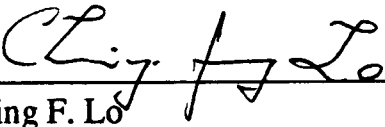


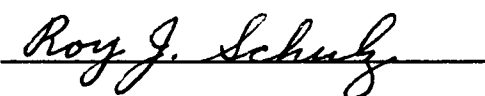
To the Graduate Council:

I am submitting herewith a thesis written by Peter A. Montgomery entitled "A Prototype Expert System For Performance Evaluation Of Liquid-Fueled Rockets." I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aerospace Engineering


Ching F. Lo
Major Professor

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and recommend its acceptance:





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Date 5-9-90

**A PROTOTYPE EXPERT SYSTEM
FOR PERFORMANCE EVALUATION
OF LIQUID-FUELED ROCKETS**

**A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville**

Peter A. Montgomery

May 1990

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ii

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The purpose of this research was to develop a prototype expert system with three main capabilities related to the test and operation of liquid-fueled rockets. These include monitoring the test for any values exceeding safety limits, initiating a rapid post-test performance optimization analysis, and providing the ability to create a life history, including performance optimization advice given, of any rocket examined by the system. Hard copies of all displayed information are available for more traditional analysis and storage when desired.

This thesis describes the capabilities of the expert system, including hypertext presentation, in detail and uses a simulated test run, with three previously created data files of the appropriate numerical information, to show it in actual operation. Also included is a discussion of future expansion for this work. This prototype expert system is utilized to demonstrate the feasibility and the potential benefit of the operational system. The modification and improvement will be required to complete the final version of the operational system.

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CHAPTER I

INTRODUCTION

1.1 Background

This past decade has seen the retirement of some of the best this country has to offer in the aerospace engineering profession. As time goes by this trend will continue, leaving many parts of the field lacking in trained experts. With this disturbing decline has come a real concern in the aerospace industry for preserving the knowledge and experience of these trained professionals for the next generation of aerospace engineers to build upon and lead the United States of America into the twenty-first century. A partial possible solution to this problem has been steadily winning favor among government agencies and throughout the private sector. Coming out of the artificial intelligence field, expert systems are rapidly emerging as the method of choice for preserving the expertise of engineers in an easily accessible form. An expert system is defined as "...a computer program that relies on a body of knowledge to perform a somewhat difficult task usually performed only by a human expert. The principal power of an expert system is derived from the knowledge the system embodies rather than from search algorithms

and specific reasoning methods. An expert system successfully deals with problems for which clear algorithmic solutions do not exist. Human experts reason and arrive at conclusions based on their knowledge; expert systems reason and arrive at conclusions based on the knowledge they possess" [1]¹. Once an expert system is endowed with the knowledge of a trained expert, it may assist less experienced engineers in solving difficult problems in areas of familiarity but extending beyond their area of expertise. It also may assist equally experienced engineers in solving problems more rapidly by providing another expert to confer with. If the knowledge of several trained engineers goes into the creation of an expert system, it can become rather like conferring with an entire team of experts.

Aerospace engineering is one of many fields that lends itself well to the application of expert systems. With a knowledge of liquid-fueled rocket propulsion as well as a background in the creation of expert systems for aerospace applications, a problem was sought, for a master's thesis in aerospace engineering, that would require someone with this combination to solve. Initial studies indicated a great deal of work had

¹ Numbers in brackets refer to similarly numbered references in the bibliography.

been done concerning expert systems for rocket engine design ([9]), but very little could be found on expert systems for rocket engine design ([9]), but problems that occur after a rocket system has been designed and testing has begun.

1.2 Research Objective

The purpose of the research presented in this thesis was to create a prototype expert system to assist in three areas of liquid-fueled rocket testing. These areas include: 1) monitoring an operating test and providing immediate warnings when safety limits are exceeded; 2) initiating a rapid post-test performance optimization analysis; and 3) building a life-history of selected rockets tested. The performance optimization analysis was to provide assistance in the form of advice as to suggested parameter changes that would lead to an improvement in performance without violating the safety limits imposed initially by the user of the system.

One additional feature of the expert system was to be the ability to compare important performance parameters of the rocket engine with those of previous tests of the same rocket and to store them and the advice provided in a database for construction of a life history of a rocket. Another ability of the expert system was to be the computation

of expected performance values and storage of them for later comparison with actual values to further assist in the performance analysis and optimization. A system with these abilities would serve as a valuable tool for decreasing the amount of testing that must be done to determine the conditions under which a given rocket design achieves its best possible performance. It would also be an important reference in the design of similar rocket engines as well as tracking the behavior of a given rocket engine over time.

1.3 Expert System Construction Tools

A wide variety of tools were used in the development of this system. The implementation of the expert system was done using **VP-EXPERT** version 2.1 [14], a rule-based expert system development tool developed by Paperback Software. A rule-based expert system represents its information in a product rule of the form

If premise
Then conclusion ;

or in the form

conclusion
If premise ;

Fulfillment of the premise provides the appropriate advice indicated in the conclusion or in later conclusions whose rule premises' are fulfilled by this conclusion [1]. This package was chosen for its low cost and wide

availability, compared with other expert systems, as well as for the limited graphics and hypertext development capabilities available in this latest software version. The use of hypertext will be discussed later in this report.

Creation of the database files, for documentation of a rocket engine's history, was done using the **dBase III+** and **Lotus 123** software. Only the appropriate field names were entered through their utilization, however. Any storage, modification, or deletion of data from the database files is done through the expert system after the initial file creation. As a result, the user of this expert system need only have the **VP-EXPERT** package [14] and not all three. All necessary files would be provided along with the system.

The **dBase III+** and **Lotus 123** software were used for two reasons. First of all, files created using them are in a format acceptable for use by **VP-EXPERT** [14]. Secondly, they create files that may contain text strings as well as numerical data. This feature was necessary for the storage of performance parameter information together with advice provided for performance optimization in the same file. It also proved useful in the creation and modification of the

hypertext files containing any short comments relating to each test.

1.4 The Description and Utilization of Hypertext

Although not a separate tool in itself, the hypertext capabilities of the **VP-EXPERT** software [14] should be discussed. Hypertext is textual information that is linked to other blocks of text information. When a hyperword, a word linked to a hypertext block of information, is selected, it is replaced by the corresponding hypertext block. A block of hypertext information appears initially to be the same as an ordinary block of text information. Within a hypertext block, however, are previously set key words that when selected, usually with a mouse, provide new blocks of text generally related to the term chosen. In this rocket engine testing expert system, this feature was used to link a short list of six lines of comments for a particular rocket engine test to the name given to the engine test. As a result, after retrieving a listing of all engine tests currently in the database, the selection of a particular engine test may be made, and the comments related to the selected engine will appear on the screen. The engine test listing will disappear from the screen. This becomes very useful for a quick examination of the tests in the database.

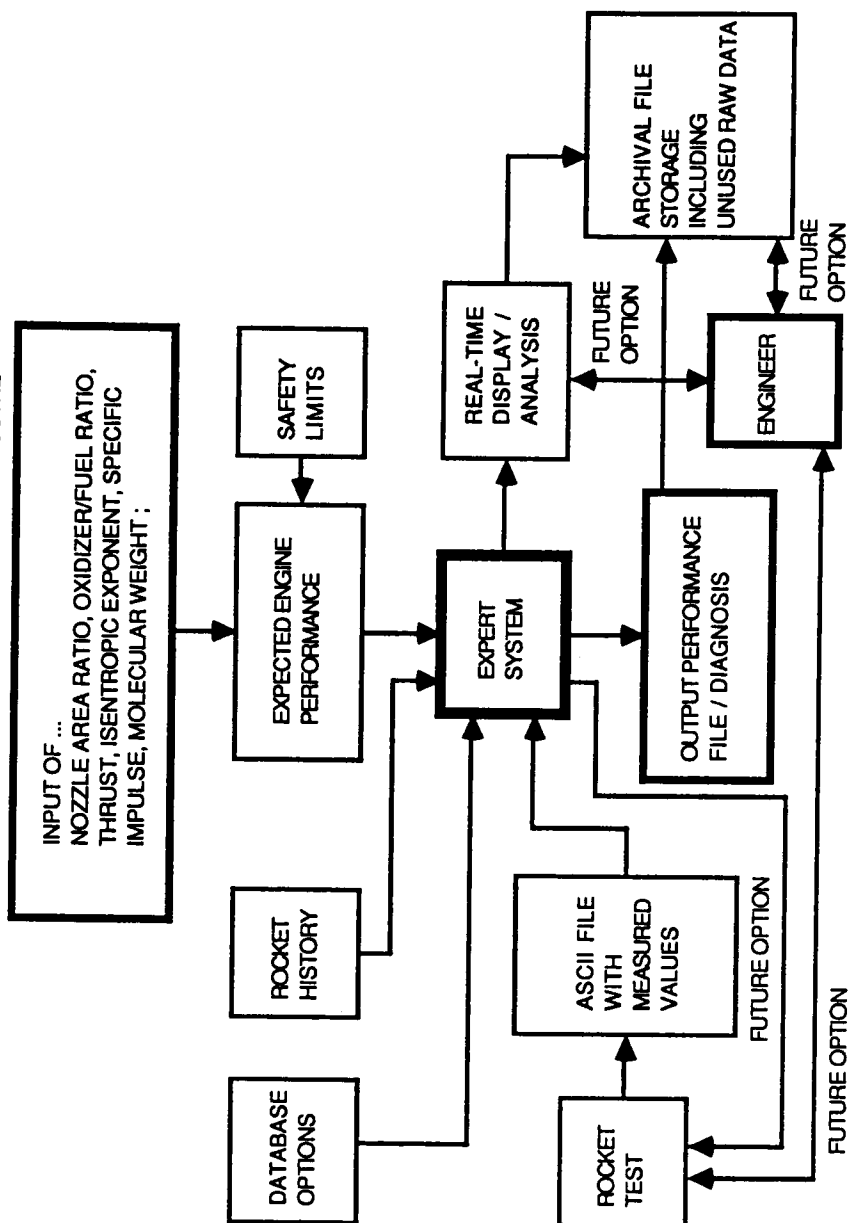
1.5 Rocket Engine Data

To demonstrate the operation of the expert system, typical liquid-fueled rocket engine performance data was desired. Such a set of data was created by referring to the technical manuals for the H-1 rocket engine, models H-1A, H-1B, H-1C, and H-1D [2], and the F-1 rocket engine [3]. These manuals are restricted by the United States government to a limited distribution, so they were only used herein as guidelines. No actual numbers or text information from the H-1 or F-1 engines was utilized in this report or in the expert system itself. The H-1 engine series was also used as a baseline to determine the typical parameters measured in a liquid-fueled rocket engine test as well as the most important safety parameters to consider.

