation of measured engine response at steady power points. It will be noted that the approximation of linear spool speed to throttle response above about 5 degrees throttle is quite good. However, thrust and airflow are nonlinear functions of throttle. Since these are the primary model outputs, a tabular relationship can be developed for these variables as a function of rotor speed or throttle. Table 3.3 shows these functions tabulated against compressor speed. The cold and hot thrust figures refer to the thrust developed by the exhaust flow and duct flow of the engine at an assumed equilibrium rotor speed match. During transients, rotor speeds are not matched and an alternate formulation of these curves must be developed.

An approximate set of curves for the thrust and airflow variables can be developed if it is assumed that fan airflow and core airflow are dependent only on the rotational speeds
Figure 3.1 Test Stand Data for Spey Engines
Table 3.3
Spey Engine Outputs as a Function of Steady State Corrected Engine Speed

**CORRECTED HOT STREAM THRUST ~ T_n/δ (lbs)***

<table>
<thead>
<tr>
<th>% SPEED ( N_L/\sqrt{\theta} )</th>
<th>% SPEED ( N_H/\sqrt{\theta} )</th>
<th>INDICATED AIRSPEED - KNOTS</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>( V = 0 )</td>
</tr>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>55.0</td>
<td>84</td>
<td>1023.2</td>
</tr>
<tr>
<td>70.5</td>
<td>89.5</td>
<td>2402.7</td>
</tr>
<tr>
<td>78.9</td>
<td>92.5</td>
<td>3714.2</td>
</tr>
<tr>
<td>86.0</td>
<td>95</td>
<td>4834.6</td>
</tr>
<tr>
<td>94.4</td>
<td>98</td>
<td>5867.0</td>
</tr>
<tr>
<td>109.9</td>
<td>103.515</td>
<td>7013.5</td>
</tr>
</tbody>
</table>

**CORRECTED COLD STREAM THRUST ~ T_c/\delta (lbs)***

<table>
<thead>
<tr>
<th>% SPEED ( N_L/\sqrt{\theta} )</th>
<th>% SPEED ( N_H/\sqrt{\theta} )</th>
<th>INDICATED AIRSPEED - KNOTS</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>( V = 0 )</td>
</tr>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>55.0</td>
<td>84</td>
<td>796.7</td>
</tr>
<tr>
<td>70.5</td>
<td>89.5</td>
<td>1660.2</td>
</tr>
<tr>
<td>78.9</td>
<td>92.5</td>
<td>2494.5</td>
</tr>
<tr>
<td>86.0</td>
<td>95</td>
<td>2977.9</td>
</tr>
<tr>
<td>94.4</td>
<td>98</td>
<td>3301.0</td>
</tr>
<tr>
<td>109.9</td>
<td>103.515</td>
<td>3532.3</td>
</tr>
</tbody>
</table>
Table 3.3 (Concluded)

<table>
<thead>
<tr>
<th>% SPEED $N_L/\sqrt{\theta}$</th>
<th>% SPEED $N_H/\sqrt{\theta}$</th>
<th>CORRECTED ENGINE AIRFLOW $m_a/\sqrt{\theta}$ (Slugs/Sec.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>55.0</td>
<td>84.</td>
<td>2.514</td>
</tr>
<tr>
<td>70.5</td>
<td>89.5</td>
<td>4.003</td>
</tr>
<tr>
<td>78.9</td>
<td>92.5</td>
<td>4.964</td>
</tr>
<tr>
<td>86.0</td>
<td>95.</td>
<td>5.507</td>
</tr>
<tr>
<td>94.4</td>
<td>98.</td>
<td>5.935</td>
</tr>
<tr>
<td>109.9</td>
<td>103.515</td>
<td>6.488</td>
</tr>
</tbody>
</table>
of the compression elements. This is normally a more accurate assumption than the constant by-pass ratio assumed in Table 3.3. In this case, hot thrust is a function of the compressor speed only. Cold thrust and net airflow are determined by the fan speed. During transient motion each speed can vary independently.

Data is required to determine the equilibrium rotor speed match. This data was measured from the flight tape when the engine had settled at a constant power. Figure 3.2 shows this match and verifies the assumed linearity of the match in this power range. Figure 3.2 and Table 3.3 can be used to determine a new functional representation of cold thrust and mass flow plotted against fan speed. Then, during simulation, nonequilibrium fan speeds are used to calculate instantaneous cold thrust and mass flow for the engines.

Similar steady operating line data can be derived from the flight records for the intermediate simulation variables vis. fuel flow, throttle angle, and pressures as a function of the states of the simulation. Relationships derived from the data are shown in Figure 3.3 for fuel flow and throttle angle as a function of rotor speed. This line should be compared to Figure 2.14 which is the predicted linear relationship based on the governor characteristics. The operating assumptions of the linear operating linear hold quite well for this particular flight regime. Steady state linear operating equations are listed below:

\[
0 = 0.7212 \left( \delta_T - 31.5\right) - (N_H - 100%)
\]
\[
N_L = 2.81 N_H - 181%
\]
\[
W_f = 7.13 N_H - 613%
\]
\[
PTD = 1.32 N_L - 32%
\]
\[
PT6 = 4.0 N_H - 300%
\]

where the 100% values of the variables are listed in Table 3.4.
Figure 3.2 Rotor Speed Match (AWJSTOLRA Left Engine) Flight 323/324

\[ N_L = 2.81 \frac{N_H}{100} - 181 \]
Figure 3.3 Fuel Flow Steady State Data
(AWJSTOLRA Left Engine)
Flight 323/324

* 1% = 50 pph
Table 3.4
100% Values of Engine Variables

<table>
<thead>
<tr>
<th>VARIABLE</th>
<th>VALUE</th>
<th>UNITS</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_T$</td>
<td>31.5</td>
<td>Degrees</td>
</tr>
<tr>
<td>$\delta_{po}$</td>
<td>-957</td>
<td>Counts</td>
</tr>
<tr>
<td>$W_f$</td>
<td>5000</td>
<td>pph</td>
</tr>
<tr>
<td>$N_H$</td>
<td>12135</td>
<td>RPM</td>
</tr>
<tr>
<td>$N_L$</td>
<td>10075</td>
<td>RPM</td>
</tr>
<tr>
<td>PT6</td>
<td>3775</td>
<td>Counts</td>
</tr>
<tr>
<td>PTD</td>
<td>5036</td>
<td>Counts</td>
</tr>
</tbody>
</table>

It should be noted that the operating line linearity of engine variables, e.g. the rotor speed match is predicted from previous engine experience near full power. Governor linearity is predicted from the functional characteristics of the centrifugal counter-weight forces and the square root pressure drop - mass relationship used in the metering orifice. The latter relationship can be used to derive other characteristics of this control response as is discussed in Section 3.2.5.

3.2.2 Throttle Hysteresis

Classical methods can be used to approximate the hysteresis width since data containing periodic excitation is available. The ramp inputs are sufficiently close to sinusoids that phase lag measurements can be used as a good measure of the hysteresis width. The sinusoidal describing function for hysteresis is shown in Figure 3.4. Considering
Figure 3.4 Hysteresis Describing Function Characteristic
only the phase shift characteristics, measured values can be related to the hysteresis width and input amplitude. The characteristic is independent of frequency which allows run-to-run comparison of results. The phase shift measurement allows a far more accurate estimate of the hysteresis width than amplitude roll-off when graphical data is plotted. An example of the phase shift measurement is shown in Figure 3.5. Hysteresis points were taken at several values of input magnitude and frequencies. These points are shown in Figure 3.6 with a least squares fit of the hysteresis characteristic:

\[ \phi = f_H(a/A) + \phi_b \]

where \( f_H \) is shown in Figure 3.4, \( \phi_b \) is a residual phase shift unexplained by hysteresis, and \( A \) is the amplitude of the periodic throttle inputs.

The throttle curve in Figure 3.6 shows the phase shifts, \( \phi_T \), between the commanded throttle input and the throttle pickoff. Since no frequency dependent lags are present, all data points can be overplotted. The fit shows the hysteresis width to be about 0.48 degrees with only 2° of phase shift unexplained by the hysteresis. This residual phase is within the accuracy bounds of the data so it can be concluded that a simple hysteresis is an excellent model of the throttle linkage.

Phase measurements were made between the high rotor speed response and both commanded and measured throttle inputs at a single frequency (\( \omega = 1.26 \text{ rad/sec} \)). The amplitude dependent portion of the response can be modeled as an additional hysteresis within the fuel control. The curve for the governor response indicates hysteresis width of about 0.19 degrees. This estimate may be less accurate due to nonlinearities in the governor response which will tend to cause amplitude dependent phase shifts.

The hysteresis figures calculated above can be used to correct the phase response of the high rotor speed to throttle.
Figure 3.5 Phase Calculation for Hysteresis Estimates
Figure 3.6 Identification of Hysteresis from Phase Lag
inputs. This is shown in Figure 3.7 for various frequencies. A fit of a first order lag phase response is shown which minimizes the sum squared error of the corrected points. The time lag associated with the fitted phase characteristic is 0.9 sec which should be close to the high rotor time constant (closed-loop).

3.2.3 Rate Limiting

The governor operation described in Chapter II indicates that throttle inputs act as rate commands to the pressure drop governor. Since the inputs during the flight test were throttle ramps, the rotor speed rates which are attained will reflect rate limiting by the governor if it exists. Rates were measured for various acceleration/deceleration magnitudes in the flight records. Rotor speed rates are plotted against throttle rates in Figure 3.8. The test run data was augmented by miscellaneous points which occurred when ramp inputs were generated as commands during the normal cockpit operation. Rates were measured for various acceleration/deceleration magnitudes in the flight records. Rotor speed rates are plotted against throttle rates in Figure 3.8. The test run data was augmented by miscellaneous points which occurred when ramp inputs were generated as commands during the normal cockpit operation. For a linear operating line, and no rate limiting by the control, the RPM response rate should be determined by the steady state operating line as is shown below:

\[(\text{Steady State}) (N_H - 100\%) = 0.7212 (\delta_T - 31.5)\]

\[(\text{Steady Rate}) \quad \dot{N}_H = 0.7212 \dot{\delta}_T\]

This line is drawn on the figure. The results show good correspondence to the predicted operation of the control. For throttle rates less than 5.5 deg/sec during accelerations or decelerations, the governor characteristic is linear. Rate limiting occurs at higher throttle input rates due to the saturation of the throttle sleeve on the stops. It appears that the acceleration and deceleration stops are not symmetrically adjusted so that large engine declerations are
Figure 3.7 Throttle to Rotor Speed Phase Difference (Data Corrected for Linkage Hysteresis) AWJSTOLRA Left Engine, Flight 323/324
Figure 3.8 Fuel Control Rate Characteristic (AWJSTOLRA Left Engine) Flight 323/324
somewhat slower. For maneuvers during test sequences, the governor does not rate saturate and this effect can not be precisely identified.

3.2.4 Fan and Compressor Dynamics

In the development of the dynamics of the engine in Chapter II, it was noted that the engine rotors have two modes of response, namely, a unison mode and a fan rotor relaxation mode. The magnitude of these time constants can be approximated from the flight data by examining the rotor phase relationship at various frequencies. This phase angle is not dependent on the characteristics of the fuel control. It is assumed that fuel flow primarily enters the compressor speed equation, then the transfer function between the rotor speeds can be written as follows:

\[
\frac{N_L}{N_H} = \frac{\tau_H s + 1}{\tau_L s + 1}
\]

where \( \tau_H \) and \( \tau_L \) are the time constants of the two modes. The phase differences were measured at three forcing frequencies and are plotted in Figure 3.9. A function representing the first order phase difference was fit to the data,

\[
\phi(N_L) - \phi(N_H) = \tan^{-1}(\tau_H \omega) - \tan^{-1}(\tau_L \omega).
\]

A time constant of 1.0 sec was determined for the compressor. This corresponds quite well with the fuel flow to rotor speed phase calculation corrected for hysteresis. The fan speed time constant is predicted to be 2.6 sec. This is a rather large value for typical turbofan configurations without the duct bleed. Since only three points were used, the accuracy figures are unknown and the estimate could be inaccurate.
Figure 3.9  Spool Dynamics Identification (from Fan and Compressor Phase Difference) AMJSTOLRA Left Engine - Flight 323/324
3.2.5 Control Parameters

The control model developed for identification has a number of static elements which can be estimated from steady state and dynamic responses of the controlled engine. These calculations are discussed below. The assumption is made that the null position in the speed governor is independent of engine speed. This is nearly true for moderate power changes. In this case, the governor error, $\varepsilon$, can be written from the steady state operating line as follows:

$$N - 0.7212 \delta_T - 77.28 = -\varepsilon \ (\% \ RPM)$$

The governor error is zero in steady state; thus, the pressure drop governor must cancel the engine dynamics to form a pure integrator (or rate servo). This implies that the engine fuel to rotor speed gain, $c_f$, is inversely balanced by the pressure drop regulator characteristic rotor speed to fuel flow (pressure drop) gain. This situation is represented in the model. The value of $c_f$ is determined from the operating line relationship,

$$W_f = 7.13N - 613\%$$

or

$$c_f = 7.13 \ (\% \ F.F./\% \ RPM)$$

The remaining control parameters are the governor forward loop gain and nonlinearity, $\alpha$. The control law for the engine and pressure drop regulator can be written as follows (neglecting the nonlinearity)

$$N_H = K_{AF} A_m \Delta$$

$$= \frac{C_{PF} K_{AF}}{T_H} (0.722 \delta_T - N_H - 77.28)$$
The time constant of the closed-loop system is \( \tau_{CL} = \tau_R/C_F K_{AF} \) and a reasonable guess of the value for \( K_{AF} \) is 1.0 sec\(^{-1}\). The nonlinearity, \( \alpha \), which results in slower accelerations than decelerations is difficult to estimate. A nominal value of zero is chosen for the initial estimation runs.

3.2.6 Summary of Initial Parameter Estimates

The discussion in Section 3.2 has provided the basis for initial parameter estimates for the engine/control model developed in Chapter II. These estimates are summarized in Table 3.5 and identified with the appropriate computer program symbology (see Chapter V for estimation results). Initial estimates for scale factor and instrumentation biases are also included. These were measured directly from the flight data.
Table 3.5
Initial Parameter Estimates

<table>
<thead>
<tr>
<th>PARAMETER</th>
<th>SYMBOL</th>
<th>INITIAL ESTIMATE</th>
<th>UNITS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Throttle Hysteresis</td>
<td>DLINK1</td>
<td>0.48</td>
<td>Deg</td>
</tr>
<tr>
<td>Control Hysteresis</td>
<td>DLINK2</td>
<td>0.19</td>
<td>Deg</td>
</tr>
<tr>
<td>Throttle Pickoff Scale Factor</td>
<td>BPOT1</td>
<td>-25</td>
<td>Deg/Counts</td>
</tr>
<tr>
<td>Throttle Pickoff Bias</td>
<td>BPOTØ</td>
<td>950</td>
<td>Counts</td>
</tr>
<tr>
<td>Throttle/RPM Gain</td>
<td>KTH</td>
<td>0.7212</td>
<td>Deg/% RPM</td>
</tr>
<tr>
<td>Area to Fuel Gain</td>
<td>KAF</td>
<td>1.0</td>
<td>Sec⁻¹</td>
</tr>
<tr>
<td>Governor Nonlinearity</td>
<td>ALPHA</td>
<td>0.0</td>
<td>(% RPM)⁻¹</td>
</tr>
<tr>
<td>Flowmeter Damping Ratio</td>
<td>ZETA</td>
<td>0.2</td>
<td>---</td>
</tr>
<tr>
<td>Flowmeter Natural Frequency</td>
<td>OMN</td>
<td>13</td>
<td>Sec⁻¹</td>
</tr>
<tr>
<td>Compressor Lag</td>
<td>CLAG</td>
<td>1.0</td>
<td>Sec</td>
</tr>
<tr>
<td>Fan Lag</td>
<td>FLAG</td>
<td>2.6</td>
<td>Sec</td>
</tr>
<tr>
<td>Fuel/RPM Gain</td>
<td>CF</td>
<td>7.13</td>
<td>% RPM/% PPH</td>
</tr>
<tr>
<td>Fuel to Fan Coupling</td>
<td>CFL</td>
<td>0.0</td>
<td>% RPM/% PPH</td>
</tr>
<tr>
<td>Fan Speed Bias</td>
<td>XNLBIS</td>
<td>0</td>
<td>% RPM</td>
</tr>
<tr>
<td>Tailpipe Pressure Lag</td>
<td>TLAG</td>
<td>0.0</td>
<td>Sec</td>
</tr>
<tr>
<td>Duct Pressure Lag</td>
<td>DLAG</td>
<td>0.0</td>
<td>Sec</td>
</tr>
<tr>
<td>Tailpipe Pressure Scale Factor</td>
<td>TGAIN</td>
<td>4.0</td>
<td>%/% RPM</td>
</tr>
<tr>
<td>Duct Pressure Scale Factor</td>
<td>DGAIN</td>
<td>1.32</td>
<td>%/% RPM</td>
</tr>
<tr>
<td>Tailpipe Pressure Bias</td>
<td>PT6BIS</td>
<td>-300</td>
<td>%</td>
</tr>
<tr>
<td>Duct Pressure Bias</td>
<td>PTDBIS</td>
<td>-32</td>
<td>%</td>
</tr>
</tbody>
</table>
A program was developed to process the flight acquired data and estimate parameters of the model. The details of the software are presented in this chapter. A program listing is included in Appendix A.

4.1 THEORETICAL BACKGROUND

The Spey engine model developed in Chapter II is used as the foundation of the parameter estimation procedure. This model is shown in Figure 4.1. The model equations can be represented as follows:

\[ \dot{x} = f(x, u, \theta) \quad (4.1) \]
\[ y_k = h(x_k, u_k, \theta) + \nu_k \quad (4.2) \]

where the \( x \) is the engine state vector, \( y \) is the measurement vector, \( f(\cdot, \cdot, \cdot) \) and \( h(\cdot, \cdot, \cdot) \) are the engine/control dynamic equations and measurement distribution matrix, respectively. Table 4.1 lists the element of the state, output, and control vectors. The measurement noise is assumed to have a Gaussian distribution with known covariance.

The maximum likelihood parameter estimates are obtained by minimizing the log-likelihood function for the parameter estimate given the data. A complete discussion of the theory is given in Ref. 6.

\[ J = \min_{\theta} \sum_{i=1}^{N} (y(i) - \hat{y}(i))^T R^{-1}(y(i) - \hat{y}(i)) \quad (4.3) \]

where \( R = \text{diag}(\sigma_1^2, \ldots, \sigma_p^2) \) is the covariance matrix of the independent measurement errors, and

\[ \hat{y}(i) = h(\hat{x}(i), u(i), \hat{\theta}) \quad (4.4) \]
Figure 4.1 Proposed Spey Engine Identification Model
Table 4.1
Engine State, Output and Control Vectors

<table>
<thead>
<tr>
<th>STATE</th>
<th>SYMBOL</th>
</tr>
</thead>
<tbody>
<tr>
<td>High Rotor Speed</td>
<td>XNH</td>
</tr>
<tr>
<td>Low Rotor Speed</td>
<td>XNL</td>
</tr>
<tr>
<td>Tailpipe Pressure</td>
<td>PT6</td>
</tr>
<tr>
<td>Duct Pressure</td>
<td>PTD</td>
</tr>
</tbody>
</table>

| CONTROL VECTOR               | DELTT   |
| Throttle Position            | DELTT   |

| MEASUREMENT VECTOR           |        |
| Throttle Pickoff             | DELTPO  |
| Fuel Flow                    | WFMSN   |
| High Rotor Speed             | XNHMSN  |
| Low Rotor Speed              | XNLMSN  |
| Tailpipe Pressure            | PT6MSN  |
| Duct Pressure                | PTDMSN  |

subject to the differential constraint
\[ \dot{x} = f(x, u, \theta). \]  (4.5)

For the Spey engine, Eq. (4.5) is discretized internally in the simulation program to form the discrete constraint equations:
\[ \dot{x}(i+1) = \phi x(i) + \Gamma u(i). \]  (4.6)

The minimization in Eq. (4.3) is carried out using either a Quasi-Newton method or Levenberg-Marquart procedure. The Levenberg Marquart algorithm provides a stable search for the minima near a priori estimates. The parameter step for this algorithm is as follows:
\[ \theta(i+1) = \theta(i) - (\frac{\partial J}{\partial \theta} + \lambda I)^{-1} \frac{\partial J}{\partial \theta} \]  (4.7)

59
where the expression,
\[
\frac{\partial^2 J}{\partial \theta^2},
\]
is approximated from gradient calculations and the Marquart parameter, \( \lambda \), is chosen based on minimization performance during the search. More detail on these algorithms is contained in Ref. 6.

The parameter estimation procedure described produces the maximum likelihood estimates for a given data record assuming the correct model structure and no process disturbances. These assumptions are well satisfied in the engines. The model structure has been carefully studied and initially evaluated to assure a good a priori structure.

An additional identification option is included in the program to facilitate the initial startup of the procedures. Convergence of the maximum likelihood procedure is slowed by poor initial estimates and by improperly excited modes. To improve initial parameter estimates before the full algorithm is run, a partial identification procedure can be used. The program logic allows utilization of measured variables as inputs to model blocks in the simulation. In this way, smaller identification problems can be run to obtain accurate a priori guesses for the full system. As an example, measured fan speed can be used as one input to the duct dynamics. Gain and time constants can be found which then are used as estimates for the full identification. For the full run, the derived fan speed is used as the input and the overall response is estimated.

4.2 STATISTICAL PROPERTIES

The identification algorithm generates an estimation of the statistics of the parameters. These figures may be used to
evaluate the parameter accuracies and combine estimates from
different data records to calculate the best overall para-
meter results. Matrix outputs from the program are discussed
below.

The cost function, \( J(\theta) \) is calculated from normalized
measurement error covariances as follows:

\[
J = \sum_{i=1}^{P} \frac{R_{0}}{R_{i}} \sum_{j=1}^{N} \varepsilon_{i}(j)^{2} \quad (4.8a)
\]

\[
\varepsilon_{i}(j) = y_{i}(j) - \hat{y}_{i}(j) \quad (4.8b)
\]

where \( R_{0} = \sigma^{2} \).

Thus, the measurement accuracies are weighted with respect to
the accuracy of the first instrument. The scaled information
matrix is derived from the second derivative of \( J \) with respect
to the parameters. This matrix is developed during the iter-
ative solution of the minimization problem. The scaled inform-
ation matrix is defined as follows:

\[
H = \frac{\partial^{2} J}{\partial \theta^{2}} \quad (4.9)
\]

The eigensystem (eigenvalues and eigenvectors) of \( H \) are
calculated. The overall conditioning of \( H \) and linear combina-
tions of parameters corresponding to small eigenvalues of \( H \)
are diagnostic tools in evaluating the convergence. An esti-
mate of the parameter accuracies can be determined from the
dispersion matrix as follows:

\[
D = N^{*} R_{0} H^{-1}
\]

and

\[
\text{cov}(\theta \theta^{T}) > D \quad (4.10)
\]

where the matrix \( D \) forms the Cramer-Rao bound on the esti-
mates if \( N^{*} = 1 \).
The value of $N^*$ is chosen to weight the independence of the data. A good approximation to this value is the ratio of the number of points between uncorrelated residuals, or, if the residual correlation time constant is approximately determined to be $\tau_{\text{COR}}$, then $N^*$ can be chosen as follows:

$$N^* = \frac{\tau_{\text{COR}}}{TN_{\text{TOT}}} \quad (4.11)$$

where $T$ is the sampling interval and $N_{\text{TOT}}$ is the total number of data points. Values of $N^*$ are typically chosen between 1 and 20.

The fractional variance is calculated from the unit dispersion matrix ($D$ for $N^* = 1$). This equation is

$$\hat{\sigma}_{\theta_i} = \sqrt{N^* D_{ii} / \hat{\theta}_i} \quad (4.12)$$

where $\sigma_{\theta_i}$ is the fractional confidence interval on the estimate of the parameter, $\theta_i$.