1.6 Expert System Operations Process Overview

Figure 1 shows the operations structure of the expert system. Before the test begins, the important design parameter default values are presented to the user for review and modification. Then the working relationships for ideal rocket performance analysis [Appendix A] are used to determine expected engine performance. Next, safety limitations are entered into the system by the user. Also the method of disposal for the gas generator flow, either dumped overboard or put through the nozzle, is requested. This affects all mass flow rate

FIGURE 1. EXPERT SYSTEM STRUCTURE



calculations, which will be discussed in detail later in this material. Now the rocket test begins and an output stream of raw data (digital and/or analogue signals) is sent to the expert system, after being converted to an ASCII file. The data stream is periodically updated to include current measured values. The ASCII file structure will be discussed in detail later in this report. The necessary manipulations needed to get the information from the rocket test output data stream into the ASCII file in the appropriate format is assumed to be provided by each individual company or agency utilizing the system, since their rocket test information is returned in their own format.

As the expert system reads data from the ASCII file, it compares the values to their earlier defined safety limits and checks for those falling outside acceptable limits. After completely examining the file's information, the user is given the option to continue monitoring the test by reading the continually updated file once again. Some of the data retrieved from the file will be unused in the later performance analysis and is referred to in figure 1 as raw data. This information would be necessary to provide a life history of the engine that includes all conceivable parameters in case the user decides they should be included in the database. These parameters, along with the current time and

elapsed time in the test, are automatically stored and it is then up to the user if this information is to be later included in the database.

Once a test has been completed, the post-test analysis can begin. This analysis includes options to make comparisons of the measured performance values with expected values calculated earlier. The comparisons of measured performance values with the performance values of the tests currently in the database will determine whether the new data values fall within the minimum and maximum values for each parameter of the database, and to request advice on improving performance. The latter is based on the closeness of measured to expected performance values and considers the nearness of the influencing parameters to their safety limits as well. If there is still room for change, the expert system recommends an increase or decrease in the parameter, whichever is appropriate, to achieve better performance. These recommendations are added to the numerical information already stored in an archive file. This will be discussed in detail later in this report.

Future options for this system are also recommended in figure 1. These include the direct control of the rocket engine test by the expert

system with changes made by an engineer to the near-real-time display and archive file directly affecting the actual test. This ability is required to be developed in the future, but this would definitely have a highly potential benefit. This possibility as well as many other extensions for the future of this expert system will be discussed in the conclusions section at the end of this thesis report.

An easy and clear method for monitoring rocket tests, the most important parameters to consider when analyzing and optimizing the performance of a liquid-fueled rocket engine as well as the appropriate safety limits to consider, and the most important parameters to include when constructing the life history of an engine were learned in the conduction of this research and development of this expert system. The final system developed, a simulated test run's results, using precreated data files, and suggestions for the future continued development of this prototype system are included in the following chapter.

CHAPTER II

THE STUCTURE AND DETAILED SECTIONS OF THE EXPERT SYSTEM

2.1 Expert System Structure

When developing an expert system for an aerospace application, it is necessary to come at the problem from different angles. The problem must be examined from a purely programming point of view as well as from an entirely application point of view. Finding the balance, although difficult, results in an excellent expert system for technical utilization. With any reasonably sized programming problem, developing a solution in modular independent form is usually best.

This problem was no exception. The clear concise break-up into pieces of the problem presented in figure 1 is no coincidence. This is in fact the way that the problem was broken up for programming purposes. The entire system is menu driven with the main controlling menu represented in the center of the figure by the entire expert system block. An expanded view of this center block as well as a slightly differently structured diagram for the system may be examined in figure 2. From this it may be seen that the expert system contains the production rule base, representing the knowledge of the system in the rule format discussed in chapter 1, as well as the hypertext system,

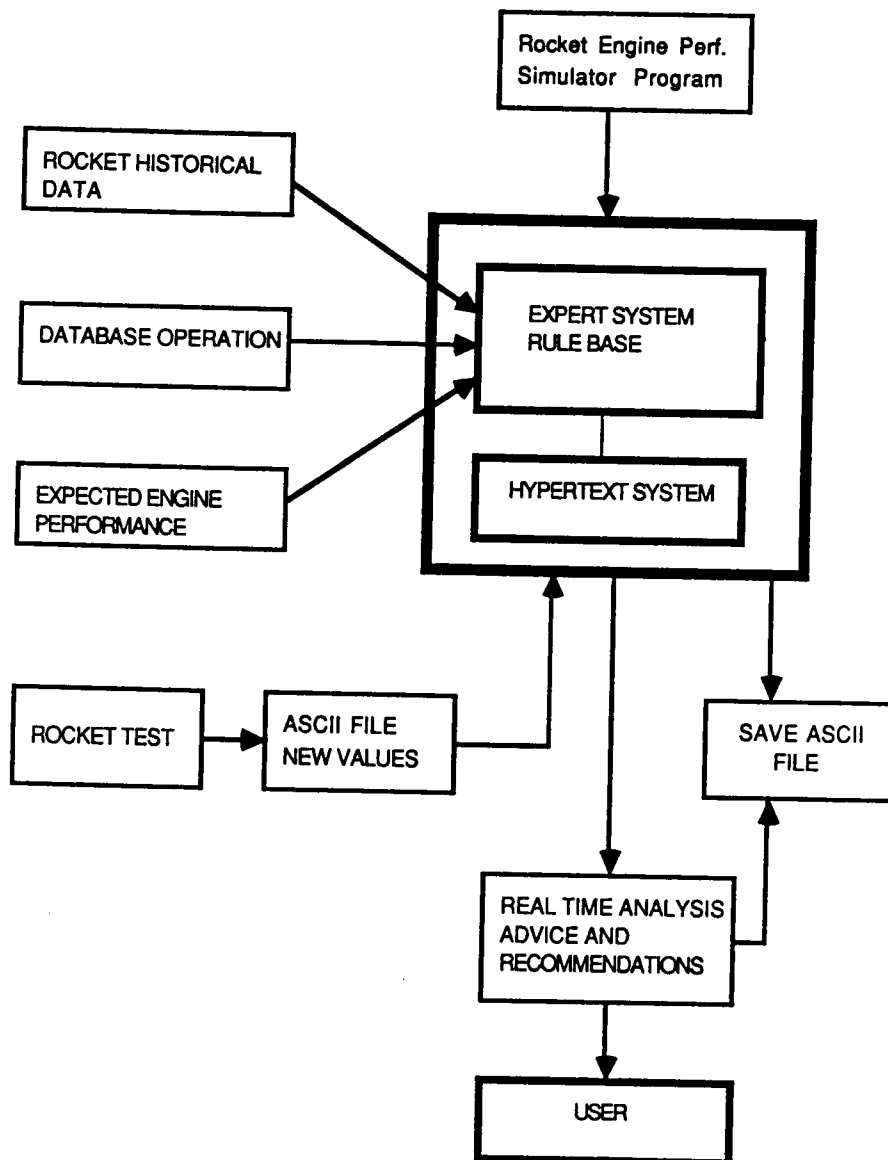


FIGURE 2. EXPERT SYSTEM STRUCTURE

used extensively in the database operations portion of the system for linking engine labels with short summary comments of their tests and with the other engine tests contained in the database.

2.2 Expert System Benchmarking Parameter Inputs

Before entering the main controlling menu, three operations are performed whose results are stored for later use while the test is running. It is assumed at this point, that the test has not yet begun, but it is not absolutely necessary for this assumption to be met. The expert system will operate the same if started at a point before or during the test. It should be noted, however, that once these operations are completed they are never returned to the status during the test. This is a purely one-way link, unlike the other links of figures 1 and 2. This is due to the fact that no change in the parameters defined by these operations is desired during the test since the typical test time is relatively small.

The first operation defines the user designated design characteristics of the liquid-fueled rocket engine currently being tested. They include the nozzle expansion ratio, the oxidizer to fuel flow rate ratio, the thrust the engine was designed to produce, the specific heat

ratio of the propellant, the molecular weight of the propellant, the specific impulse which the engine was designed to produce, and the ambient pressure. Default values are provided for user acceptance or modification. From these design parameters, expected performance values are calculated and stored for later comparison to measured values if so desired.

The second operation is the performance value calculation. Using the working relationships for ideal rocket performance analysis, found in Appendix A, the propellant flow rate (equation 1), the effective exhaust velocity (equation 3), the thrust coefficient (equation 6), and the characteristic velocity (equation 8) are calculated. All performance parameters are then stored in a text file for retrieval by the expert system.

The third operation is the definition of safety limit parameter values and the specification of gas generator output flow handling. The safety limits should be monitored by the expert system and considered when providing optimization advice as well. Figure 3 provides a typical list of these parameters. All limits, with the exception of environmental temperature, are considered maximum values. Environmental

Minimum Environmental Temperature
Maximum Environmental Temperature
Fuel Pump Inlet Pressure Limit
Fuel Pump Inlet Temperature Limit
Fuel Pump Outlet Pressure Limit
Fuel Pump Outlet Temperature Limit
Oxidizer Pump Inlet Pressure Limit
Oxidizer Pump Inlet Temperature Limit
Oxidizer Pump Outlet Pressure Limit
Oxidizer Pump Outlet Temperature Limit
Chamber Pressure Limit
Chamber Injector Pressure Limit
Chamber Temperature Limit
Turbopump Speed Limit
Turbine Exit Static Pressure Limit
Gas Generator Pressure Limit
Gas Generator Temperature Limit
Turbopump Lubricant Consumption Limit
Cooling Flow Rate Limit

Figure 3. Safety Limit Parameters

temperature safety limits are minimum and maximum values.

Gas generator output may be dumped overboard or brought into the rocket engine nozzle for exhaust. The reply to the inquiry, to determine which method is being used, affects the calculated exhaust flow rate. If the gas generator flow enters the nozzle, it must be included in the flow rate calculation. Otherwise it is excluded. This operation completes the parameter inputs for the user with any further inputs being measured from the rocket test.

2.3 Expert System Operations

2.3.1 Expert System Task Selection

These three initial operations now complete, the expert system main controlling menu is accessed and allows user selection of test monitoring or post-test analysis operations. Each possible selection in the main menu represents a line connecting to the center blocks of figures 1 and 2 and a portion of the expert system operations that may be accessed when desired (see figure4).

A short explanation of each line in the main menu can be obtained by clicking the mouse on the light-blue title word "Expert." When this is done, the main menu text disappears and is replaced by predefined hypertext providing the desired explanations (see figure 5A and 5B).