The $F$-ratio of the parameter estimate indicates the significance of the parameter to the model. The $F$-ratio is the inverse of the fractional variance, or,

$$F_{\theta_i} = 1/\sigma_{\theta_i}^2 \quad (4.13)$$

Finally, the fractional gradient of the likelihood function indicates the "closeness" of convergence of the algorithm. This quantity is defined as follows:

$$G_{\theta_i} = \partial J / \partial \theta / \theta \quad (4.14)$$

The quantities listed above are output for each parameter estimated. An example of this output is given in Figure 4.2.
### Scaled Information Matrix

<table>
<thead>
<tr>
<th>Parameter</th>
<th>ALO</th>
<th>AL1</th>
<th>BPOT</th>
<th>EPOT</th>
</tr>
</thead>
<tbody>
<tr>
<td>ALO</td>
<td>0.35749386701290E+07</td>
<td>0.60253017431526E+08</td>
<td>1.0627777076374E+10</td>
<td></td>
</tr>
<tr>
<td>AL1</td>
<td>0.60253017431526E+08</td>
<td>1.0627777076374E+10</td>
<td></td>
<td></td>
</tr>
<tr>
<td>BPOT</td>
<td>0.24198146703951E+05</td>
<td>0.42134672619870E+06</td>
<td>6.459760308903E+04</td>
<td></td>
</tr>
<tr>
<td>EPOT</td>
<td>0.60907488052260E+06</td>
<td>1.0572979409260E+08</td>
<td>1.5310305746570E+06</td>
<td>3.6550192076097E+07</td>
</tr>
</tbody>
</table>

### Eigenvalues of Information Matrix

- 4.999696
- 1.58342
- 3.55574E+07
- 1.06530E+10

### Eigenvectors

### Decomposition Performance Index

- 209972

### Unit Dispersion Matrix

<table>
<thead>
<tr>
<th>Parameter</th>
<th>ALO</th>
<th>AL1</th>
<th>BPOT</th>
<th>EPOT</th>
</tr>
</thead>
<tbody>
<tr>
<td>ALO</td>
<td>0.62967353191217E-05</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>AL1</td>
<td>0.35673493124507E-06</td>
<td>2.118974136616E-07</td>
<td></td>
<td></td>
</tr>
<tr>
<td>BPOT</td>
<td>0.59564973365873E-05</td>
<td>1.1553843441793E-06</td>
<td>2.1715305431902E-01</td>
<td></td>
</tr>
</tbody>
</table>

Figure 4.2 Example of Spey Program Output Statistics
FINAL RESULTS OF OPTIMIZATION

QUASI-NEWTON METHOD

<table>
<thead>
<tr>
<th>PARM NO.</th>
<th>VARIABLE LOCATION</th>
<th>VALUE</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>ALO</td>
<td>1.253775</td>
</tr>
<tr>
<td>2</td>
<td>ALI</td>
<td>8.37142E-02</td>
</tr>
<tr>
<td>3</td>
<td>UPOT1</td>
<td>2.87698</td>
</tr>
<tr>
<td>4</td>
<td>SPOT1</td>
<td>-2.359692</td>
</tr>
</tbody>
</table>

FINAL ROOT MEAN SQUARE RESIDUALS

<table>
<thead>
<tr>
<th>ERROR NO.</th>
<th>RESIDUAL LOCATION</th>
<th>VALUE</th>
<th>WEIGHT</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>THRERR</td>
<td>4.01635</td>
<td>1.00000</td>
</tr>
</tbody>
</table>

AVERAGE RMS ERROR = 4.01635

FINAL STATISTICS

RESIDUAL SUM OF SQUARES = 5.2425.9
NORM OF THE GRADIENT = 6.92278
NO. OF FUNCT EVALUATIONS = 112,000
EST NO. OF SIGN. DIGITS = 7,63067
FINAL MARQUARDT PARAMETER = 0
NUMBER OF ITERATIONS = 0

NORMAL TERMINATION

Figure 4.2 (Concluded)
Typically, for identification runs on different data records, the parameter estimates will be different. This is primarily due to the stochastic nature of the inputs. In some situations, accurate values of the parameters may be obtained when an optimum input time history excites the system response in a favorable way. Other data records may contain information about the parameters but not of the same quality as optimum inputs. The results of the parameter estimation runs can be combined to form the optimum estimate for the overall series of data records by weighting the average by the predicted parameter covariance.

The exact formula is given in terms of the information matrices, $M_i$, for each of $q$ runs:

$$
\overline{\theta} = \left[ \sum_{i=1}^{q} M_i \right]^{-1} \left[ \sum_{i=1}^{q} M_i \hat{\theta}_i \right] \tag{4.15}
$$

where $M_i = \frac{1}{N_i} R_o^{-1} H_i$ and $\hat{\theta}_i$ is the parameter estimate for the $i$th record. An approximate formula can be used with reasonable accuracy as follows:

$$
\overline{\theta}_j = \left[ \sum_{i=1}^{q} (D_{jj}^i)^{-1} \right]^{-1} \left[ \sum_{i=1}^{q} (D_{jj}^i)^{-1} \theta_i \right] \tag{4.16}
$$

where $D_{jj}^i$ is the $j$th diagonal element of the $i$th dispersion matrix. The above formula can be used to obtain weighted averages of the parameter values accurately reflecting the overall responses.
The model developed in Chapter II was used as the foundation of a parameter estimation program. In this way, the code used for identification could be directly transferred to the AWJSTOLRA simulation without reformulation of the modeling equations. The parameter values would be the same as identified from the data.

The identification procedure is described in Chapter IV and a listing of the simulation/identification program is included in Appendix A. Preliminary parameter estimates were graphically calculated from the flight data as described in Chapter III.

The simulation can be used to identify parameters in portions of the engine using measured or derived variables as input. The parameters which are identified can be optionally selected. Table 5.1 shows the parameters of the engine simulation which are available to estimate. Variable name definitions associated with block diagram variables (Figure 5.1) are shown in Table 5.2. The program was sequentially exercised to identify throttle parameters, engine pressure dynamics, engine dynamics, and engine/control dynamics. The identification setup used and the results are presented in the following subsections.

5.1 THROTTLE IDENTIFICATION

The throttle hysteresis value and pickoff bias and scale factor were identified. In addition, a power level dependent hysteresis width was hypothesized and added to the parameter list. Figure 5.2 shows the program set-up. Measured throttle commands are used as input, the throttle pick-off is commanded
Table 5.1

SPEY ENGINE SIMULATION PARAMETERS
AMSTOL RESEARCH AIRCRAFT
SYSTEMS CONTROL, INC (VT)
1001 PAGE MILL ROAD
PALO ALTO, CALIFORNIA

----------
* * *
* RIGHT ENGINE *
* FLIGHT NUMBER 324 *
* RUN NUMBER 5 *
* STI *
* 56000 *
----------

<table>
<thead>
<tr>
<th>THROTTLE</th>
<th>CONTROL</th>
<th>ENGINE</th>
<th>DUCT/EXHAUST</th>
</tr>
</thead>
<tbody>
<tr>
<td>THROTTLE POSITION</td>
<td>31.50 DEGREES</td>
<td></td>
<td></td>
</tr>
<tr>
<td>THROTTLE PICKOFF</td>
<td>976.0 COUNTS</td>
<td></td>
<td></td>
</tr>
<tr>
<td>MAIN FUEL FLOW</td>
<td>5750 LBM/HOUR</td>
<td></td>
<td></td>
</tr>
<tr>
<td>COMPRESSOR SPEED</td>
<td>1.335E +05 REV/MIN</td>
<td></td>
<td></td>
</tr>
<tr>
<td>FAN SPEED</td>
<td>1.008E +05 REV/MIN</td>
<td></td>
<td></td>
</tr>
<tr>
<td>AUGMENTOR PRESSURE</td>
<td>50.00 LBF/SQIN</td>
<td></td>
<td></td>
</tr>
<tr>
<td>DUCT PRESSURE</td>
<td>50.00 LBF/SQIN</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

FULL POWER POINT

----------
* * *
* OPTIMIZATION PARAMETERS *
* * *
| NO. OF MEASUREMENTS | 6 |
| NO. OF PARAMETERS | 4 |
| NO. OF ERROR TERMS | 1 |
| NO. OF DATA POINTS | 3250 |
| NO. OF FIRST POINT | 1 |
| NO. OF INCREMENT | 1 |
| DATA PRINT INTERVAL | 0 |
| CONSTANT PRINT FLAG | 0 |
| EPSILON | 1.00E +03 |
| DELTA | 1.00E +03 |
| MAX FUNCTION EVALS | 300 |
| NO. OF SIGNIF DIGITS | 4 |

----------
Figure 5.1 Proposed Spey Engine Identification Model Showing Program Symbology
Table 5.2
Spey Parameter Index

<table>
<thead>
<tr>
<th>PARAMETER INDEX</th>
<th>SYMBOL</th>
<th>NAME</th>
</tr>
</thead>
<tbody>
<tr>
<td>11</td>
<td>ALO</td>
<td>Constant throttle hysteresis</td>
</tr>
<tr>
<td>12</td>
<td>AL1</td>
<td>Linear throttle hysteresis</td>
</tr>
<tr>
<td>13</td>
<td>AL2</td>
<td>Quadratic throttle hysteresis</td>
</tr>
<tr>
<td>14</td>
<td>BPOTØ</td>
<td>Throttle pickoff bias</td>
</tr>
<tr>
<td>15</td>
<td>BPOT1</td>
<td>Throttle pickoff scale factor</td>
</tr>
<tr>
<td>16</td>
<td>DLINK2</td>
<td>Internal control hysteresis</td>
</tr>
<tr>
<td>17</td>
<td>KTH</td>
<td>Throttle to RPM transfer factor</td>
</tr>
<tr>
<td>18</td>
<td>ALPHA</td>
<td>Throttle nonlinearity constant</td>
</tr>
<tr>
<td>19</td>
<td>KAF</td>
<td>Area to fuel gain</td>
</tr>
<tr>
<td>20</td>
<td>CFCNTR</td>
<td>Pressure drop governor feedback</td>
</tr>
<tr>
<td>21</td>
<td>ZETA</td>
<td>Fuel flowmeter damping ratio</td>
</tr>
<tr>
<td>23</td>
<td>OMN</td>
<td>Fuel flowmeter natural frequency</td>
</tr>
<tr>
<td>26</td>
<td>CLAG</td>
<td>Compressor speed time constant</td>
</tr>
<tr>
<td>29</td>
<td>CF</td>
<td>Fuel to RPM gain</td>
</tr>
<tr>
<td>30</td>
<td>AFL</td>
<td>Fuel to low rotor speed gain</td>
</tr>
<tr>
<td>31</td>
<td>CHL</td>
<td>High rotor to low rotor gain</td>
</tr>
<tr>
<td>32</td>
<td>FLAG</td>
<td>Fan speed time constant</td>
</tr>
<tr>
<td>35</td>
<td>TLAG</td>
<td>Tail pipe pressure time constant</td>
</tr>
<tr>
<td>38</td>
<td>TGAIN</td>
<td>Tail pipe pressure to RPM gain</td>
</tr>
<tr>
<td>39</td>
<td>DLAG</td>
<td>Duct pressure lag</td>
</tr>
<tr>
<td>42</td>
<td>DGAIN</td>
<td>Duct pressure to RPM gain</td>
</tr>
<tr>
<td>43</td>
<td>KA1</td>
<td>Flowmeter estimator gain</td>
</tr>
<tr>
<td>44</td>
<td>KA2</td>
<td>Flowmeter estimator gain</td>
</tr>
<tr>
<td>46</td>
<td>EPSBIS</td>
<td>Throttle bias</td>
</tr>
<tr>
<td>47</td>
<td>KFSF</td>
<td>Flowmeter scale factor</td>
</tr>
<tr>
<td>48</td>
<td>KFBIAS</td>
<td>Flowmeter bias</td>
</tr>
<tr>
<td>50</td>
<td>PTDBIS</td>
<td>Duct pressure bias</td>
</tr>
<tr>
<td>51</td>
<td>PT6BIS</td>
<td>Tail pipe pressure bias</td>
</tr>
<tr>
<td>52</td>
<td>XNLBIS</td>
<td>Fan speed pressure bias</td>
</tr>
</tbody>
</table>

70
Figure 5.2 Setup for Throttle Hysteresis Identification

with the derived pick-off signal and parameters adjusted to maximize the fit. Program convergence was quite consistent on a run-to-run basis. The fit results for two runs are shown in Table 5.3. An examination of the fit errors indicate that the right engine modeling accuracy is significantly poorer than the left engine. The data shows a large amount of noise on both the throttle command and pick-off signal relative to the left engine. This noise is averaged by the identification procedure and does not significantly affect the accuracy of the resulting parameter estimates. Table 5.4 shows that the power
Table 5.3
Throttle Input Fits

<table>
<thead>
<tr>
<th>RUN</th>
<th>RECORD</th>
<th>NO. OF POINTS</th>
<th>RMS THRERR LEFT (COUNTS)</th>
<th>RMS THRERR RIGHT (COUNTS)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>1</td>
<td>6000</td>
<td>1.67</td>
<td>6.10</td>
</tr>
<tr>
<td>2</td>
<td>5</td>
<td>3250</td>
<td>1.49</td>
<td>3.02</td>
</tr>
</tbody>
</table>

level dependent hysteresis width is not a significant parameter in the model. This may be inferred by the small F-ratio and large fractional standard deviation. Table 5.5 shows the average values of the final model for the throttle hysteresis and pick-off.