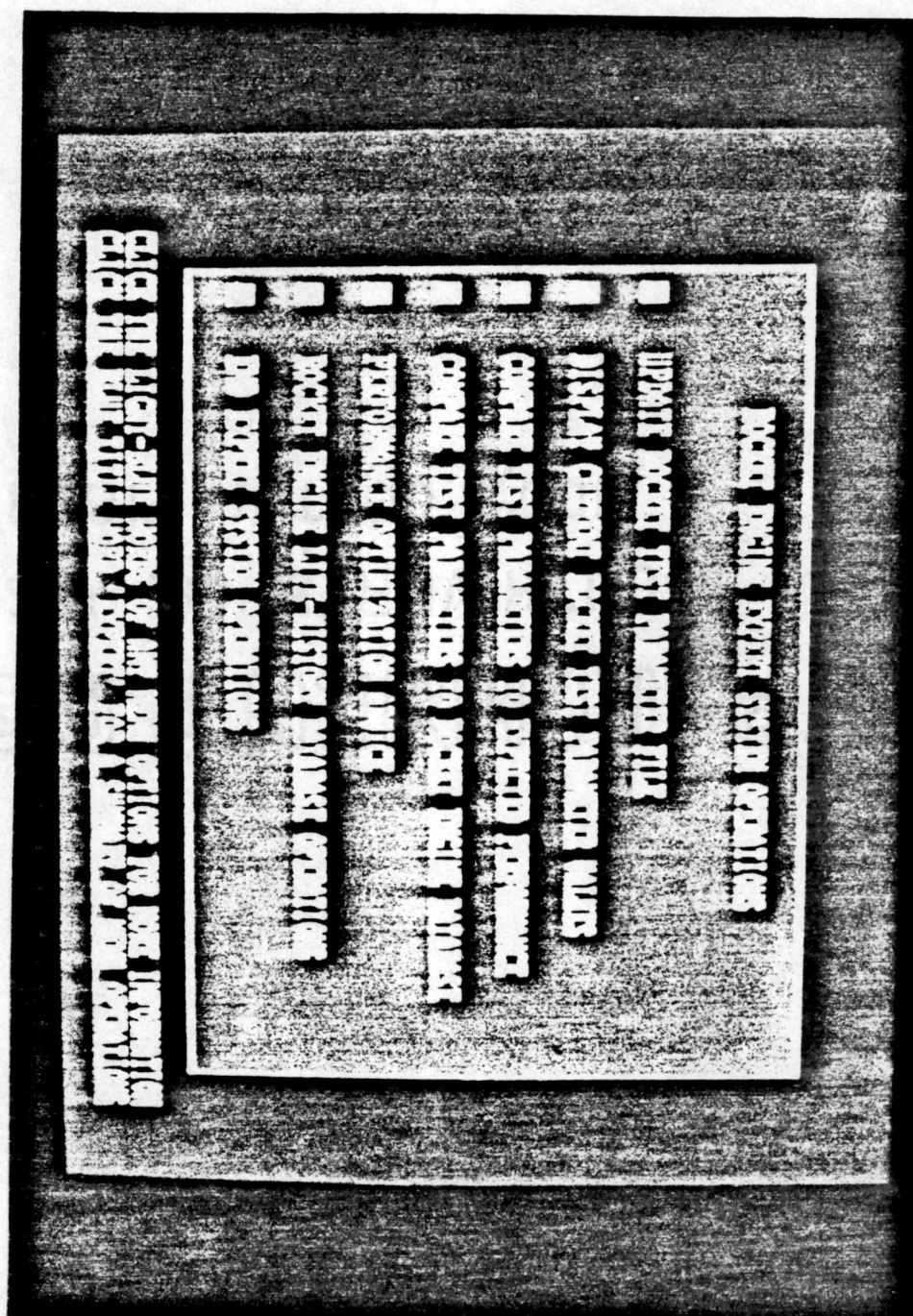


Figure 4. Expert System Main Menu

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
MAIN MENU OPERATIONS SUMMARY
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
. THE FIRST MENU SELECTION CAUSES THE CURRENT PARAMETER
  VALUES FROM AN ONGOING ROCKET ENGINE TEST TO BE READ
  IN. FROM A CONSTANTLY UPDATED ASCII FILE. AND COMPARED
  TO THE USER-DEFINED SAFETY LIMITS.
. THE SECOND MENU SELECTION CAUSES THE CURRENT ROCKET
  PARAMETER DATA SET TO BE DISPLAYED FOR REVIEW ALONG
  WITH ANY RECOMMENDATIONS MADE SO FAR.
. THE THIRD MENU SELECTION CAUSES A COMPARISON OF THE
  CURRENT MEASURED PERFORMANCE VALUES TO THE EXPECTED
  PERFORMANCE VALUES AND THE CALCULATION OF THE RATIO
  OF THE MEASURED TO EXPECTED (IDEAL) VALUES.
CLICK HERE FOR NEXT PAGE:      PAGE2
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION.
CLICK THE BLUE TITLE WORD 'EXPERT' FOR A SUMMARY OF MENU OPERATIONS.
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

```

Figure 5A. Operations Summary-Page 1

```

#####
#####
#####
#####
#####
##### THE FOURTH MENU SELECTION CAUSES A COMPARISON OF THE #####
##### CURRENT MEASURED PERFORMANCE VALUES TO THE MINIMUM #####
##### AND MAXIMUM PERFORMANCE VALUES OF THE ROCKET DATABASE. #####
#####
##### THE FIFTH MENU SELECTION CAUSES ADVICE TO BE PROVIDED #####
##### TO EXPLAIN WHY THE MEASURED VS EXPECTED PERFORMANCE #####
##### RATIOS ARE NOT EQUAL TO ONE. SUGGESTIONS FOR IMPROVING #####
##### PERFORMANCE ARE PROVIDED AS WELL. #####
#####
##### THE SIXTH MENU SELECTION CAUSES A DATA-BASE OPERATIONS #####
##### MENU TO BE DISPLAYED. FROM THIS, CHOICES MAY BE MADE #####
##### TO ADD THE CURRENT DATA SET TO THE ROCKET DATABASE, #####
##### DELETE A SELECTED DATA SET FROM THE DATABASE, OR REVIEW #####
##### THE DATA SETS CURRENTLY IN THE DATABASE. #####
#####
##### CLICK HERE TO GO BACK TO THE MAIN MENU: RETURN #####
#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONSD###
#####

```

Figure 5B. Operations Summary-Page 2

There are two screens of information so a light-blue "PAGE2" appears at the bottom of the first screen. Clicking this with the mouse replaces the first page with the hypertext of the next page. To return to the main menu, another light-blue word labeled "RETURN" is provided. Clicking on this term replaces the text with the main menu display once again.

More detailed descriptions of the first, fifth, and sixth menu lines are available when needed, as indicated by the light-blue color of each of their first words. Clicking on these words with the mouse causes the menu to be replaced on the screen by the hypertext explanation of the selected option's function. The first option, "UPDATE ROCKET TEST PARAMETER FILE," has the hypertext explanation found in figure 5C. The fifth option, "PERFORMANCE OPTIMIZATION ADVICE," has the hypertext explanation found in figure 5D. The sixth option, "ROCKET ENGINE LIFE-HISTORY DATABASE OPERATIONS," has the hypertext explanation found in figure 5E. All options are selected by clicking the mouse on the color square next to the one desired. The hypertext capabilities just discussed may be utilized anytime the user is in the main menu. After completing all hypertext examinations desired for the moment, the user is ready to select an option.

2.3.2 Access of the ASCII Input File

Figure 5C. Options One Explanation