5.2 IDENTIFICATION OF PRESSURE DYNAMICS

The tailpipe and duct dynamics were modeled as pressure lags. It was expected that exhaust pressure lag would be insignificant. The duct lag was more uncertain. An attempt was made to identify the terms in a linear model of the flow as shown in Figure 5.3. The rotor speed measurements are used as input quantities. The error term is calculated from the measured and derived pressure. The terms modeled are the time lag, flow gain, and bias. Sensor scale factors are lumped with the flow gain. Input and output sensor biases are combined in the bias term. The flow lag and any sensor response lags are modeled in the time constant term. Typical transient pressure sensor response times are less than 10 msec and the rotor speed lag is less than this. Any identified lag should be associated with the flow dynamics in the exit volume.
Table 5.4
Parameter Estimation Results for Throttle Hysteresis Using Flight Data
of Records 1 and 5

<table>
<thead>
<tr>
<th>TABLE 5.4</th>
<th>Parameter Estimation Results for Throttle Hysteresis Using Flight Data of Records 1 and 5</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>LEFT ENGINE</strong></td>
<td></td>
</tr>
<tr>
<td><strong>PARAM NO.</strong></td>
<td>VARIABLE</td>
</tr>
<tr>
<td>1</td>
<td>AL0</td>
</tr>
<tr>
<td>2</td>
<td>AL1</td>
</tr>
<tr>
<td>3</td>
<td>BPOTO</td>
</tr>
<tr>
<td>4</td>
<td>BPOT1</td>
</tr>
<tr>
<td><strong>RIGHT ENGINE</strong></td>
<td></td>
</tr>
<tr>
<td><strong>PARAM NO.</strong></td>
<td>VARIABLE</td>
</tr>
<tr>
<td>1</td>
<td>AL0</td>
</tr>
<tr>
<td>2</td>
<td>AL1</td>
</tr>
<tr>
<td>3</td>
<td>BPOTO</td>
</tr>
<tr>
<td>4</td>
<td>BPOT1</td>
</tr>
</tbody>
</table>
### Table 5.5
Combined Throttle Estimates

<table>
<thead>
<tr>
<th></th>
<th>RUN 1 EST/FSD (%)</th>
<th>RUN 2 EST/FSD (%)</th>
<th>AVERAGE EST</th>
<th>UNITS</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>LEFT</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>ALO</td>
<td>0.252/2.5</td>
<td>0.254/1.4</td>
<td>0.252</td>
<td>Deg</td>
</tr>
<tr>
<td>BPOTO</td>
<td>977/.17</td>
<td>987/.17</td>
<td>982</td>
<td>Counts</td>
</tr>
<tr>
<td>BPOT1</td>
<td>-26.2/.27</td>
<td>-26.4/.29</td>
<td>-26.3</td>
<td>Counts/Deg</td>
</tr>
<tr>
<td><strong>RIGHT</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>ALO</td>
<td>0.251/16</td>
<td>0.254/1.0</td>
<td>0.254</td>
<td>Deg</td>
</tr>
<tr>
<td>BPOTO</td>
<td>937/.16</td>
<td>928/.16</td>
<td>942</td>
<td>Counts</td>
</tr>
</tbody>
</table>

* EST/FSD = Estimate/Fract. Std. Dev.

---

![Figure 5.3](image-url)

**Figure 5.3 Equation Error Identification of Pressure Dynamics**
An estimation run on record one was made. Results are presented in Table 5.6. The fits are shown in Appendix C. It is apparent that the linear model of the pressure is not completely satisfactory in measuring large pressure responses. This is probably due to a combination of sensor and pressure response nonlinearity. The estimated pressure gains and bias are best fits to the particular data record. The exhaust lag is identified as zero reenforcing the intuitive conclusion. The duct lag time constant is 0.049 sec. This figure represents an insignificant value relative to the engine dynamics. It is consistent with the lumped volume time constant of the flow.

5.3 IDENTIFICATION OF ENGINE PARAMETERS.

The parameters determining the dynamic response of the engine to fuel flow inputs were identified. The program setup is shown in Figure 5.4. The measured fuel flow is the input used. Error signals are derived from rotor speed measurements and pressure measurements. Pressure gains and scale factors are introduced as free parameters in addition to the linear model engine equations. Data records representing triangle wave and doublet inputs were used.

Table 5.7 lists the fit errors for identification of left and right engine parameters for the three sets of data. Representative time history matches are included in Appendix C. The results of the estimation runs are shown in Tables 5.8 and 5.9 for the left and right engines, respectively. In general, the engine variables are accurately and consistently estimated. Intermediate values such as pressure gains and bias are not as consistent. These terms have been more accurately identified in setup in Section 5.2. The inconsistencies arise because of a nearly unidentifiable combination of bias and scale factor on the instruments and pressure response nonlinearity. This does not affect the accuracy of the engine variables.
Table 5.6
Identification of Pressure Dynamics
(Record No. 1, 6000 Data Points)

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* Not identifiable.
** Estimate not reliable because of large sampling period relative to lag.
Figure 5.4 Engine Parameter Identification
Table 5.7
Fit Errors for Engine Parameter Identification Runs

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### Table 5.9
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<table>
<thead>
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</tr>
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<tr>
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<td>CHL</td>
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</tr>
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<td>DGAIN</td>
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</table>
Table 5.10 shows the results averaged by the estimated variances. Accurate estimates of rotor time constants and coupling were obtained from the data processing. The engine-to-engine variation in parameters is less than 10% except for the compressor time constant. The data indicates that the right engine time constant is 35% larger than the left engine. Compressor and turbine build differences, seal leakage, bleed differences, deterioration, etc., are possible reasons for this type of variation. It should be observed that the fan and compressor response roots differ by about 50%. Modeling engine transients with a single time constant can lead to erroneous results as described in Chapter II.

5.4 IDENTIFICATION OF CONTROL PARAMETERS

There are three parameters in the Spey which determine the characteristics of the hydromechanical speed governor as modeled in the simulation. The setup for identification is shown in Figure 5.5. The assumptions made in the analysis of the governor produce this simple form. The results of the parameter estimation will verify these assumptions.

The assumption that the positive feedback from the pressure drop governor just cancels the natural engine damping is an important simplification in the simulation. The rationale is briefly reviewed below. The transfer function from fuel to speed is a first order lag. If it is assumed that the pressure drop governor provides positive feedback with gain \( C_P \), then the metering area to RPM transfer function is as follows:

\[
\frac{\delta N_H}{A_m} = K_{AF} \frac{C_F}{1 - \frac{C_F}{C_F}} \frac{1}{\tau_{PD}s + 1}
\]

where

\[
\tau_{PD} = \frac{\tau_H}{1 - \frac{C_F}{C_F}}
\]
Table 5.10
Parameter Estimates for Engines

<table>
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<td>RUN 3</td>
<td>RUN 1</td>
<td>RUN 2</td>
<td>RUN 3</td>
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<td>$r$</td>
<td>$\theta$</td>
<td>$r$</td>
<td>$\theta$</td>
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<td>$\bar{\theta}$</td>
<td>$\bar{r}$</td>
<td>$\theta$</td>
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<td>.379</td>
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* Not used in average

$\theta$ = Parameter estimates

$r$ = Functional standard deviation
Figure 5.5 Simulation Setup for Control Parameter Identification
Clearly, when \( C_F = C_F^* \), the pressure drop governor has caused the engine to behave like a rate servo. The speed governor has a control as follows (assume a linear metering characteristic):

\[
\delta A_m = K_{TH} \delta_T - \delta N_H
\]

and the closed-loop transfer function can be written as follows:

\[
\frac{\delta N_H}{\delta_T} = K_{TH}^\mu \frac{1}{\frac{r_H^\mu}{K_{AF}C_F} + s + 1}
\]

where

\[
\mu = 1 + \left(\frac{1 - C_F/C_F^*}{K_{AF}C_F}\right)^{-1}
\]

which indicates that the parameter \( C_F^* \) is difficult to identify since \( K_{TH} \) proportionally determines the d.c. gain and \( K_{AF} \) proportionally determines the closed-loop time constant. The "identifiability" of \( C_F^* \) decreases as \( \mu \) approaches 1. There is a good justification that \( \mu \) is designed to be nearly 1. This can be seen from the metering area d.c. gain as follows:

\[
\delta A_m = \frac{1 - C_F/C_F^*}{1 - C_F/C_F^* + K_{AF}C_F} \delta_T \quad \text{(d.c.)}
\]

It is desirable to hold the metering area at null since the acceleration limits are determined by physically limiting \( \delta A_m \). Null d.c. \( \delta A_m \) is held if \( C_F = C_F^* \). This assumption is made in the simulation. Mismatches in those two values cause the estimated value of \( K_{AF} \) to be slightly biased, but this effect should be second order since \( \mu \approx 1 \).
Using this assumption, $K_{AF}$ is identified with the closed-loop response determination and $K_{TH}$ with the operating (static) behavior. The values of $K_{TH}$ are accurately determined by steady ground trim runs and verification at selected in-flight points. This value is not estimated.

The results of the estimation for the parameter, $K_{AF}$, are shown in Table 5.11 for two data records. Appendix C contains sample plots of the time history match to the predicted fuel flow. The loop time constant, $(\tau_{H})_{CL}$, is calculated in the table. These values are reasonably close to those obtained from the graphical analysis.

5.5 SUMMARY

Component blocks of the Spey engine simulation have been used to estimate parameters from the flight data. Accuracy values have been calculated for each configuration. Several runs have been used to generate each set of results. In general, the run-to-run consistency has been excellent. Table 5.12 lists the final parameter estimates for each engine and the uncertainty limits associated with these estimates. The confidence interval suggested for the parameters is calculated using the fractional standard deviation and a subjective estimate of the overall fit of the data estimation. Worst case values may lie with an interval two or three times larger than the suggested bounds.
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<th>(τ_H)_CL (SEC)</th>
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Table 5.12
Final Spey Engine Parameter Estimates

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<tr>
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<td>±0.05</td>
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</table>

^a/ Not significant
^b/ Estimate from inspection of data
^c/ Defined as equal to CF - See Chapter II
^d/ Derived from steady-state operating line
^e/ No significant estimate derived from data
6.1 SUMMARY

The Spey engines on the AWJSTOL research aircraft represent a complex system of coupled dynamics. Flight data was acquired during controlled throttle tests in the powered lift approach configuration of the aircraft. A dynamical model of the throttle linkage, control system, engines, and duct was developed from an analysis of the physical configuration of the engine. The flight data was utilized to develop initial estimates of parameter values using classical frequency response and describing function techniques. A system identification program was used to process the flight and calculate estimates of the parameter value in the engine model. The resulting engine simulation should provide an accurate component representation for inclusion in the AWJSTOLR aircraft simulator.

6.2 CONCLUSIONS

The quality of the flight data provided was excellent. Inputs controlled by the avionics computer were exceptionally repeatable and accuracies associated with parameters were quite good for this reason. The data processing was nearly routine with the identification and run-to-run repeatability of the parameter estimates was good.

The results indicate that the engine is represented, for outer loop simulation and control evaluation, as a second order system. Thrust outputs must be scheduled on fan and compressor speeds separated to model the forces generated by the engines. The model developed is accurate above about 18° throttle angle.
The behavior of the engine and control, aside from the hysteresis, is quite linear. An insignificant asymmetry in the acceleration and rates was found for these power levels.

The engine-to-engine repeatability of the parameter estimates was within 10% except for the compressor time constant. The right engine time constant is 35% longer than the left. Fan speed responses are the same. The fuel flow to rotor speed gains are nearly the same, hence, the engine fuel consumptions are nearly equal. Fan speed responses are 50% slower than compressor time constants.

The utilization of automated parameter identification procedures has provided a significant improvement over classical methods. The number of parameters estimated is far greater than previously available from graphical techniques. Run-to-run repeatability has been demonstrated for many of the variables.

The established system identification routines can be used in the design and evaluation of future flight test applications. Several of these areas are mentioned below.

The tests described in this report primarily produced linear dynamic results. The scope of the effort was restricted to generally linear models relating the engine response near full power. A more complete representation would use response data from a larger power envelope to estimate both steady state and dynamic coefficients.