```

#####
#####
#####
#####
##### THIS MENU OPTION PROVIDES ADVICE AS TO WHICH TEST #####
##### PARAMETERS TO CHANGE TO IMPROVE PERFORMANCE OF THE #####
##### ROCKET. THESE RECOMMENDATIONS ARE BASED ON INFORMATION #####
##### GATHERED ON THE MOST IMPORTANT TEST PARAMETERS TO #####
##### CONSIDER WHEN OPTIMIZING PERFORMANCE. BEFORE ANY #####
##### ADVICE IS GIVEN, THE PARAMETERS IT WILL AFFECT ARE #####
##### CHECKED FOR NEARNESS TO THEIR SAFETY LIMITS AND THE #####
##### VALIDITY OF ANY INPUT CHANNELS INVOLVED IS CONSIDERED #####
##### AS WELL. #####
#####
#####
#####
#####
#####
#####
##### PRESS ANY KEY TO RETURN TO THE MAIN MENU #####
#####
#####
#####
#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION#####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS#####
#####

```

Figure 5D. Options Five Explanation

[illegible]

Figure 5E, Options Six Explanation

As soon as all user-defined inputs are completed, the rocket test is begun and the elapsed-time-in-test clock is started. Data immediately starts pouring into the ASCII file. To access this file and monitor the ongoing rocket test, the user should select the "UPDATE ROCKET TEST PARAMETER FILE" option from the main menu by clicking on the colored box next to it. This causes the screen to clear and a set of twenty-seven measured values from the test to be read into the expert system. Also nine additional parameters are calculated from these measurements. These thirty-six parameters are listed in figure 6 and include those important for performance evaluation as well as values only for developing a life history of a liquid-fueled rocket engine to be stored in the rocket database.

As was mentioned earlier, each company, agency, etc. operating the system will have a slightly different arrangement of information collection. As a result, it is assumed that they have handled the necessary manipulation of their output files to provide an ASCII file containing only the numerical values listed in figure 6, one value per channel, and in that order. This format must be adhered to for the expert system to receive the appropriate information.

2.3.3 Performance Parameter Calculation from Measured Values

Measured

Environmental Temperature
 Gas Generator Pressure
 Gas Generator Temperature
 Turbine Exit Static Pressure
 Turbopump Speed
 Turbopump Lubricant Consumption
 Fuel Pump Inlet Pressure
 Fuel Pump Inlet Temperature
 Fuel Pump Outlet Pressure
 Fuel Pump Outlet Temperature
 Oxidizer Pump Inlet Pressure
 Oxidizer Pump Inlet Temperature
 Oxidizer Pump Outlet Pressure
 Oxidizer pump Outlet Temperature
 Chamber Injector Pressure
 Chamber Temperature
 Coolant Flow Rate
 Thrust Chamber Fuel Flow
 Thrust Chamber Oxidizer Flow
 Gas Generator Fuel Flow
 Gas Generator Oxidizer Flow
 Nozzle Static Exit Pressure
 Thrust
 Fuel Quantity
 Oxidizer Quantity

Calculated

Chamber Pressure
 O/F Ratio
 Exhaust Velocity
 Thrust Coefficient
 Specific Impulse
 Total Propellant Flow
 Effective Exhaust Velocity
 Characteristic Velocity

Figure 6. ASCII File Test Measurement Parameters

Once the values have been read into the expert system, some are used to calculate the total propellant weight flow rate (See Appendix A, Equation 9A or 9B), the total propellant mass flow rate (See Appendix A, Equation 10), the measured oxidizer-fuel flow ratio (See Appendix A, Equation 11), the measured specific impulse (See Appendix A, Equation 12), the measured effective exhaust velocity (See Appendix A, Equation 13), the thrust chamber nozzle stagnation pressure (See Appendix A, Equation 14 and 15), the measured thrust coefficient (See Appendix A, Equation 17), and the measured characteristic exhaust velocity (See Appendix A, Equation 18). All measured and calculated values are stored in an archive file that may later be retrieved by the current time when it was stored.

The thrust chamber nozzle stagnation pressure is calculated from the pressure of the injector at the other end of the thrust chamber using equation 15 in Appendix A. Since this is an important parameter in performance optimization, great care was taken in its calculation. It could be assumed that its value is approximately that of the injector pressure, but instead the calculation is made under the worst case assumption, minimum difference between the two pressures, so as to make the safety limit check very realistic. This is important due to the

fact that its degree of accuracy influences the degree of accuracy of the optimization advice. This influence will be discussed in detail in chapter three.

2.3.4 Comparing Operational Performance Values to Safety Limits

Next, the values are compared to their safety limits, when applicable, and displayed one by one after each comparison. Whenever a parameter value exceeds its safety limit, it is displayed in red, an asterisk is displayed beside it, to mark out-of-limits parameters if a hard copy is desired, the limiting value is displayed in red at the bottom of the screen, and the user is given the option to ignore further readings from this channel if it is considered to be returning bad data. If a data channel is chosen to be ignored, it will no longer be displayed in red or marked in any way if it happens to again exceed its safety limit. After the ignore-the-channel inquiry has been answered, the next parameter value is checked against its safety value and displayed in the appropriate fashion. When the calculated parameter values are displayed, their values and the nominal values are shown side by side for easy comparison. The format of the final display may be seen in figure 7A. These nominal values are those expected performance values

PRESS, PRINT SCREEN? FOR HAND COPY

Figure 7A. Monitor Display With Warning Flags

calculated before the test was begun. The current time displayed is obtained internally from the computer itself. The elapsed time is obtained by taking the current time and subtracting the time stored when the elapsed time was first calculated.

Once all parameters have been displayed, the ability to obtain a hard copy by pressing the "print screen" key is noted and the option to continue monitoring the test is given to the user. If monitoring is continued, the ASCII file is read again, retrieving the latest parameter values, and the process just described is repeated. If monitoring is discontinued, the user is returned to the main menu and may return to monitoring at any time by again selecting the first menu option. A post-test analysis of the test should not occur, however, until the completion of the test.

2.3.5 Archive File Retrieval

After completion of the test the user has a variety of options to choose from the main menu. The second option is labeled "DISPLAY CURRENT ROCKET TEST PARAMETER VALUES" and allows the user to retrieve archive files stored at various times during the test. Clicking on the colored box next to this line causes the main menu to be replaced by

a listing of archive files in order of time stored. The user then decides which file to examine and the parameter information is then displayed on the screen in the format of figure 7B. Any recommendations made from a performance analysis for the selected file are included in this display as well. This is due to the fact that this option may be utilized during the test as well as in the post-test analysis. Each archive file is updated anytime recommendations for improving performance are made and this information may be retrieved along with the numerical parameter values using this option. At the bottom of the screen, the user is reminded that a hard copy may be obtained by pressing the "print screen" key. Return to the main menu is achieved by pressing any key.

2.3.6 Comparing Operational Performance Values to Expected Values

The third option available from the main menu is labeled "COMPARE TEST PARAMETERS TO EXPECTED PERFORMANCE VALUES." Clicking on the colored square next to this line causes the expected and measured performance parameters of a selected archive file to be compared. This file is chosen from an archive file list in order of time stored. Dividing the measured by the expected values for thrust, specific impulse, effective exhaust velocity, thrust coefficient,

CURRENT ROCKET ENGINE TEST DATA

STORAGE TIME:	02:24:57	TIME ELAPSED (SEC):	908.1
ENV. TEMP. (DEG. F):	52	COOLANT FLOW RATE (LB/S):	725.68
GG PRESSURE (PSIA):	671.49	TC FUEL FLOW (LB/S):	267.39
GG TEMPERATURE (DEG. F):	900	TC OXID FLOW (LB/S):	820.25
VB STATIC PRESS. (PSIA):	34.96	GG FUEL FLOW (LB/S):	10.21
TP SPEED (RPM):	4100	GG OXID FLOW (LB/S):	4.39
TP LUB. CONSUM. (GPM):	5	NOI EX STAT PRESS. (PSIA):	14.5
FP INLET PRESS. (PSIA):	57	MEASURED THRUST (LB):	209700
FP INLET TEMP. (DEG. F):	60	FUEL QUANTITY (LB):	40000
FP OUTLET PRESS. (PSIA):	256	OXID QUANTITY (LB):	90000
FP OUTLET TEMP. (DEG. F):	80	TC NOI STAG PRESS. (PSIA):	605.86
OP INLET PRESS. (PSIA):	63	OPF RATIO:	2.02
OP INLET TEMP. (DEG. F):	-290	TOTAL FLOW RATE (LB/S):	990.31
OP OUTLET PRESS. (PSIA):	310	SPECIFIC IMPULSE (SEC):	235.40
OP OUTLET TEMP. (DEG. F):	-285	EFF. EXH. VEL. (FT/S):	7579.21
TC INJECTOR PRESS. (PSIA):	653.03	THRUST COEFF:	1.93
TC TEMPERATURE (DEG. F):	2000	CHAR. EXH. VEL. (FT/S):	4941.16

RECOMMENDATIONS SINCE START OF TEST:

PRESS 'PRINT SCREEN' FOR HARD COPY

PRESS ANY KEY TO RETURN TO THE MAIN MENU

Figure 7B. Retrived Archive Display

characteristic velocity, oxidizer-to-fuel ratio, and propellant weight flow rate provides seven efficiencies. The first six should typically fall between 0.85 and 0.99 [4]. The seventh efficiency should typically fall between 1.01 and 1.18, since more propellant is always required than in the ideal expectation. The measured and expected values are displayed for the parameters along with their efficiencies in the format seen in figure 8. The contrast between theory and measured values for performance parameters can clearly be seen in this type of format. A hard copy of this comparison may be obtained by pressing the "print screen" key. The user is returned to the main menu by pressing any key.

2.3.7 Comparing Operational Performance Values to Database Extremes

The fourth option in the main menu is labeled "COMPARE TEST PARAMETERS TO ROCKET ENGINE DATABASE." This selection provides a comparison of selected archive file parameters to the minimum and maximum performance values in the database to determine if the new test values fall in the typical range of past test cases. Clicking on the colored square next to this line causes the menu to be replaced by a archive file list in order of time stored. The user then selects the file to retrieve including a table of thrust, specific impulse, oxidizer-to-fuel

CURRENT ROCKET ENGINE TEST DATA

STORAGE TIME:	02:24:57	TIME ELAPSED (SEC):	508.1
ENV. TEMP. (DEG. F):	62	COOLANT FLOW RATE (LB/S):	325.88
GG PRESSURE (PSIA):	671.49	TC FUEL FLOW (LB/S):	297.59
GG TEMPERATURE (DEG. F):	900	TC OXID FLOW (LB/S):	623.23
YB STATIC PRESS. (PSIA):	34.96	GG FUEL FLOW (LB/S):	10.21
TP SPEED (RPM):	4100	GG OXID FLOW (LB/S):	4.39
TP LUB. CONSUM. (GPM):	5	NOI EX STAT PRESS. (PSIA):	14.5
FP INLET PRESS. (PSIA):	57	MEASURED THRUST (LB):	209700
FP INLET TEMP. (DEG. F):	60	FUEL QUANTITY (LB):	40000
FP OUTLET PRESS. (PSIA):	256	OXID QUANTITY (LB):	90000
FP OUTLET TEMP. (DEG. F):	80	TC NOI STAG PRESS. (PSIA):	635.86
OP INLET PRESS. (PSIA):	63	C/F RATIO:	0.32
OP INLET TEMP. (DEG. F):	-290	TOTAL FLOW RATE (LB/S):	990.81
OP OUTLET PRESS. (PSIA):	310	SPECIFIC IMPULSE (SEC):	233.40
OP OUTLET TEMP. (DEG. F):	-285	EFF. EXH. VEL. (FT/S):	7579.91
TC INJECTOR PRESS. (PSIA):	653.03	THRUST COEFF:	1.50
TC TEMPERATURE (DEG. F):	2000	CHAR. EXH. VEL. (FT/S):	4941.16

RECOMMENDATIONS SINCE START OF TEST:

PRESS 'PRINT SCREEN' FOR HARD COPY

PRESS ANY KEY TO RETURN TO THE MAIN MENU

Figure 8. Data Display Menu Option Output

ratio, fuel weight flow rate, oxidizer weight flow rate, and total propellant weight flow rate values for the archive file and the database minimums and maximums. The last column of the table displays an asterisk for each archive file test value which is outside the range formed by the database minimum and maximum. Figure 9 shows the typical format.

2.3.8 Performance Analysis

The fifth main menu option is labeled, "PERFORMANCE OPTIMIZATION ADVICE." This portion of the expert system is so important that the entire next chapter is devoted to its description. As a result, it will not be discussed here.

2.3.9 Database Updating

The sixth option available in the main menu is labeled "ROCKET ENGINE LIFE-HISTORY DATABASE OPERATIONS" and allows the user to interact directly with the life-history rocket engine database. If the user selects this option, the main menu is replaced again with a list of archive files in order of time stored. The user then selects the file to be added to the rocket engine database, if so desired, while in this portion of the expert system.

All operations in this succession are controlled from the menu, figure 10

PARAMETER: CHARACTERISTIC EXHAUST VELOCITY

VALUES: EXPECTED CHARACTERISTIC EXHAUST VELOCITY (FT/S): 5094.8
 MEASURED CHARACTERISTIC EXHAUST VELOCITY (FT/S): 4911.16
 MEASURED/EXPECTED RATIO: 0.91

ADVICE: THIS RATIO IS AN INDICATION OF COMBUSTION AND INJECTOR PERFORMANCE. IT IS LOWER THAN THE USER-DEFINED ACCEPTABLE PERCENTAGE WHICH IMPLIES POOR COMBUSTION AND POOR INJECTOR PERFORMANCE. TO INCREASE THE CHARACTERISTIC EXHAUST VELOCITY, A GREATER PROPELLANT MASS FLOW RATE MUST BE USED TO INCREASE THE INJECTOR PRESSURE AND THUS THE NOZZLE STAGNATION PRESSURE. LIMITS THAT NEED TO BE CONSIDERED WHEN DOING THIS ARE THE INJECTOR PRESSURE LIMIT, THE NOZZLE STAGNATION PRESSURE LIMIT, AND THE THRUST CHAMBER TEMPERATURE LIMIT. THE LAST CONSIDERATION MUST BE MADE DUE TO THE FACT THAT AN INCREASE IN THE CHAMBER PRESSURE WILL BRING ABOUT AN INCREASE IN CHAMBER TEMPERATURE. AS A RESULT, THE VALUES OF THE MENTIONED PARAMETERS AND THEIR LIMITS ARE DISPLAYED BELOW.

NOTE: THRUST CHAMBER NOZZLE STAGNATION PRESSURE: 502.86
 THRUST CHAMBER PRESSURE LIMIT: 1000
 THRUST CHAMBER INJECTOR PRESSURE: 653.03
 THRUST CHAMBER INJECTOR PRESSURE LIMIT: 1200
 THRUST CHAMBER TEMPERATURE: 2000
 THRUST CHAMBER TEMPERATURE LIMIT: 2500
 PRESS ANY KEY TO CONTINUE

Figure 9. Performance Value Comparison

[illegible]

Figure 10. Database Main Menu

and the display window, figure 11. From the display window, the entire list of cases stored in the database file may be examined and any one case selected for more detailed information. Also, as a quick reference, the name of each case is a hyperword linked to hypertext displayed when the name is clicked on with the mouse. This hypertext is the six lines of brief comments about each test allowed. As a result, the user can scroll through the case list and click on any name for quick comments. If more information is desired, then selecting the case number beside its name causes more information to be displayed.

After selection of an archive file for consideration, the database operations main menu is displayed (see figure 10). If the user wishes to add the previously selected archive file, complete with numerical and performance analysis recommendation information, to the database, the colored box by the line "ADD NEW ROCKET ENGINE TEST CASES TO DATABASE" is clicked on with the mouse. When adding this case, the expert system checks the new values against the database's minimum and maximum values and makes them the new minimum and maximum values information is then asked for along with six possible lines of summary comments about the test.

ENGINE NUMBER ONE
ENGINE NUMBER TWO
H.C. TEST NUMBER ONE

11
12
13
14
15
16
17
18
19
20

CLICK ENG NAME AND SEL CASE EDITOR FOR TEXT AND NUM INFO

FIRST PAGE
PREVIOUS PAGE
NEXT PAGE
CURRENT PAGE
LAST PAGE
SELECT CASE NUMBER
EXIT

Figure 11. Database Display Window

If the user desires to delete a rocket engine from the database, selection of the "DELETE ROCKET ENGINE TEST CASES FROM DATABASE" from the database operations menu is made. Once this choice is made, the display window (figure 11) provides a listing of all engines in the database. The user then selects the "Select Case Number" button on the side of the screen and types the number of the case to delete. The chosen test case will be erased from the database file and new minimum and maximum performance parameter values are calculated. The user is then returned to the database operations main menu.

Selecting the "DISPLAY ROCKET ENGINE TEST CASES IN DATABASE" option from the database operations menu, causes the display window to appear (see figure 11). The user may now click on any engine name for summary comments about the test or click the "Select Case Number" button and type the case number to retrieve all numerical information, summary comments, and recommendations for improving performance made. Clicking the "Exit" button at the side of the screen returns the user to the database operations menu. Clicking the colored box by the "RETURN TO ROCKET ENGINE MAIN MENU OPERATIONS" option causes the main expert system menu to reappear.

The structure of the two database files of greatest importance can be seen in figures 12 and 13. Figure 12 is the engine historical database file. Figure 13 is the minimum and maximum performance parameter database file. As was mentioned earlier, these were created using Lotus 123 and dBase III+, but are completely accessible and usable to the expert system. Neither one of these pieces of software need ever be used again in relation to these database files. The expert system will handle all use of them.

Figure 12. Case Info.DBF File Structure

Case
Engine
Htxt1
Htxt2
Htxt3
Htxt4
Htxt5
Htxt6
Engine_Name
Envtemp
Ggpress
Ggtemp
Tbpress
Tpspeed
Tplubcons
Fpipress
Fpitemp
Fpopress
Fpotemp
Opipress
Opitemp
Opopress
Opotemp
Pcinj
Tc
Coolfrate
Tcff
Tcof
Ggff
Ggof
Pem
Fmeas

Engine Thrust Spimpulse Ofratio Fuel_Flow Oxid_Flow Total_Flow

Mn. Values

Mx. Values

Figure 13. Extremes.DBF File Structure

CHAPTER III

ROCKET ENGINE PERFORMANCE ANALYSIS

3.1 Important Parameters and Their Influence on Rocket Performance

Intricate machines of high technology, rockets are developed to deliver payloads to desired points with a specific velocity at the time of delivery. Rocket propulsion system operation is based on energy conversion principles with the typical rocket converting some stored form of energy such as chemical energy, nuclear energy, or electrical energy to another form of energy called kinetic energy of the propellant flow. This conversion process generates a force called the rocket thrust which propels the rocket vehicle. Analysis of the energy conversion process is therefore the basis for identifying true rocket performance standards [4]. Such an analysis is provided in Appendix A along with the working relationships for ideal rocket performance analysis. From these equations and references [2], [4], [5], [6], and [7] the important parameters influencing the performance of a liquid-fueled rocket engine were determined, and the expected performance of each engine being tested was found to be calculated by using eqns 1-8 in Appendix A.

3.1.1 Specific Impulse

Equation 1 in Appendix A clearly reveals specific impulse to be the key parameter for performance optimization. The higher the I_{sp} the less weight flow of propellant is required to produce the same thrust. The engine therefore becomes increasingly efficient with increasing I_{sp} and performance improves. The maximum I_{sp} that may be achieved for a given engine occurs when thrust is maximum. This can be seen in equation 12 of Appendix A. Once maximum thrust has been reached, increasing the propellant weight flow only decreases the specific impulse.

3.1.2 Nozzle Exit Static Pressure

From equation 19 of Appendix A it may be determined that, for a given altitude, maximum thrust occurs when the nozzle is designed such that its exit static pressure equals the ambient static pressure. With this condition the nozzle is referred to as "perfectly expanded" for the ambient pressure. The nozzle exit static pressure depends on the nozzle expansion ratio and on the chamber pressure. Assuming that the chamber pressure is sufficiently high, as the nozzle expansion ratio increases, the velocity of the exhaust gases increases, increasing the

momentum thrust (first term on the right side of the equation 19).

However, as the nozzle expansion ratio increases, the viscous drag forces on the nozzle walls also increase. A trade-off must therefore be undertaken in nozzle design so that the thrust increases due to higher nozzle area ratio, minus the thrust loss due to viscous drag, reaches a maximum.

When P_e is greater than P_a the flow is considered to be underexpanded and the nozzle expansion ratio needs to be increased to lower the P_e . When this condition occurs not all of the thermal energy of the flow is being converted to the maximum possible kinetic energy for the given ambient pressure, thus causing a waste of energy and a decrease in engine efficiency. When P_e is less than P_a the flow is considered overexpanded. Separation usually occurs inside the nozzle, when P_e is significantly less than P_a . It tends to occur at the area ratio where the pressure is approximately equal to one-half the ambient static pressure. When this condition occurs, the nozzle expansion ratio is too large for the engine chamber pressure, and, if operation is to be continued, the chamber pressure must be increased until separation is driven out of the nozzle.

3.1.3 Thrust Coefficient

Specific impulse may be expressed in several forms, presented in Appendix A. In one form, equation 23, it is a function of the thrust coefficient, C_f , and the characteristic velocity, c^* . The fact that the thrust of a rocket is significantly higher than that which would be produced by simple pressure imbalance due to the presence of the throat in the aft end of the motor leads to the concept of a thrust coefficient, which is, in a sense, a measure of the effect of the nozzle. The pressure imbalance itself would result in a force which was equal to the product of the chamber pressure and the area of the throat. By factoring this product out of the thrust equation (See Appendix A, equation 19), an expression for the theoretical thrust coefficient is obtained [5].

Basically the quantity C_f reflects the results of the expansion of the exhaust gas, and its properties, and the design quality of the nozzle. As can be seen by equation 27 of Appendix A, it is a function of the specific heat ratio, thrust chamber pressure, ambient pressure, and the nozzle expansion ratio. The specific heat ratio is determined by the user from thermo-chemical calculations, e.g. [12], based on the propellant used. A

smaller value of specific heat ratio gives a higher exhaust velocity for a given nozzle area ratio, hence, smaller exhaust gas specific heat ratio yields a higher value for C_f . Keeping this in mind, rockets are designed to produce the highest value for the ratio of the chamber temperature to propellant molecular weight for optimum performance.

As for the effect of nozzle expansion ratio and ambient pressure on C_f , the same conditions apply as mentioned earlier. The nozzle expansion ratio should be as high as possible, subject to the limitations previously discussed. The maximum value of C_f will occur when this condition is met.

3.1.4 Thrust Chamber Pressure

The thrust chamber pressure is an extremely important parameter, as has been shown previously. As can be seen by equation 27 of Appendix A, the effective chamber pressure or nozzle stagnation pressure $(P_c)_{ns}$ appears in two pressure ratios $P_e/(P_c)_{ns}$ and $P_a/(P_c)_{ns}$. The ratio $P_e/(P_c)_{ns}$ has one value for a given nozzle expansion ratio and specific heat ratio. $(P_c)_{ns}$ therefore influences C_f only through the negative term $-P_a/(P_c)_{ns}$. An increase in $(P_c)_{ns}$ decreases this negative

term and hence increases C_f . This effect is more pronounced when P_a is high. Since the thrust is proportional to both $(P_c)_{ns}$ and C_f , it is clear how an increase in $(P_c)_{ns}$ in a given thrust chamber will increase the thrust [6]. Also as $(P_c)_{ns}$ increases for a given thrust, the nozzle and rocket motor size decrease. $(P_c)_{ns}$ therefore sizes the engine physically, as well.

3.1.5 Characteristic Exhaust Velocity

The characteristic velocity, c^* , is primarily used to rate the propellant combustion performance, and it reflects the effective energy level of the propellants and the design quality of the injector and combustion chamber. From equation 24 of Appendix A, it is clear that c^* measures combustion performance in a given combustion chamber by indicating how many pounds per second of propellant must be burned to maintain the required nozzle stagnation pressure. A lower value of propellant consumption under the given conditions indicates a combustion process of higher energy and efficiency and gives a corresponding higher value of c^* [6].

Also from equation 24 of Appendix A, it becomes apparent that c^* is

a function of the properties of the combustion product gas at the exit of the combustion chamber, i.e., at the nozzle inlet; namely, specific heat ratio, gas constant R , and the nozzle inlet stagnation temperature $(T_c)_{ns}$.