A nonlinear model can be used to match the response using this data for the full envelope of operation including nozzle angle and choke effects. Pressure data in the duct and tailpipe could be matched much more accurately using a quasi-linear model of the following form:

\[ \dot{p} = -\lambda(p - \xi(N)) \]
where the function, \( f(\cdot) \), describes the nonlinear static operating line response in pressure. The results for duct pressure response indicate that the choice of \( \lambda \) is not important since \( \lambda > 20 \text{ sec}^{-1} \) for both the duct and tailpipe dynamics.

The accuracy of the identification procedure can be improved using optimal input design. These techniques build upon the results of this study. Estimates for model parameters developed from sawtooth and doublet inputs form the input design model. Throttle input time series are generated from the design procedure which could be programmed on the AWJSTOLR aircraft flight computer as throttle rate commands. Data from these flight tests would be used with the identification program to calculate a revised set of parameter estimates. This procedure would produce far more efficient utilization of the flight test time by enhancing parameter identifiability. Effects of nozzle position and choke deflection on thrust dynamics could be efficiently evaluated.
REFERENCES


PROGRAM SPEY(INPUT,OUTPUT,TAPES=INPUT,TAPE6=OUTPUT,TAPE2/=12000) JUN17 1
  SPEY 3
  SPEY 4
  SPEY 5
  SPEY 6
  SPEY 7
  SPEY 8
  SPEY 9
  SPEY 10
  SPEY 11
  SPEY 12
  SPEY 13
  SPEY 14
  SPEY 15
  SPEY 16
  SPEY 17
  SPEY 18
  SPEY 19
  SPEY 20
  SPEY 21
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  SPEY 37
  SPEY 38
  SPEY 39
  SPEY 40
  SPEY 41
  SPEY 42
  SPEY 43
  SPEY 44
  SPEY 45
  SPEY 46
  SPEY 47
  SPEY 48
  SPEY 49
  SPEY 50
  SPEY 51
  SPEY 52
  SPEY 53
C NAMELIST / INPUT / START, CH
A DLINK1, AL1, AL2, UPOT1, POT1, DLINK2, KTH, ALPHA, KAF, CFCNTH
B ZETA, OHM, SIG, CLAG, CFL, AIL, FLAG1, TAG1, TAG2, DLAG, TLAG2
C DGA1N, KAI, K2, ENRG, EPSBIS, PTDBIS, PT6BIS, XNBIS
D NIN, NP, NF, NDF, NFIRST, NLAST, NINC, SPRINT, CPRT, IPAR, ERRPN
E R+NS16, EPSDELTA, MAXFN, IOPT, PATT11, KPSF, KPSB, BIT
F XLEN2, TTIME, FLT, MUNDEL, MUNDF1, MUNFF1, MUNNL1
G HUNPT, HUNFD1, DFAC, CPLOT, IOPTIM, IEHESS
H HESS, CH
C DATA (RLAB(I)*1+2)/4NPARM+4HEIER/
C *** INITIALIZE VARIABLES ***
C CALL SETUP
75 C
C DFAC=0.1
C CPLOT=0.0
C PLOTF=0.0
C IOPTIM=0
C IEHESS=0
C *** READ INPUT DATA ***
C READ(5, INPUT)
C CALL CPRT
C C
C DT=0.02*NINC
C *** READ DATA ***
C IF(DIT, LT, 0.0) CALL RDATA
C IF(DIT, GE, 0.0) CALL TDATA
C *** INSERT INITIAL PARAMETER GUESSES ***
C DO 220 I=1, NP
C IND=IPAR(I)
C P(I)=PAR(I)*IND
C 220 CONTINUE
C *** PFR1RUB INITIAL GUESSES FOR BIT CASE ***
C IF(BIT, LT, 0.0) GO TO 200
C DO 225 I=1, NP
C P(I)=P(I)*1.0+DFAC
C 225 CONTINUE
C 200 CONTINUE
C IF(IOPTIM, EQ, 1) GO TO 235
C *** OPTIMIZATION ROUTINE ***
C CALL ZXSIII(FUNC, NF, NP, NSIG, EPS, DELTA)
C MAFXN, IOPT, PATT11, KPSF, KPSB, BIT
C XJAC, NP, XJMT, XNL, XNF, INFER, IER)
GO TO 230

C ***QUASI-NEWTON OPTIMIZATION***
C
235 CONTINUE
   IF(IEHESS,E0,0) GO TO 239
   IND=NP+NP(I)/2
   DO 237 I=1,IND
      XJTI(J(I))=E0,0
   237 CONTINUE
   DO 238 I=1,NP
      INDI(I+1)/2
      XJTI(IND(I))=HESS(I)
   238 CONTINUE
C
239 CONTINUE
   CALL 2XMN(FUNCT,NP,NSIG,MAXFN,IEHESS,
            + P,XJTI,XJAC,SIG,WORK,IER)
C
230 CONTINUE
C ***OUTPUT RESULTS***
C
140 CALL CPRINT
C
140 WRITE(6*100)
130 FORMAT(111*10X,29HFINAL RESULTS OF OPTIMIZATION/)
C
145 IF(IOPTIN,EQ,0) WRITE(6*122)
145 IF(IOPTIN,EQ,1) WRITE(6*125)
122 FORMAT(111*32HMARQUAT NONLINEAR LEAST SQUARES/)
123 FORMAT(111*19HQUASI-NEWTON METHOD/)
130 IF(NF,EQ,0) GO TO 50
130 WRITE(6*101)
101 FORMAT(111*8HPARAM NO.4X8HVALUE,12X)
* LOCATION4X8HVALUE)
C
150 DO 2 I=1,NP
150 IND=IPARN(I)
150 WRITE(6*102) I,CSYBA(IND),IND,P(I)
150 FORMAT(111*13HAX,5X13,5X,12,4)
2 CONTINUE
C
50 CONTINUE
   IF(IOPTIN,EQ,1) CALL FUNC(P,NP,NF,F)
   WRITE(6*101)
   103 FORMAT(111*10X32HFINAL ROOT MEAN SQUARE RESIDUALS/)
C
165 IF(NF,F0,0) GO TO 51
165 WRITE(6*104)
104 FORMAT(111*9HERROR NO.4X8HVALUE,12X)
* LOCATION4X8HVALUE,12X6HWEIGHT)
C
170 DO 3 I=1,NF
170 IND=ERRPNT(I)
170 NP01NF=NDP/NINC

PROGRAM SPEY

10 FORMAT(1X,17HCONVERGENCE CODE=13/)
269 CONTINUE

C 205 IF(IOPTIM, EQ, 1) GO TO 265
WRITE(6,108) IER 
108 FORMAT(1X,17HCONVERGENCE CODE=13/)
269 CONTINUE
C 210 IF(IER, EQ, 0) WRITE(6,109) 
109 FORMAT(10X,16HNORMAL TERMINATION) 
C 215 ***ABNORMAL TERMINATION ENDING MESSAGES*** 
C 220 IF(IER, EQ, 129) WRITE(6,110) 
220 IF(IER, EQ, 129) WRITE(6,110) 
270 CONTINUE
C 225 IF(IER, EQ, 130) WRITE(6,111) 
225 IF(IER, EQ, 130) WRITE(6,111) 
C 110 FORMAT(10X,3SHSINGULAR JACOBIAN-RECOVERY FAILED)
111 FORMAT(10X,2I1) INPUT PARAMETER ERROR
112 FORMAT(10X,3L1) MAXIMUM HARKWATD PARAMETER EXCEEDED
113 FORMAT(10X,3L1) SINGULAR JACOBIAN - RECYCLE FAILURE
114 FORMAT(10X,3L1) NO. OF FUNCT EVAL EXCEEDED
115 FORMAT(10X,3L1) NMANNING JACOBIAN ZERO - STATIONARY POINT
121 FORMAT(10X,25H UNGO ERROR TOO LARGE)
271 CONTINUE
C ***PRINT INFORMATION MATRIX***
C IF(NP.EQ.0.0) GO TO 310
DO 300 I=1,NP
IND=IPARM(I)
RLAB(I+2)=CSYMW(IND)
300 CONTINUE
C WRITE(6,116)
116 FORMAT(1H1*,/30X+25HSCALFD INFORMATION MATRIX)
C CALL USMBSM(RLAB,0,0,0,NP,XJTJ)
C ***CALCULATE EIGENSYSTEM***
C WRITE(6,117)
117 FORMAT(/10X+3H EIGENSYSTEM OF INFORMATION MATRIX/)
C CALL EIGRSC(XJTJ,NP,2,DEIG,ZVECT,NP,IK,IERS)
C WRITE(6,118) (DEIG(I)\ I=1,NP)
118 FORMAT(10X+12H EIGENVALUES\10X+12.6)
C WRITE(6,119)
119 FORMAT(/10X+13H EIVENVECTORS/)
C WRITE(6,124) IJK(I)
124 FORMAT(10X+3H DECOMPOSITION PER INDEX=2X+12.6)
C CALL USHTFM(00,0,0,NP,NP,ZVECT)
C ***CALCULATE DISPERSION MATRIX***
C CALL LINV2P(XJTJ,NP,DISP,TDGT,D1,D2,WORK,IERS)
C IF(IERS.EQ.2) GO TO 310
C WRITE(6,130)
130 FORMAT(/10X+2H dispersion matrix)
C CALL USMLSM(RLAB,0,0,0,NP,DISP)
C ***ERROR ANALYSIS***
C IF(IOPTH.EQ.1) GO TO 330
C K=(NP+1)*NP/2
```fortran
DO 331 I=1,IP
    XJAC(I)=WORK(I+J)
331 CONTINUE
330 CONTINUE

290 C   DO 332 J=1,IP
    XJAC(J)=XJAC(J)+P(J)/SSD
332 CONTINUE
C
205 WRITE(6,131)
131 FORMAT(/11X,8HPARM NO.,1X,BVARIABLE,1X,5HVALUE,
     $+ 6X,14HFRACT,STD.DEV.,13X,7HF RATIO,1X,13HFRACT,SENSIT.)
C DO 333 I=1,IP
300  IND=IPARM(I)
    IDIAG=I*(I+1)/2
    FSTDV=SQRT(DISP(IDIAG))
    IF(P(I).NE.0.0) FSTDV=FSTDV/P(I)
    FRATIO=1./FSTDV**2
305 WRITE(6,132) IND,P(I),FSTDV,FRATIO,XJAC(I)
132 FORMAT(13X,13X,8X,A6,13X,4(G12.6,2X))
333 CONTINUE
C
C
310 C ***SET UP PLOTS IF REQUIRED***
C
310 CONTINUE
C
315 C
STOP
END
```
SUBROUTINE FUNCT(NP,P,F)
DIMENSION P(1)
DIMENSION FIND(10)
C
COMMON /CONSIS/ STARTCH(8)
A DLINK1,AL0,AL1,AL2,BP0T0
B NBF1,DLINK2,KTH,ALPHA,KAF
C CFCH1,ZETA,AM0,OMH,SIG,DT
D CLAG,CLAGD,CLAGG,CF
E AFL,CHL,FLAG,FLAGD,FLAGG
F TLAG,TLAGD,TLAGG,TD
G DLAG,DLAGD,DLAGG,DDA
H KAI,KA2,ENGMO,EPRFIS
I KIFS,KBIAST,BIT
C
COMMON /CONSIS/
A PTBIS,PT6BIS,XNLBIS
B NM,NP,NP,FIRSTNYNC,SPRINT,CPRINT
C IPARM(20),ENPRNT(6),R(6),NSIG
D EPS,DELT,MAAFN,IOPT,PARMK1(4)
E FILEN,STIM,FLGINO,HUNDEL,HUNDPO
F HUNNFF,HUNNCHUNNL,HUNPT5,HUNPTD
COMMON /CONSIS/DU2(100)
C
C ***GENERATE FUNCTIONS FOR MINIMIZATION***
C
CALL FUNC(P,NP,NF,FIND)
C
C ***GENERATE SUM OF SQUARES***
C
F=0.0
DO 1 I=1,NF
F=F+FIND(I)**2
1 CONTINUE
RETURN
END
**Routine ENGINE**