As mentioned before, the highest possible ratio of chamber temperature to propellant molecular weight is desired. Also for constant $(T_c)_{ns}$, c^* will increase if R increases; i.e., the gas molecular weight decreases.

Thus, a higher value of R will yield a higher engine performance.

Choosing a propellant with the largest possible gas constant and ratio of chamber temperature to propellant molecular weight provides the best performance [6].

This leaves the $(P_c)_{ns}$ and $(T_c)_{ns}$ parameters to consider. In the c^* equation (see Appendix A, equation 24), clearly an increase in $(P_c)_{ns}$ causes an increase in c^* , for a constant propellant mass flow rate. Such an increase in $(P_c)_{ns}$ thus means that $(T_c)_{ns}$ must increase. As can be seen in equation 24, this causes an overall increase in c^* . These effects are somewhat smaller at $(P_c)_{ns}$ above 300 psi, but the overall increase still occurs [6]. As a result, the expert system checks the values of $(P_c)_{ns}$, by converting the measured value of P_{inj} as before [8], and

$(T_c)_{ns}$ against safety values and suggests an increase in $(P_c)_{ns}$ if possible and safe.

3.1.6 Oxidizer to Fuel Flow Rate Ratio

Another way to improve a liquid-fueled rocket engine's performance is to improve the combustion process taking place within the thrust chamber. Complete combustion of a given amount of fuel requires corresponding amount of oxidizer that depends on the molecular composition of the fuel. That mixture ratio which effects complete combustion, with no leftover of either fuel or oxidizer, is called the stoichiometric mixture ratio. This ratio depends on the type of propellants used, and theoretical temperature and heat release are usually maximum at or very near to this ratio. In rocket engines, however, where the highest possible exhaust velocity is desired, optimum conditions often prevail at other than stoichiometric ratios. The gas properties of molecular weight and gas constant strongly affect exhaust velocity. As was discussed earlier and can be seen in equations 20 and 26 of Appendix A, the lower the molecular weight, the higher the exhaust velocity, other things being equal. Operating the rocket engine with a variety of O/F ratios near the computed optimum values can

make clear the optimum point of balance between energy release (heat) and composition (molecular weight) of the gas, a portion of which will consist of gasified but unburnt propellants ([6] & [7]) if the propellant combination is sub-stoichiometric.

As a rule, the optimum mixture ratio is richer in fuel than the stoichiometric mixture ratio. Also the optimum mixture ratio of some propellant combinations shifts slightly with changes in chamber pressure so it is important to hold this value constant when determining the ratio. Also, in actual application the mixture ratio may have to be shifted away from the optimum value to keep the chamber temperature within safety limits with the indicated coolant flow ([6] & [7]).

3.1.7 Fuel and Oxidizer Pump Inlet and Outlet Pressures

The final parameters considered important in their effects on rocket engine performance are fuel and oxidizer pump inlet and outlet pressures. The propellant flow rates and the inlet and outlet pressures of the fuel and oxidizer pumps can be used to measure the power consumption of the pumps. As a result, an increase in the ratio of the outlet to the inlet pressure for a given flow rate results in an increase in the pump input power. A decrease in the ratio of pump outlet and inlet pressures for a given flow rate represents a reduction in pump power

requirements. This can be seen by examining equations 9a, 9b, and 12 in Appendix A. Moreover, decrease causes the thrust chamber flow rate to decrease resulting in an overall total propellant flow rate decrease (equations 9a and 9b). This in turn increases the specific impulse (equation 12) and therefore performance. Increased specific impulse also increases the characteristic exhaust velocity (see equation 18 of Appendix A) and therefore improves the combustion efficiency of the rocket [6].

3.2 Performance Analysis by the Expert System

The performance parameters discussed in the previous section were determined to be those having the greatest influence on the performance of a liquid-fueled rocket engine. The expert system executes a post-test performance analysis of a rocket engine by comparing the expected to measured values of the key performance parameters that can be modified at this stage and provides advice as to changes that may be made to improve the rocket's performance. These include specific impulse, thrust coefficient, characteristic exhaust velocity, thrust chamber nozzle stagnation pressure, and oxidizer to fuel flow ratio. The nozzle exit static pressure is not considered since this is a

design parameter that may not be changed at this point without redesigning the rocket.

First, the user selects the "PERFORMANCE OPTIMIZATION ADVICE" option from the expert system main menu, by clicking the mouse on the colored box next to this line. Next, a list of the archive files for the current test are provided in order of time stored. After the user has selected the file to be analyzed, the archive's information is recalled along with the expected performance values for the rocket and the user-defined safety limits. Then ratios of the measured to expected key performance parameters are calculated and the user is asked to provide a lower limit of acceptability for these all ratios except that of the measured to expected oxidizer to fuel ratio. Since this ratio could exceed the expected value, the limit of acceptability is considered to be a minimum and maximum. The default value provided is 92%.

The first ratio considered is for specific impulse. If the value of the ratio falls below the limit, the expert system determines that advice for improving this parameter is needed. At the top of the screen the parameter name is provided, in this case specific impulse, the expected and measured values, and the measured to expected ratio are displayed,

and the user is informed that the specific impulse is not at its optimum value. The fact that the characteristic exhaust velocity and thrust coefficient must be optimized for maximum specific impulse is provided as advice, and their analysis is performed next. If the ratio value is above the limit then the parameter name, its expected and measured value, and the measured to expected ratio are displayed along with the advice that the specific impulse is optimized under the current conditions.

As mentioned above, if the ratio of measured to expected specific impulse falls below the acceptable minimum, the thrust coefficient and characteristic exhaust velocity are analyzed next by the expert system. First, the thrust coefficient is considered. For this parameter, there are three possible forms of advice. The first occurs when the following conditions are met: the measured to expected ratio is below acceptable, the thrust chamber injector pressure is within nominal values, the calculated nozzle stagnation pressure is also within nominal values, and the thrust chamber temperature is within nominal limits. The advice given is that the nozzle is not performing as expected and should be redesigned, since thrust coefficient is an indicator of nozzle performance,

as mentioned earlier, and that a greater propellant mass flow rate should be utilized to increase injector pressure and thus the chamber pressure. The pressures and their limits are displayed to give the user an idea of the degree of increase possible without exceeding limits. As was discussed earlier, this will increase the thrust coefficient. An alternate piece of advice is given when the thrust coefficient comparison ratio is less than acceptable and the other parameters, injector pressure, chamber pressure, and/or chamber temperature, are less than the nominal acceptable values. In this case, nozzle redesign is only an option, and this information is then given; the poor performance is mainly attributed to poor thrust chamber performance, and this advice is given. The third piece of advice is given when the comparison ratio is acceptable. In this case, the user is informed that under the current conditions, the thrust coefficient is optimized. All advice given to the user is stored in the retrieved archive file for later recall or inclusion into the database.

Next, the characteristic exhaust velocity is considered. For this parameter, there are three possible forms of advice. The first occurs when the following conditions are met: the measured to expected ratio is below acceptable, the thrust chamber injector pressure is less than the

limit, the calculated nozzle stagnation pressure is less than its limit, and the thrust chamber temperature is less than its limit. The advice given is that poor combustion and injector performance is the probable cause of low measured characteristic exhaust velocity. A greater propellant mass flow rate would have to be utilized to increase injector pressure, and thus the chamber pressure, and hence, the thrust. Note that the specific impulse would not be improved if this correction to low characteristic exhaust velocity was used by the engineer. The pressures and their limits are displayed to give the user an idea of the degree of increases possible without exceeding limits. As was discussed earlier, unless the injector/combustion performance is improved, measured characteristic exhaust velocity and specific impulse will not improve, unless combustion automatically improves when the propellant flow rate through the engine improves. This is what can occur with throttlable rocket engines such as the Apollo lunar decent engine. The second piece of advice is given if only the characteristic exhaust velocity comparison ratio is less than acceptable and none of the other conditions of the first advice constraints have been met. In this case, improvement of the injector and combustion performance of the rocket is the only option,

and this information is then given. The third piece of advice is given when the comparison ratio is acceptable. In this case, the user is informed that under the current conditions, the characteristic exhaust velocity is optimized. All advice given to the user is stored in the retrieved archive file for later recall or inclusion into the database.

The next parameter examined is the measured to expected propellant oxidizer to fuel ratios ratio. If this value fall below expected, then the chamber temperature, fuel pump inlet pressure, and oxidizer inlet pressure are checked against their limiting values. If they have not reached their values yet, then the advice is given to decrease fuel pump inlet and outlet pressures and/or increase oxidizer pump inlet and outlet pressures. This will, as was discussed earlier in this chapter, increase the oxidizer to fuel ratio as desired to improve performance. If the important values are at their limits, then no recommendations are possible for improving this parameter, and this is information is provided to the user. All advice given to the user is stored in the retrieved archive file for later recall.

By considering the safety limitations and their effects on the other parameters, the expert system can provide the necessary advice to the user for optimizing the performance of the rocket in the next test

conducted. From this part of the expert system, the user is returned to the main menu, after the recommendations and current parameter values, important for developing a life history of the engine, are saved in an archive file. These important parameters are listed in figure 6. This file may now be included in the database if desired.

This process may be repeated with any of the archive files. Once the user has taken the advice under consideration and made the appropriate changes, a new test may be initiated and a new post-test performance analysis completed. In this way, the expert system guides the user to the optimum performance for a given rocket design. By examination of the recommendations in the archive files included in the database, the path towards optimization of the performance of a given rocket may be followed.

4.1 Research Summary