**SPEY ENGINE SIMULATION FOR NASA AWJSTOLKA AIRCRAFT**

```
REAL KH, KAF, KA1, KA2, KFSF, KFBIA
COMMON /CONSIS/: START, CH
COMMON /CONSIS/: PTDIS, PBDIS, XNLBIS
COMMON /CONSIS/: NM, NF, NFPRT, SPRINT, CPRNT
COMMON /CONSIS/: IPERM(20), ERRRTI(6), R(6), NSIG
COMMON /CONSIS/: EPS, DELTA, MAXFN, IOPT, PARAM(4)
COMMON /CONSIS/: FILENO, STINE, FLGTO, HUNDEL, HUNDPO
COMMON /CONSIS/: HUNF0, HUNH, HUNNL, HUNPT0, HUNPTD
COMMON /CONSIS/: SOFT(XI, XM, XB) = AMAX1(0, (XI - XB)/(XB - XM))
```

```
IF(START, EQ, 0, 0) GO TO 10
```

```
***INITIALIZATION PASS***
```

```
THERR = 0.0
FULERR = 0.0
XHSErr = 0.0
XLSErr = 0.0
PTDERR = 0.0
PTDERN = 0.0
DLINK1 = AL2 + DLELC**2 + AL1 + DLELC + AL0
DLINK1 = SOFT(DLINK1, 100, 0, 05)
DEL1 = DELT - DLINK1
DEL1S = DELT - DLINK2
```

```
```
C ***STABILITY LIMITS***

60

  CLAG=SOFT(CLAG,-100.,.1 )
  FLAG=SOFT(FLAG,-100.,.1 )
  TLAG=SOFT(TLAG,-100.,.005 )
  DLAG=SOFT(DLAG,-100.,.005 )
  KAF =SOFT(KAF ,-.100.,.5 )
  CF  =SOFT(CF ,-.100.,.01 )
  DLINK2=SOFT(DLINK2,-100.,.01 )

C ***FLOATING SETUP***

70

  SIG=-1./ZETA*OMN)
  OM=SQRT(OMN**2-SIG**2)
  F11=EXP(SIG*DT)*COS(OM*DT)
  F12=EXP(SIG*DT)*SIN(OM*DT)
  F22=F11
  F21=F12
  GS=(EXP(SIG*DT)*(SIG*SIN(OM*DT)+OM*COS(OM*DT))+OM)/OMN**2
  GC=(EXP(SIG*DT)*(SIG*COS(OM*DT)+OM*SIN(OM*DT))-SIG)/OMN**2
  G1=(OM/2)*(GS-GC)
  G2=G1

C WFMEST=WFMN

C ***ROTOR SPEED SETUP***

80

  XNH=XNHMSN
  CLAGD=0.0
  IF(CLAG>NE.0,0) CLAGD=EXP(-DT/CLAG)
  CLAGG=1.-CLAGD

90

  XNL=XNLMSN
  FLAGD=0.0
  IF(FLAG>NE.0,0) FLAGD=EXP(-DT/FLAG)
  FLAGG=1.-FLAGD

C ***PRESSURE DYNAMICS SETUP***

95

  PT=PTMSN
  TLAGD=0.0
  IF(TLAG>NE.0,0) TLAGD=EXP(-DT/TLAG)
  TLAGG=1.-TLAGD

100

  PTN=PTNMSN
  DLAGD=0.0
  IF(DLAG>NE.0,0) DLAGD=EXP(-DT/DLAG)
  DLAGG=1.-DLAGD

C ***MAIN ROUTINE***

105

10 CONTINUE

C ***LINKAGE TO THRUST***

CALL HYSTER(DELTC+DELT+DLINK1)
**THROTTLE PICKUP**

DELTAPO = BPT01 \* DELIT + BPT02
DELTT1 = DELIT
DELTT2 = (DELTMS - HPOT01) / BPT01

**ENGINE CONTROL MODEL**

CALL HYSTER(DELTMS, DTHRO1, PLINK2)
XH = XN1H
IF (CH(1), EQ. 1) XH = XN1MSN
EPSRPM = KTH * (DTHRO1 - 31.5) (XH = 100.) + EPSBIS

**FLOWSHUTTER***

IF (START, EQ. 0, 0) GO TO 20
FH51 = 0.5 * (KHPB = 100.)
FH52 = FH51
FH5L1 = FH51
FH5L2 = FH52

**FUEL FLOK FILTER**

FH51 = FH51 + KAI1 * (KFNH = KFMES1)
FH5L2 = FH52 + KAI1 * (KFNH = KFMES1)

**BACKWARDS CALCULATION***

WF1 = (1. / G1) * (FMFS1 = FMFS1 + F12 * FMFSL1 + F12 * FMFSL2)
WF2 = (1. / G2) * (FMFS2 = FMFS2 + F12 * FMFSL1 + F12 * FMFSL2)
FMFEST = 0.5 * (KAI1 + KAI2) + 100.
FMFS1 = FMFS1
FMFS2 = FMFS2

**COMPRESSOR**

WF = WFRM = KEIAS
IF (CH(2), EQ. 11) WF = WFMES1 = KBIAS
IF (CH(2), EQ. 22) WF = (KFNH = KFBIAS) / KSF

XMH = CLAG exports
C  
C ***FAN ROTOR***  
C  
175  
XN=XNH  
IF(CH(5),EQ,1) XN=XNHMSN  
C  
XNL=FLAGD*(XNL-100.)*FLAGG*CHL*(((XN-100.)*0.1)-AFL)  
A  
C  
XNL=XNL+XNLDIS  
C  
***TAIL PIPE***  
C  
185  
XN=XNH  
IF(CH(6),EQ,1) XN=XNHMSN  
C  
PT6=TLAGD*(PT6-100.)*TLAGG*(((XN-100.)*TGAIN)+100.  
PT6=PT6+PT6DIS  
190  
C  
***DUCT***  
C  
195  
XH=XNL=XNLDIS  
IF(CH(7),EQ,1) XH=XNLMSN=XNLDIS  
C  
PTD=DLAGD*(PTD-100.)*DLAGG*(((XH-100.)*TGAIN)+100.  
PTD=PTD+PTDDIS  
C  
START=0.0  
C  
RETURN  
C  
END  
ENGINE 155  
ENGINE 156  
ENGINE 157  
ENGINE 158  
ENGINE 159  
ENGINE 160  
ENGINE 161  
ENGINE 162  
ENGINE 163  
ENGINE 164  
ENGINE 165  
ENGINE 166  
ENGINE 167  
ENGINE 168  
ENGINE 169  
ENGINE 170  
JUN178 15  
JUN178 16  
ENGINE 171  
ENGINE 172  
ENGINE 173  
JUN17A 2  
JUN17 5  
ENGINE 176  
JUN178 17  
JUN178 18  
ENGINE 179  
JUN21 1  
JUN21 2  
ENGINE 160  
ENGINE 161
SUBROUTINE SETUP

REAL KTH, KAF, KA1, KA2, KSF, KFBIAS
COMMON /CONSTS/ START, CH(9)

A DLINK1, ALO, AL1, AL2, BPO10
B APO11, DLINK2, KTH, ALPHA, KAF
C CFCTR, ZET1, OM, OMN, SIG, DT
D CLAG, CLAGD, CLAGG, CF
E AFL, CHL, FLAG, FLAGD, FLAGG
F TLAG, TLAGD, TLAGG, TGAIN
G DLAG, DLAGD, DLAGG, DGA1N
H KAI, KAZ, ENGHU, EPBIAS
1 KFSF, KFBIAS, BIT

COMMON /CONSTS/

A PI8BIS, PTEBIS, XMNRIS
B NP, NF, NDNP, HFLST, HQI, SPRINT, CPRNT
C IPARM(20), ERHPT(6), R(6), NSIG
D EPS, DELTA, MAXFN, IOPT, PARMHQ(4)
E FILENO, STIME, FLGSTO, HUNDEL, HUNPO
F HUNISF, HUNH, HUNHL, HUNPT6, HUNPT0

DIMENSION DUH(150), EQUIVALENCE(IUHIII, START)

DO 1 I=1,150
1 DUH(I)=999.0

C *** INITIAL PARAMETER GUESSES ***

C JUNE 1, 1977

START=1.0

DO 2 I=1,8
2 CH(I)=0.0

C DLINK1=

ALO=0.1
AL1=0.0
AL2=0.0
BPO10=-29.1
BPO11=40.22

DLINK2=0.5

DLINK2=0.1
KTH=0.7212
ALPHA=0.10
KAF=5.09
CFCTR=0.140
ZET1=0.2
OM=
OMN=13.0
SIG=

DT=0.020
CLAG=0.50

C CLAG=

C CLAGG=

CF=0.140

AFL=0.0

CHL=2.61

FLAG=1.0

SETUP 2
SETUP 3
SETUP 4
SETUP 5
SETUP 6
SETUP 7
SETUP 8
SETUP 9
SETUP 10
SETUP 11
SETUP 12
SETUP 13
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SETUP 48
SETUP 49
SETUP 50
SETUP 51
SETUP 52
SETUP 53
SETUP 54
SETUP 55
SETUP 56
SETUP 57
SETUP 58
C FLARG= 61  
C FLAGD=  
60 TLAG = 0.05  
C TLADD=  
C TLAGG=  
C DLAG = 0.10  
C DLAGU=  
65 DLAG=  
D GAIN = 0.70  
D GAIN = 0.70  
D KAI = 0  
D KAI2 = 0  
D ENNNO = 1  
E EPSBIS = 0.0  
E KFSF = 1.0  
E KFTAS = 0.0  
F BIT = 999  
75 F1DIBIS = 0.0  
F1DIBIS = 0.0  
XNLBIS = 0.0  
C  
80 NH = 0  
NPF = 0  
NP = 0  
NO P = 1  
NFIRST = 1  
NTNC = 1  
85 SPRINT = 0  
CPRNT = 0  
DO 3 I = 1, 20  
3 IPARN(I) = 4 + I  
DO 4 I = 1, 6  
4 EPRPNI(1) = 1  
DO 5 I = 1, 6  
5 R(I) = 1.0  
NSIO = 0  
EPS = 1.0 E = -4  
MAXFN = 20  
DELT A = 1.0 E = -4  
INF T = 0  
PARNHO(1) = 0.0  
PARNHO(2) = 0.0  
PARNHO(3) = 0.0  
PARNHO(4) = 0.0  
FILENO = 1.0  
STINE = 0.0  
FLCFTH = 0.0  
HUNDCL = 31.5  
HUNPDO = 976  
HUNMHF = 5750  
HUNNHF = 13350  
HUNHL = 10075  
HUNP1 = 50  
HUNPTD = 50  
RETURN  
END  
SETUP 59  
SETUP 60  
SETUP 61  
SETUP 62  
SETUP 63  
SETUP 64  
SETUP 65  
SETUP 66  
SETUP 67  
SETUP 68  
SETUP 69  
SETUP 70  
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SETUP 108  
SETUP 109  
SETUP 110  
SETUP 111  
SETUP 112  
SETUP 113  
SETUP 114  
SETUP 115
115  IF (I.EQ.NLAST) GO TO 8
INDEX=INDEX+1
IF (INDEX.NE.SPRINT OR SPRINT.EQ.0) GO TO 4
INDEX=0
CALL DPRINT(0)

120  C 4 CONTINUE
C 8 IF (SPRINT.NE.0) CALL DPRINT(1)
C
125  C ***SCALE FUNCTIONS***
C 130  IF (INF.EQ.0) GO TO 53
GO TO 11 INF
IND=ERROR(I)
F(I)=F(I)/R(IND)
F(I)=SORT(F(I))
10 CONTINUE
C 53 CONTINUE
IF (ICPRINT.EQ.1) WRITE(6+110) (F(I),I=1,NF)
C
RETURN
END
SUBROUTINE FUNC(P, NP, NF, F)

***DRIVER PROGRAM FOR ENGINE SIMULATION AND IDENTIFICATION***

DIMENSION F(1), P(1), X(1), E(1), PARM(1), Y(1)
REAL KTH, KAF, KA1, KA2, KAFS, KPHAS
LEVEL 2 * XDATA
COMMON /ENGINE/
A PRINT, TIME, DELIC, DELTMS, WFSN
B XHMSH, XNLMSN, PT6MSH, PTDH
C DELTPO, WFEQST, XNH, XNL, P16
D PID, DELT, DELT2, WFHS, WFLWM
E VREST, DELT1, DELT2, EPSRF
F NTHNOT, DAREA, FHS, FHS2, FMSF
G, SFMSF, THREHR, FLUREHR, XNHERR
H XLSRR, PTMHER, PTIDRR
COMMON /CONSIS/, START, CH(8)
A DINK1, AL0, AL1, AL2, AP0, A
B DPTN1, DPTN2, KTH, ALPHA, KAF
C CFNTR, ZETA, OH, OON, SIG, DT
D CLAG, CLAGD, CLAGS, CF
E APL, CHL, FLAS, FLAGD, FLAGG
F SLAG, TLAG, FLAG, FLAG2
G DLAG, DLAGD, DLAGS, DGA1
H KAI, KA2, ENGN0, EPSRAS
I KSF, KPHI, BIT
COMMON /CONSIS/
A PTDMH, PTDMH2, XNL0IS
B Nh, NFX, NF, NFIRS, NPIC, SPRINT, CPRINT
C PARM(20), ERH, HTR, R, RNSIG
D EPS, DELTA, MAXFN, I0PT, PARMH(4)
E FILEND, STIME, FLGNO, HUNDEL, HUNPD
F HUNRFF, HUNNH, HUNNL, HUNPT6, HUNPTD
COMMON /DATA/, XDATA(10,1)