The purpose of the research presented in this thesis was to create a prototype expert system to assist in three areas of liquid-fueled rocket testing. These areas include: 1) monitoring an operating test and providing immediate warnings when safety limits are exceeded; 2) initiating a post-test performance optimization analysis; and 3) building a life-history of selected rockets tested. All three modules were completed successfully, and Appendix B demonstrates the full capabilities of the system by using a simulated rocket test based on representational data from a set of actual rocket engine tests, to develop and to illustrate the capabilities of the expert system.

Preparing this demonstration suggested a fourth area of application for the expert system as well; it can be used to train personnel to monitor rocket tests. By providing carefully prepared data files for a simulation, or real, past rocket test data files, personnel can be trained to watch for trends in data that may lead to problems in the test and to handle such problems when they occur. This makes the expert system a very valuable training tool as well.

In developing the monitoring portion of the expert system, a study of the requirements for displaying rocket test data, in a way most useful to a test engineer, was made. These requirements were determined to be the following: 1) Clear, easy-to-read-at-a-glance data presentation; 2) Clearly flagged safety limit violations; 3) Easy comparison to nominal values for important parameters; 4) Quick retrieval of a hard copy of displayed information at any time; 5) Option to ignore bad channels to avoid unnecessary and distracting flagging of values. These requirements have all been met by this expert system. All data is presented on the screen in two columns of variable names and numerical values. Any parameters exceeding their safety limits are displayed in red, flagged by an asterisk, and may be ignored if bad data is being returned by the channel. Nominal values for the performance parameters are displayed in a separate column next to the corresponding measured values. Finally, a hard copy of displayed information may be obtained at any time by pressing the "print screen" key. It should be noted that the asterisk marking, for values violating safety limits, is utilized purely for hard copy purposes, since a differently colored display will not appear on paper. An additional

feature of importance is the storage of all parameter values in archive files, referenced by the points in time they were stored, for later retrieval and analysis.

4.1.2 Performance Analysis Module

In developing the post-test performance optimization analysis portion of the expert system, a study was made of the most important parameters to consider when optimizing performance and the other parameters that influence them. The three most important parameters were determined to be specific impulse, thrust chamber pressure, and oxidizer to fuel flow ratio. Parameters influencing specific impulse were found to be thrust coefficient, characteristic exhaust velocity, thrust chamber nozzle stagnation pressure, thrust chamber injector pressure, thrust chamber temperature, and the total propellant flow rate.

Parameters influencing chamber temperature are the propellant types, their flow rates, and the efficiency of the combustion process, primarily.

Parameters influencing the oxidizer to fuel flow ratio were found to be thrust chamber temperature, fuel pump inlet and outlet pressures, and oxidizer pump inlet and outlet pressures. Each of the influencing parameters are compared to their safety limits and advice is then

provided as to changes that may be made to each if possible to improve the primary parameter values and thus the rocket engine's performance. All recommendations made are stored with the parameter values of the archive file being examined for later retrieval and/or storage in the rocket engine life history database.

4.1.3 History Database Module

In developing the rocket engine life history database portion of the expert system, a study was made of the most important parameters to include when constructing a life history for a rocket engine. This was only half of the task, however. The other half was the easy recall and manipulation of the database information. To solve this, the new graphics and hypertext capabilities of the VP-Expert software were utilized fully. All database operations are controlled from a database main menu. When adding to the database, an engine name and test identifier are selected along with up to seven lines of brief comments about the test. The expert system then adds this information to a database file and to a hypertext file.

In the retrieval operation, hypertext was determined to be the best way to allow scrolling through a list of engine names longer than a single

screen will allow. This list is stored in the hypertext file under page numbers and recalled by specifying the page to be examined. Before storage, Basic routines are used to manipulate the page number strings into the appropriate hypertext format. Engine names are then added to the end of a page list until the maximum of twenty is reached. The next engine name to be added will be held until a new page number is inserted. The engine name is then added after the new number. A second hypertext file stores the comments for each test after the corresponding engine names. These names must be manipulated into the correct format using Basic routines before being added to the hypertext file. This is necessary for the recall of associated comments when an engine name is selected from the list.

When deleting from the database, each line of the database file is read into a dummy file except the line corresponding to the deleted case. This dummy file is then copied back into the original and the hypertext files are deleted and recreated by retrieving the necessary information from the new database file. This procedure is repeated for each case deleted.

Selection of the cases for deletion or information recall is provided by displaying the engine name list, with its scrolling options and

requesting the number beside the case name to be manipulated. This number corresponds to the row in the database file. The information in this row is then recalled for display. If deleting the case, the process will then begin. If not, the database operations menu is redisplayed.

Before selection of a case number, the comments associated with each engine name of the list may be recalled by clicking the mouse directly on the name. The engine list will then be replaced by any comments available for the selected engine test. To return to the engine list, the field labeled "Current Page" is clicked with the mouse, and the list is once again displayed. This procedure may be repeated for any engine listed.

This completes the summary of the expert system capabilities and methods utilized in its creation. The result of this research is a working prototype capable of monitoring a rocket test, initiating a post-test performance optimization analysis, and building a life history of the engine. The next section will discuss future work that may be done to expand this prototype to cover other areas as well as increase the depth of the work already completed.

4.2 Future Expansion of the Prototype

4.2.1 Long-Term Expansion

As indicated in the title of this thesis, the developed expert system is a prototype. The ultimate goal of this project would be to complete an expert system capable of actually operating the rocket test. Such an expert system would have the following capabilities: 1) Correction of the appropriate influencing parameter values when others have exceeded their safety limits; 2) Real-time performance optimization analysis and appropriate corrections made to achieve optimum performance of the rocket; 3) Complete diagnostic readout on rocket engine components for any time selected from any rocket test; 4) Recall of trends in operating parameters and performance for past tests of a rocket and comparison to similar rockets; 5) Modification of test parameters by engineers directly to a monitor causing real-time changes in the test. Although these are in the far future, they are certainly very promising future steps for expert systems.

4.2.2 Near Short-Term Improvement

Suggested modifications to the expert system in the much nearer future will now be discussed. The first change that might be made is to allow the user to add their own recommendation files for performance optimization, to be recalled under their defined conditions, if the

recommendations provided in a certain situation are either unsatisfactory or incomplete. This would mean modularizing the performance advice routines so that any files with defined conditions and advice statements, to be displayed when the conditions are met, would be able to "plug" into the expert system. By putting the conditions and advice into files to be recalled, instead of inside the main code of the expert system, the ability to utilize any number of files, and, therefore, any number of recommendations, is given to the expert system. Files may be easily written to and modified where internal code may not be. This expert system is entirely menu-driven. It is suggested that an additions maintain this operations structure. Handling of recommendation file recall and modification would best be accomplished by a operations menu similar to the one used for the database operations and the main control menu

Another possible addition to the expert system is the calculation of mean and standard deviation values for individual rocket tests and collective tests for a given rocket. These values could be utilized in the comparison routines along with the performance comparisons provided currently. The number of standard deviations away from the mean measured values fall into would serve as a further check of their

validity and concurrence with other tests of the same rocket or similar rockets. Just as with performance, these comparisons could be made to expected values and with database values. A data uncertainty analysis of measurement could also be utilized shortly thereafter. The earlier comparisons would serve as a good check of the uncertainty analysis.

An option that may be considered as well is to provide access to more accurate computer codes for calculating the ideal performance of the rocket. This would be a simple matter of adding an inquiry, before expected performance calculation, to determine which code should be used. A menu of the codes available could be provided, and the default could be the ideal, one-dimensional, isentropic equations currently being utilized. Calling the other codes from the expert system is not a problem as long as they have a .EXE, .COM, or .BAT extension. The .EXE extension is preferable.

A larger simulation file base is another possible addition to the expert system. By increasing the number of data files, for the purpose of training personnel to recognize trends in test data leading into problems, the exposure to a variety of incoming data becomes more varied. Also a limited random number generator could be used to vary the data even more for any number of files, as long as the number

generation was limited to a reasonable range for that parameter.

Currently the archive files stored always contain the last three inputs from the test ASCII file. This can be expanded to cause the creation of archive files every so many seconds or minutes. This time interval would best be defined by the user for each test, and this would provide a much more accurate account of parameter values over a reasonably long test time. As a result, more than three sets of test information could then be evaluated, examined, and perhaps included in the database.

These mentioned improvements to the prototype would greatly enhance its capabilities. Many more can be suggested, but these were considered most important at this stage of the expert system's development. This can be an ongoing project for many years with a great deal of potential. It will no doubt evolve greatly in time. This thesis served to provide the base from which all further work may be done.

BIBLIOGRAPHY

BIBLIOGRAPHY

1. " Expert Systems for Experts," Parsage and Chignell, Wiley and Sons, Inc., New York , 1988
2. " H-1 Rocket Engine," NASA CR-70. 024, November 16, 1965, Limited
3. "F-1 Engine Qualification Test Procedure," NASA CR-78.882, July 14, 1966 Limited
4. "Rocket Propulsion Course Notes," University of Tennessee Space Institute, Fall Quarter 1988
5. "Fundamentals of Solid Rocket Motors," Dr. R.B. Kruse, Huntsville, AL, 1988
6. "Design of Liquid Propellant Rocket Engines," NASA SP-125, Huzel and Huang, NASA, Washington, D.C., 1971
7. "Theroetical Bases for the Desgin of Liquid-Fueled Rocket Engines," M.I., Shevelyok, Oborongiz, Moskva, 1960
8. "Rocket Propulsion," Marcel Barr`ere, New York, 1960