EQUIVALENCE (START, PARM(NP))
1 (TIME * X(1)),
2 (THREHR * E(1)),
3 (DELIP0 * Y(1))

***INSERT PARAMETERS***

IF(CPRINT, NE, 1, 0) GO TO 100
CALL CPRINT
WRITE(6, 110) (P(I), I = 1, NP)
110 FORMAT(2X, 6G20.10)
100 CONTINUE

CALL INEX = 1
IF(NP, EQ, 0) GO TO 50
DO 1 I = 1, NP
IND = I
1 PARM(IND) = P(I)

***INITIALIZE SIMULATION***
C 50 CONTINUE
START = 1.0
NM=NF
DO 2 I=1+ND
2 X(I)=XDATA(I+FIRST)
C 65 POINT=FIRST
CALL ENGINE
IF(SPRINT,NE.,0.,0.) CALL DPRINT(0)
C C 70 ***ZERO INNOVATIONS***
C IF(NF.EQ.0.) GO TO 51
DO 3 I=1+NF
3 F(I)=0.0
C 75 ***SIMULATION LOOP***
C 51 CONTINUE
NLAST=FIRST+(ND+1)*NINC
DO 4 I=FIRST+NLAST+NINC
POINT=I
C 80 CALL ENGINE
C C 85 ***TRANSFER OUTPUTS TO MEASUREMENTS FOR TEST***
C IF(NF*NE.0.,OR.BI1.LT.0.0) GO TO 54
C 90 DO 5 J=1+NM
X(J+2)=Y(J)
XDATA(J+2)=Y(J)
5 CONTINUE
C 95 54 CONTINUE
C C 99 ***INSERT MEASUREMENTS***
C DO 5 J=1+ND
5 X(J)=XDATA(J+I)
C C ***CALCULATE ERRORS***
C DO 6 J=1+NM
6 E(J)=Y(J)-X(J+2)
C C ***ACCUMULATE ERRORS***
C IF(NF*FR.0.) GO TO 52
DO 7 J=1+NF
IND=ERRPHOT(J)
7 F(J)=F(J)+E(IND)**2
C 92 CONTINUE
SUBROUTINE CPRINT

COMMON /CONSTS/ START*CH(8)
   A DLINK1, ALU, AL1, AL2, BPORT
   B BPORT, DLINK2, KTH, ALPHA, KAF
   C CFCNTR, ZETA, DH, DNL, SIG, DT
   D DLAG, DLAG, DLAG, DLAG
   E AFL, CHL, FLAG, FLAG, FLAG
   F TLAG, TLAG, TLAG, TAGAIN
   G DLAG, DLAG, DLAG, DLAG
   H KAI, K42, ENHON, EPSHIS
   I KFSF, KPOINS, WI

COMMON /SYMBUL/NDSYM, NCSYM, DSYM(50), CSYM(150)

DIMENSION ENLAB(2), IND1(20), IND2(20), IND3(20), IND4(20), PARH(20)

EQUIVALENCE (PARH(1:), START)

DATA (IND1(I):I=1,10)/
* 10 10 11 12 13 14 15 21 22 43/
DATA (IND2(I):I=1,10)/
* 17 19 16 20 44 46 47 48 22 24 44/
DATA (IND3(I):I=1,10)/
* 26 32 29 30 51 52 27 28 33 34/
DATA (IND4(I):I=1,10)/
* 5 35 38 39 42 50 36 37 40 41/

DATA ENLAB/ENLEFT,*ENRIGHT/

WRITE(6,100)

100 FORMAT(1H1///)

WRITE(6,101)

101 FORMAT(50X,3H5 SPEY ENGINE SIMULATION PARAMETERS/ 
* 50X*25HAWSTOL RESEARCH AIRCRAFT/ 
* 50X*25HSYSTEMS CONTROL* INC (VT)/ 
* 50X*20H1801 PAGE HILL ROAD / 
* 50X*20H1801 ALTO, CALIFORNIA//)

WRITE(6,102)

102 FORMAT(56X*21(1H1))/

IF(DIT.LT.0.0) GO TO 10

WRITE(6,103) BIT

103 FORMAT(56X*1H1*19X,1H1/ 
* 56X*1H1*3X*3HBUILT IN TES1*3X*1H1/ 
* 56X*1H1*3X*5HRAIPE*15*16 DEG / 
* 56X*1H1*19X,1H1/ 
* 56X*21(1H1))/

GO TO 20
SUBROUTINE DPHINIT

COMMON/ENGINE,X(50)
COMMON/SYMBOL,NDSYMB,NSYMB(50),GSYMB(150)
COMMON/CONST,BUM(50)
COMMON/PLOTS,CPLOT,PLUTF,IND1,A,200,7,IMAG4,5151,PDATA,200,20

DIMENSION P(10,50)

DATA IND/0/

NDSYMB=35
IND=IND+1
DO I=1,NDSYMB
P(IND,I)=X(I)
CONTINUE

IF(N.EQ.0.AND.IND.EQ.10) GO TO 200
IF(PLUTF.GT.0.0) GO TO 200

WRITE(*,100)
100 FORMAT(1X,49X,22HSPEY ENGINE SIMULATION,/*
* 50X,39H ENGINE PARAMETER IDENTIFICATION PROGRAM,/*
* 50X,24HSYSTEMS CONTROL, INC. VT),/*
* 50X,20HPALO ALTO, CALIFORNIA,/*
* 50X,11HVERSION 001,/*)

DO I=1,NDSYMB
WRITE(*,101) NSYMB(I)*(P(I)+1),IND
101 FORMAT(1X,26X,12I2,6)

IND=0

***SAVE PLOT DATA IF PLOTTING***

DO 2 CONTINUE
2 CONTINUE

IF(PLUTF.EQ.0.0) RETURN

IND=IND+1
IF(IND.GT.200) RETURN

PDATA(IND,I)=X(I)
DO 210 I=1,7
PDATA(IND+2*I)=X(I+2)
210 CONTINUE

DO 220 I=1,7
PDATA(IND+2*I+1)=X(9+I)
220 CONTINUE

RETURN
END
C 10 CONTINUE

C 60  I=ENGNO
   J=LOTHO
   K=FILENO
   L=TIME

   WRITE(*,104) ENLAB(I),J,K,L
   104 FORMAT(5X,1H*,19X,1H*/
         * 50X,1H*,2X,A55,1H* ENGINE,4X,2H*/
         * 50X,1H*,19X,1H*)
   70  CONTINUE

C 20 CONTINUE

C 75  WRITE(*,105)
   105 FORMAT(26X,8I(1H*)/26X,1H*,19X,1H*,19X,1H*,19X,1H*,19X,1H*/
         * 26X,1H*,5X,B.THROTTLE,6X,1H*,7X,7HCONTROL/
         * 5X,1H*,7X,bENGINE,6X,1H*,4X/
         * 12HDUCT/EXHAUST,3X,1H*/,
         * 26X,1H*,19X,1H*,19X,1H*,19X,1H*/26X,8I(1H*))

C 80  DO 50 I=1,10
       I1=IND1(I)
       I2=IND2(I)
       I3=IND3(I)
       I4=IND4(I)

   50 CONTINUE

C 85  WRITE(*,110) CSYHB(I1),PARM(I1),
        CSYHB(I2),PARM(I2),
        CSYHB(I3),PARM(I3),
        CSYHB(I4),PARM(I4)
   110 FORMAT(26X,4(IH*,A6,1H=,G12,6),1H*)

C 90  50 CONTINUE

C 95  WRITE(*,111)
   111 FORMAT(26X,8I(1H*)//)
   WRITE(*,120)
   120 FORMAT(26X,4I(1H*)*,6X,3A(1H*))
   WRITE(*,121)
   121 FORMAT(26X,4I(1H*)*,12X,16FULL POWER POINT,11X,1H*,6X,1H*,4X,
         * 2SHOPTIMIZATION PARAMETERS,5X,1H*))

C 100  WRITE(*,122)
   122 FORMAT(26X,1H*,39X,1H*,6X,1H*,32X,1H*)

C 105  WRITE(*,131) HUND1,1H
   WRITE(*,132) HUNDPP,NP
   WRITE(*,133) HUNDFF,NF
   WRITE(*,134) HUNH,NP
   WRITE(*,135) HUNH,NF
   WRITE(*,136) HUNP0,NINC
   IPRINT=PRINT
   WRITE(*,137) HUNP0,IPRINT
SUBROUTINE CPRINT J76/7b OPT=1

FORM 4.5*914 10/06/77 11:15:06

! 115
C
   116 FORMAT(C
   117 CPRINT
! 118 CPRINT
   119 CPRINT
   120 FORMAT(C
   121 CPRINT
   122 CPRINT
   123 CPRINT
! 124 CPRINT
   125 CPRINT
   126 CPRINT
   127 CPRINT
   128 CPRINT
   129 CPRINT
   130 CPRINT
   131 FORMAT(C
   132 CPRINT
   133 CPRINT
   134 CPRINT
! 135 CPRINT
   136 CPRINT
   137 CPRINT
   138 CPRINT
   139 CPRINT
   140 CPRINT
   141 CPRINT
   142 CPRINT
   143 CPRINT
   144 CPRINT
   145 CPRINT
   146 CPRINT
   147 CPRINT
   148 CPRINT
   149 CPRINT
   150 CPRINT
   151 CPRINT
   152 CPRINT
   153 CPRINT
   154 CPRINT
   155 CPRINT
   156 CPRINT
   157 CPRINT
   158 CPRINT
   159 CPRINT
   160 CPRINT
   161 CPRINT
   162 CPRINT

C
BLOCK DATA SYMBOLS
COMMON /SYMBOL/NDSYMB,NCSYMB,DSYMB(50),CSYMB(150)
C
DATA DSYMB,CSYMB/200*6H
C
DATA (DSYMB(I),I=1,10)/
A 6HPOINT
B 6HTIME
10 C 6HDELTC
D 6HDELTMS
E 6HMFSNS
F 6HXXMSNS
G 6HXXMSNS
H 6HPT6NS
I 6HPT6NS
J 6HDELTP0
DATA (DSYMB(I),I=1,20)/
A 6HMFSN0
B 6HXXN0
C 6HXXN0
D 6HPT6
E 6HPT6
F 6HDELT
G 6HDELT0
H 6HMFSN
I 6HMFFLM0
J 6HMFFLM0
DATA (DSYMB(I),I=21,30)/
* 6HDELT
A 6HDELT2
B 6HEPSRN
C 6HMRTHRT
D 6HDATA
E 6HFNS1
F 6HFNS2
G 6HFNS0
H 6HFNS2
I 6HTHRE0/ DATA (DSYMB(I),I=31,40)/
* 6HFULER
A 6HXMS0
B 6HXPS0
C 6HPT6ER
D 6HPT6ER
E 6H
F 6H
G 6H
H 6H
50 I 6H/ DATA (DSYMB(I),I=41,50)/
A 6H
B 6H
C 6H
D 6H
E 6H
F 6H
55
SUBROUTINE RDATA
C ***SUBROUTINE TO READ FLIGHT DATA***
C
5
C COMMON /DATA/XDATA(10,1)
C 0 LINK1, ALO, AL1, AL2, BPO10
B BPT1, BPT2, KTH, ALPHA, KAF
C CFCNTR, ZETA, OH, OMN, SIG, DT
D CLAG, CLAGD, CLAGG, CF
E APL, CHL, FLAG, FLAGD, FLAGE
F FLAGE, FLAGE, TAGE
G NLAG, DLAGE, DLAGE, TGA1
H KAI, KAF, ENAME, EPISH
I KFS, KFBIAS, GUT
C COMMON /CONST/START+CH(4)
A PIDNIS, PIDNIS, PNDS, XNRDG
B MMN, MMN, NF, NDF, NF, NDF, NSIG
C IPHM(20), EPSIGN(1), EPSIGN(1), EPSIGN(1)
D EPS, EPS, EPS, EPS, EPS, EPS
E FILENO, STIME, FLGNO, HUNDE, HUNDE, HUNDE
F HUNOFF, HUNIH, HUNNH, HUNPO, HUNPO, HUNPO
DIMENSION DUM(20,50)
C ***INITIALIZE COUNTERS***
C
30
C STIMEF=0.0
G=0.0
IF(STIME.EQ.-999.0) GO=1.0
N=0
IND=(ENGND=1.0)+10.
C ***START READ LOOP***
C
40
C 10 CONTINUE
READ(2) DUM
IF(EOF(2),NE.0, ) GO TO 20
C
00 1 I=1,50
C
45
C IF(STIME.EQ.DUM(I+1)) GO=1.0
IF(GO.EQ.0.0) GO TO 1
C IF(STIMEF.EQ.0.0) STIME=DUM(I+1)
STIMEF=1.0
C
50
C N=N+1
00 2 J=1,10
XDATA(J,N)=DUM(J+IND+1)
2 CONTINUE
C ***CORRECT SCALING PROBLEM ON N2***
C ***INPUT DATA SCALING ERROR***
C          XDATA(5+N)=DUM(5+IND+I)*0.1
C          XDATA(1+N)=XDATA(1+N)-STIME
C          IF(N.EQ.NDP) RETURN
C                  1 CONTINUE
                   GO TO 10
C                  20 NDP=N
                   IF(NDP.NE.0) RETURN
                   WRITE(6,100)
                   RETURN
100  FORMAT(10X,12HNO DATA READ)
SUBROUTINE PLOICP

COMMON /CONSIS, START/CH(8)
A DLINK1*AL0, AL1, AL2*DPLOT
B DPOT1*DLINK2, KTH, ALPHA, KAF
C CFCNT, ZETA, OH, OMN, SIG, DT
D CLAG, CLAG2, CLAGG, CF
E AFL, CHL, FLAG, FLAGD, FLAGG
F ILAG, TLAGG, TTRAIN
G DLAG, DLAGG, DLAGD, DTRAIN
H KAI, K2, ENGNO, EPSHIS
I KFSF, KFBIAS, HIT

COMMON /CONSIS/
A PTOBIS, PTOBIS, XNLIST
B NP, NF, NP, NF, NF, NF, NF, NF
C IPARM(20), EPRMH(4), R(6), NSIG
D EPS, DELTA, MAXFN, IOPT, PARM4Q(4)
E FILENO, STIME, FLTNQ, HUNDEL, HUNPO
F HUNMFF, HUNNH, HUNNL, HUNPTB, HUNPTD

COMMON /PLOTS/CPLT, PLOTF, INDA(200,7), IMAG4551, PDATA(200,20)

DIMENSION P(1), F(10)

***TRANSFER MATRIX LABELS***

DO 10 I=2,7
   DO 10 J=1,160
   A(J,I)=A(I,J)
10 CONTINUE

***GENERATE PLOT DATA***

DO 20 I=1,7
   DO 20 J=1,160
   IN=J-40
   A(IN+I)=A(J+I)
20 CONTINUE

***OUTPUT PLOTT***

CALL USPLH(PDATA(1,1), PDATA(1,2), NPOINT, 1, 200, A, IMAG441)

CALL USPLH(PDATA(1,1), PDATA(1,2), NPOINT, 1, 200, A, IMAG441)
* 2*I,200,A(I+1),IMAG*RER)
C 1 CONTINUE
C ***ADDITIONAL PLOT CODE***
C RETURN
END
**BLOCK DATA LABELS**

1. COMMON /PLOTS/CPLT,PLOT,INDIA(2007),IMAGQ(5151),PDATA(200+20)

**PLABELS**
- 1 PLABELS 3
- 2 PLABELS 2
- 3 PLABELS 1
- 4 PLABELS 66
- 5 PLABELS 65
- 6 PLABELS 62
- 7 PLABELS 61
- 8 PLABELS 60
- 9 PLABELS 59
- 10 PLABELS 58
- 11 PLABELS 57
- 12 PLABELS 56
- 13 PLABELS 55
- 14 PLABELS 54
- 15 PLABELS 53
- 16 PLABELS 52
- 17 PLABELS 51
- 18 PLABELS 50
- 19 PLABELS 49
- 20 PLABELS 48
- 21 PLABELS 47
- 22 PLABELS 46
- 23 PLABELS 45
- 24 PLABELS 44
- 25 PLABELS 43
- 26 PLABELS 42
- 27 PLABELS 41
- 28 PLABELS 40
- 29 PLABELS 39
- 30 PLABELS 38
- 31 PLABELS 37
- 32 PLABELS 36
- 33 PLABELS 35
- 34 PLABELS 34
- 35 PLABELS 33
- 36 PLABELS 32
- 37 PLABELS 31
- 38 PLABELS 30
- 39 PLABELS 29
- 40 PLABELS 28
- 41 PLABELS 27
- 42 PLABELS 26
- 43 PLABELS 25
- 44 PLABELS 24
- 45 PLABELS 23
- 46 PLABELS 22
- 47 PLABELS 21
- 48 PLABELS 20
- 49 PLABELS 19
- 50 PLABELS 18
- 51 PLABELS 17
- 52 PLABELS 16
- 53 PLABELS 15
- 54 PLABELS 14
- 55 PLABELS 13
- 56 PLABELS 12
- 57 PLABELS 11
- 58 PLABELS 10
- 59 PLABELS 9
- 60 PLABELS 8
- 61 PLABELS 7
- 62 PLABELS 6
- 63 PLABELS 5
- 64 PLABELS 4
- 65 PLABELS 3
- 66 PLABELS 2
- 67 PLABELS 1

**FORMAT**
- 10/6/77
- 11.15.065
- PAGE 1
C

DATA (A(I,5),I=16,200)

C

DATA (A(I,6),I=16,200)

C

DATA (A(I,7),I=16,200)

C

END
```
SUBROUTINE MYSTERIN, XOUT, OZL HYSTER 2

C
XOUT = XOUT
IF (XIN, GT, XOUT + 0.7) XOUT = XIN - 0.7
IF (XIN, LT, XOUT - 0.7) XOUT = XIN + 0.7

C
RETURN
END
```

APPENDIX B

FLIGHT TEST PLAN

1. Fly towards Crows Landing and climb to 7500 ft. Establish communication and telemetry with NASA2 as soon as possible. If unable to establish communication, start the throttle sweeps when 15 DME west of Crows Landing.

2. Establish the airplane in the STOL mode on a -7.1/2° flit path angle. Once stabilized at a speed of LAS + 5 kts proceed with the following matrix of throttle rate square waves.

<table>
<thead>
<tr>
<th>SQUARE WAVE DURATION</th>
<th>AMPLITUDE</th>
<th>RECORD LENGTH</th>
</tr>
</thead>
<tbody>
<tr>
<td>5 sec</td>
<td>1.25°</td>
<td>30 sec</td>
</tr>
<tr>
<td>&quot;</td>
<td>2.5°</td>
<td>&quot;</td>
</tr>
<tr>
<td>&quot;</td>
<td>5.0°</td>
<td>&quot;</td>
</tr>
<tr>
<td>&quot;</td>
<td>7.5°</td>
<td>&quot;</td>
</tr>
</tbody>
</table>

4 to 5 min + climb

3. Climb back to 7500 ft. and proceed through the following matrix.

<table>
<thead>
<tr>
<th>DURATION</th>
<th>GNJ</th>
<th>AMPLITUDE</th>
<th>GNI</th>
<th>RECORD LENGTH</th>
</tr>
</thead>
<tbody>
<tr>
<td>10 sec</td>
<td>100</td>
<td>1/2°/sec</td>
<td>10</td>
<td>40 sec</td>
</tr>
<tr>
<td>&quot;</td>
<td>&quot;</td>
<td>1.5°/sec</td>
<td>30</td>
<td>&quot;</td>
</tr>
</tbody>
</table>

3 min + climb

4. Climb back to 7500 ft and do the following case:

<table>
<thead>
<tr>
<th>DURATION</th>
<th>GNJ</th>
<th>AMPLITUDE</th>
<th>GNI</th>
<th>RECORD LENGTH</th>
</tr>
</thead>
<tbody>
<tr>
<td>50 sec</td>
<td>500</td>
<td>.25°/sec</td>
<td>5</td>
<td>2 cycles</td>
</tr>
</tbody>
</table>

2 min + climb
5. Climb back to 7500 ft. and do the following doublets.

<table>
<thead>
<tr>
<th>DURATION</th>
<th>AMPLITUDE</th>
<th>RECORD LENGTH</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>GNJ</td>
<td>GNI</td>
</tr>
<tr>
<td>10 sec</td>
<td>100</td>
<td>1° 20</td>
</tr>
<tr>
<td></td>
<td>1-1/2°</td>
<td>30</td>
</tr>
<tr>
<td></td>
<td>2°</td>
<td>40</td>
</tr>
<tr>
<td></td>
<td>4°</td>
<td>80</td>
</tr>
</tbody>
</table>
|          | 7-1/2°    | 150           | 30 sec
APPENDIX C

TIME HISTORY MATCHES FOR PARAMETER IDENTIFICATION RUNS

Figure C.1 Throttle Linkage Match - Record 1, Left Engine
Figure C.2 Throttle Linkage Match - Record 5, Left Engine
Figure C.3 Throttle Linkage Match - Record 1, Right Engine
Figure C.4 Throttle Linkage Match - Record 5, Right Engine
Figure C.5 Tailpipe/Duct Pressure Match - Record 1, Left Engine
Figure C.6 Tailpipe/Duct Pressure Match - Record 1, Right Engine
Figure C.7 Engine Parameter Estimation Match - Record 3, Right Engine
Figure C.8 Engine Parameter Estimation Match - Record 7, Right Engine
Figure C.9 Engine Parameter Estimation Match - Record 7, Left Engine
Figure C.10 Fuel Control Estimation Match - Record 3, Right Engine
Figure C.11 Fuel Control Estimation Match - Record 7, Right Engine
Figure C.1 Throttle Linkage Match - Record 1, Left Engine
Figure C.1 (Concluded)
Figure C.2 Throttle Linkage Match - Record 5, Left Engine
Figure C.2 (Concluded)
Figure C.3 Throttle Linkage Match - Record 1, Right Engine
Figure C.3 (Concluded)
Figure C.4 Throttle Linkage Match - Record 5, Right Engine
Figure C.4 (Concluded)
Figure C.5 (Concluded)
Figure C.6 (Concluded)
Figure C.7 Engine Parameter Estimation Match - Record 3, Right Engine
<table>
<thead>
<tr>
<th>Time (msec)</th>
<th>1: Measured</th>
<th>2: Modeled</th>
</tr>
</thead>
<tbody>
<tr>
<td>95.18E+02</td>
<td>222</td>
<td>211H</td>
</tr>
<tr>
<td>94.87E+02</td>
<td>222</td>
<td>22M2</td>
</tr>
<tr>
<td>94.36E+02</td>
<td>222</td>
<td>2212</td>
</tr>
<tr>
<td>93.05E+02</td>
<td>222</td>
<td>2212</td>
</tr>
<tr>
<td>92.03E+02</td>
<td>222</td>
<td>2212</td>
</tr>
<tr>
<td>91.33E+02</td>
<td>222</td>
<td>2212</td>
</tr>
<tr>
<td>91.82E+02</td>
<td>222</td>
<td>2212</td>
</tr>
<tr>
<td>90.80E+02</td>
<td>222</td>
<td>2212</td>
</tr>
</tbody>
</table>

Figure C.7 (Continued)
Figure C.7 (Continued)
Figure C.7 (Continued)
Figure C.7 (Concluded)
Figure C.8 Engine Parameter Estimation Match - Record 7, Right Engine
Figure C.8 (Continued)
Figure C.8 (Continued)
Figure C.8 (Concluded)
Figure C.9 Engine Parameter Estimation Match - Record 7, Left Engine
Figure C.9 (Continued)
Figure C.9 (Continued)
Figure C.9 (Continued)
Figure C.9 (Concluded)
Figure C.10 Fuel Control Estimation Match - Record 3, Right Engine
Figure C.10 (Concluded)
Figure C.11 Fuel Control Estimation Match - Record 7, Right Engine
Figure C.11 (Concluded)