9. "A Rocket Engine Design Expert System," NASA-TM-102373, NASA Louis Research Center, 1989
10. "Considerations in Developement of Expert Systems for Real-Time Space Applications," S. Murugesan, Ames Research Center, N89-15610
11. "VP-Expert Version Manual," Paperback Software, 1989
12. "Rocket Propulsion Elements," George P. Sutton, John Wiley and Sons, 1988
13. "Advanced Chemical Rocket Propulsion," Y.M. Timnant, Academic Press, 1987
14. " VP-Expert-2.1," Paperback Software, 1989

APPENDIXES

Appendix A

IDEAL PERFORMANCE PARAMETER EQUATIONS

Expected Performance Calculation Equations

$$(1) \quad \dot{W} = \frac{F}{I_{sp}}$$

$$(2) \quad \dot{m} = \frac{\dot{W}}{g}$$

$$(3) \quad C_{EFF} = \frac{F}{\dot{m}}$$

$$(4) \quad \frac{A_E}{A^*} = \frac{(1 + \frac{\gamma-1}{2} M_E^2)^{\sigma_1}}{M_E (\frac{\gamma+1}{2})^{\sigma_1}}$$

where

$$\sigma_1 = \frac{\gamma + 1}{2(\gamma - 1)}$$

$$(5) \quad P_{c_{max}} = \frac{P_a}{(1 + \frac{\gamma-1}{2} M_E^2)^{-\sigma_2}}$$

where

$$\sigma_2 = \frac{\gamma}{\gamma - 1}$$

$$(6) \quad C_F = C_1 + C_2$$

where

$$C_1 = \frac{\gamma M_E \left(\frac{\gamma+1}{2}\right)^{-\sigma_1}}{\sqrt{1 + \frac{\gamma-1}{2} M_E^2}}$$

and

$$C_2 = \frac{\left(\left(1 + \frac{\gamma-1}{2} M_E^2\right)^{-\sigma_2} - \frac{P_e}{P_e}\right) \cdot \left(1 + \frac{\gamma-1}{2} M_E^2\right)^{\sigma_1}}{M_E \left(\frac{\gamma+1}{2}\right)^{\sigma_1}}$$

$$(7) \quad C_{F_{max}} = \frac{\gamma M_E \left(\frac{\gamma+1}{2}\right)^{-\sigma_1}}{\sqrt{1 + \frac{\gamma-1}{2} M_E^2}}$$

$$(8) \quad C^* = \frac{C_{EFF}}{C_F}$$

Performance Calculations from Measured Values

$$(9a) \quad \dot{W}_{TOTAL} = \dot{W}_{TC_F} + \dot{W}_{TC_O}$$

$$(9b) \quad \dot{W}_{TOTAL} = \dot{W}_{TC_F} + \dot{W}_{TC_O} + \dot{W}_{GG_F} + \dot{W}_{GG_O}$$

$$(10) \quad \dot{m} = \frac{\dot{W}_{TOTAL}}{g}$$

$$(11) \quad O/F = \frac{\dot{W}_{TC_O}}{\dot{W}_{TC_F}}$$

$$(12) \quad I_{sp} = \frac{F}{\dot{W}_{TOTAL}}$$

$$(13) \quad C_{EFF} = \frac{F}{\dot{m}}$$

$$(14) \quad P_{calculated} = \frac{P_{c_{inj}} (1 + \frac{\gamma-1}{2} M_i^2)^{\sigma_2}}{1 + \gamma M_i^2}$$

where

$$0.15 \leq M_i \leq 0.45$$

$$(15) \quad P_{calculated_{MAX}} = \frac{P_{c_{inj}} (1 + \frac{\gamma-1}{2} \cdot 0.15^2)^{\sigma_2}}{1 + \gamma \cdot 0.15^2}$$

$$(16) \quad M_E = \sqrt{\frac{2}{\gamma-1} \left(\left(\frac{P_e}{P_c} \right)^{-\frac{\gamma-1}{\gamma}} - 1 \right)}$$

$$(17) \quad C_F = C_1 + C_2$$

where

$$C_1 = \frac{\gamma M_E \left(\frac{\gamma+1}{2} \right)^{-\sigma_1}}{\sqrt{1 + \frac{\gamma-1}{2} M_E^2}}$$

and

$$C_2 = \frac{\left(\left(1 + \frac{\gamma-1}{2} M_E^2 \right)^{-\sigma_2} - \frac{P_a}{P_c} \right) \cdot \left(1 + \frac{\gamma-1}{2} M_E^2 \right)^{\sigma_1}}{M_E \left(\frac{\gamma+1}{2} \right)^{\sigma_1}}$$

$$(18) \quad C^* = \frac{g \cdot I_{sp}}{C_F}$$

Equations Relevant to Performance Optimization

$$(19) \quad F = \dot{m}u_e + (P_e - P_a) \cdot A_e$$

$$(20) \quad u_e = \sqrt{\frac{2\gamma R}{\gamma - 1} T_c \left(1 - \left(\frac{P_e}{P_c} \right)^{\frac{1}{\sigma_2}} \right)}$$

$$(21) \quad \dot{m} = \sqrt{\frac{\gamma}{R}} \cdot \frac{P_c A^*}{\sqrt{T_c}} \cdot \left(\frac{\gamma + 1}{2} \right)^{-\sigma_1}$$

$$(22) \quad F = I_{sp} \dot{W}$$

$$(23) \quad I_{sp} = \frac{c^* C_F}{g}$$

$$(24) \quad c^* = \frac{P_c A^* g}{\dot{W}} = \frac{\sqrt{g\gamma R T_c}}{\gamma \sqrt{\left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma + 1}{\gamma - 1}}}}$$

$$(25) \quad \frac{P_e}{P_c} = \left(\frac{T_e}{T_c} \right)^{\sigma_2}$$

$$(26) \quad R = \frac{R}{M.W.}$$

$$(27) \quad C_F = \frac{F}{P_c A^*}$$

$$= \sqrt{\frac{2\gamma^2}{\gamma - 1} \cdot \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma + 1}{\gamma - 1}} \cdot \left(1 - \left(\frac{P_e}{P_c} \right)^{\frac{1}{\sigma_2}} \right) + \frac{A_e}{A^*} \cdot \left(\frac{P_e - P_a}{P_c} \right)}$$

PERFORMANCE PARAMETERS

γ	specific heat ratio
A^*	nozzle throat area (ft^2)
A_E	nozzle exit area (ft^2)
c^*	characteristic exhaust velocity (ft/sec)
C_{EFF}	effective exhaust velocity (ft/sec)
C_F	thrust coefficient
C_{FMAX}	maximum thrust coefficient
F	thrust (lb)
g	gravitational constant (ft/sec^2)
I_{sp}	specific impulse (sec)
$\dot{m}$	propellant mass flow rate (lbm/sec)
M_E	nozzle exit Mach number
M_i	thrust chamber propellant injector Mach number
$M.W.$	propellant average molecular weight
O/F	oxidizer to fuel ratio
P_a	ambient pressure ($psia$)
P_c	thrust chamber nozzle stagnation pressure ($psia$)
$P_{calculated}$	calculated thrust chamber nozzle stag. pressure ($psia$)
$P_{calculated_{max}}$	maximum calculated thrust chamber nozzle stag. pressure ($psia$)
$P_{c_{inj}}$	thrust chamber propellant injector pressure ($psia$)
$P_{c_{max}}$	maximum thrust chamber nozzle stagnation pressure ($psia$)

P_e	nozzle exit static pressure (<i>psia</i>)
R	universal gas constant (<i>ft - lb/lbm - mol - °R</i>)
R	specific gas constant (<i>ft - lb/lbm - mol - °R</i>)
T_c	thrust chamber temperature (<i>°R</i>)
T_e	nozzle exit static temperature (<i>°R</i>)
u_e	nozzle exhaust velocity (<i>ft/sec</i>)
$\dot{W}$	propellant weight flow rate (<i>lb/sec</i>)
$\dot{W}_{GG_F}$	gas generator fuel weight flow rate (<i>lb/sec</i>)
$\dot{W}_{GG_O}$	gas generator oxidizer weight flow rate (<i>lb/sec</i>)
$\dot{W}_{TC_F}$	thrust chamber fuel weight flow rate (<i>lb/sec</i>)
$\dot{W}_{TC_O}$	thrust chamber oxidizer weight flow rate (<i>lb/sec</i>)
$\dot{W}_{TOTAL}$	total propellant weight flow rate (<i>lb/sec</i>)

A.1 Performance Equation Derivation Assumptions

The equations presented in this Appendix are taken from references [4], [5], [6], [12], and [13]. The assumptions made in their derivation are the following:

- 1) The combustion gases obey the perfect gas law
- 2) The gases are uniform in composition throughout the chamber and the nozzle
- 3) There are no friction or heat losses in the system
- 4) The flow is steady and constant
- 5) The characteristic of the gases (i.e., Their velocity, composition, temperature, pressure, etc.) are uniform over any cross-section of the combustion chamber or nozzle
- 6) All species of the working fluid are gaseous

The first assumption is valid due to the high temperatures that accompany these. With the second assumption, the gases are assumed to transverse the nozzle so rapidly that no significant change in equilibrium composition of the gases (due to changes in temperature and pressure) can take place because of kinetic limitations. The third assumption is needed to consider the flow adiabatic. The larger and better insulated the rocket engine combustion chamber, the more valid this assumption

becomes. By neglecting frictional losses too, the overall assumption of an isentropic system becomes valid. The isentropic condition covers the next assumption as well, which means the flow is free of shocks or discontinuities. The fifth assumption requires one-dimensional flow, neglects boundary layer effects as well as any characteristic changes associated with radial flow, and the final assumption neglects two-phase flow losses [5].

A.2 Analysis of the Rocket Energy Conversion Process

The following discussion was obtained from references [4], [5], [12], and [13]. Equation 19 serves as a working equation for rocket thrust. The exhaust velocity and pressure at the exit of the rocket nozzle in this equation must be related to the internal workings of the rocket motor to provide the capability of performance evaluation in terms of motor designvariables. It is desired to relate u_e and p_e to the chemistry of the propellant system and to the design variables like "chamber pressure" that essentially define the state of the art of the motor.

The fundamental function of a rocket motor is to generate a force called thrust which propels the rocket vehicle that the motor is mounted in. The rocket motor operates on an open thermodynamic cycle and

works by converting energy from a stored form to the form of kinetic energy of the propellant. For the chemical rocket, the energy conversion process begins with the release of the internal energy of a high temperature gas through the process of combustion. This energy is then converted to kinetic energy of the high speed propellant through the process of gas dynamic expansion. The energy conversion process occurs in the expansion where the random "thermal" motion of the hot gaseous propellant is converted into a non-random directed motion in the nozzle.

The maximum velocity that can be achieved in any nozzle expansion can easily be estimated. From statistical thermodynamics, the root-mean-squared molecular speed of random thermal motion can be calculated by the following equation:

$$V_{\text{thermal}} = C_{\text{r.m.s. avg.}} = (C_{\text{avg.}}^2)^{1/2} = (3P/\text{density})^{1/2}$$

By using the ideal gas equation of state ($V_{\text{thermal}} = (3RT)^{1/2}$), we can evaluate V_{thermal} in the rocket combustion chamber. Defining T_c to be the temperature in the combustion chamber, the thermal velocity of the random motion of the molecules of gas is on the average the following:

$$V_{\text{thermal}} = (3RT_c)^{1/2}$$

The bulk fluid velocity at any station of a one-dimensional flow of an ideal gas with constant specific heat C_p is $V = (2C_p(T_c - T))^{1/2}$, where T_c is again the combustion chamber temperature. In the limit of the expansion T goes to zero, so the limiting velocity of expansion, which is the bulk, directed motion speed of the gas, is $V_{\max} = (2C_p T_c)^{1/2}$. With $C_p = (SHR)(R)/(SHR - 1)$, where SHR is the specific heat ratio C_p/C_v , the equation for $V_{\max}$ becomes $V_{\max} = ((2(SHR)(R)(T_c))/(SHR - 1))^{1/2}$.

Now the ratio of $V_{\max}$ to V_{thermal} becomes the following:

$$\begin{aligned} V_{\max}/V_{\text{thermal}} &= (2(SHR)RT_c/(SHR-1))^{1/2}/(3RT_c)^{1/2} \\ &= (2SHR/3(SHR - 1))^{1/2} \end{aligned}$$

This may be interpreted as saying that, in the expansion process of the hot exhaust passing through the nozzle, at most this amount of the random thermal velocity of the molecules in the combustion chamber can be converted to one-dimensional non-random directed velocity.

Since SHR values range from about 1.08 to 1.667, the ratio

$V_{\max}/V_{\text{thermal}}$ falls in the corresponding range 3 to about 1.3. The ratio is greater than unity and increases as specific heat ratio decreases.

This illustrates the concept of the rocket motor functioning as an energy

conversion device and crudely describes the dynamical features of the conversion of internal energy of a high temperature gas to kinetic energy of nonrandom directed motion.

Now if the one-dimensional flow of a gaseous propellant passing through a rocket motor is considered, it can first be assumed that the gas is not undergoing chemical reactions. Then, neglecting heat transfer, the energy equation for the stream can be written as:

$$h_c + u_c^2/2 = h_e + u_e^2/2$$

where "c" refers to the combustion chamber and "e" refers to any downstream station. For convenience "e" will be considered to refer to the exit or exhaust station. Assuming that the Mach number M_c is very small, i.e. for $M_c \leq 0.1$, then $u_c^2/2$ may be neglected compared to h_c .

As a result, neglecting $u_c^2/2$ and solving for u_e provides $u_e =$

$(2(h_c - h_e))^{1/2}$ valid for any non-reacting flow from "c" to "e". For a flow

with constant specific heats the equation changes to the following form:

$$u_e = (2C_p(T_c - T_e))^{1/2} = (2C_p T_c (1 - T_e/T_c))^{1/2}$$

For an isentropic flow with constant specific heats this changes to the form:

$$u_e = (2C_p T_c (1 - (P_e/P_c)^{(SHR-1)/SHR}))^{1/2}$$

Since $C_p = (SHR)R/(SHR - 1)$, this can be expressed in the form of eqn. 20:

$$u_e = ((2(SHR)R/(SHR - 1))T_c(1 - (P_e/P_c)^{(SHR - 1)/SHR}))^{1/2}$$

This relates the exhaust velocity of the rocket to the chemistry of the propellant system. The mass flow of a one-dimensional stream is $MF = (DEN)AV$, where DEN refers to the density of the stream. For steady flow:

$$MF = (DEN)_1 A_1 V_1 = (DEN)_2 A_2 V_2 = \dots = \text{const.}$$

Evaluating this equation at the "throat" or minimum flow area of a rocket, the form becomes $MF^* = (DEN)^* A^* V^*$, where superscript * refers to the throat conditions. Assuming isentropic flow the following equations become true:

$$(DEN)^* = (DEN)_c (1 + ((SHR - 1)/2)M^{*2})^{-1/(SHR-1)}$$

$$= (P_c/RT_c) (1 + ((SHR - 1)/2)M^{*2})^{-1/(SHR-1)}$$

$$V^* = M^* a^* = M^* ((SHR)RT^*)^{1/2}$$

$$\text{OR } V^* = M^* ((SHR)RT_c)^{1/2} (1 + ((SHR - 1)/2)M^{*2})^{-1/2}$$

Substituting and collecting terms, the mass flow equation becomes:

$$MF = (SHR/R)^{1/2} P_c A^* M^*/T_c^{1/2} (1 + ((SHR - 1)/2)M^{*2})^{-(SHR+1)/2(SHR-1)}$$

At the throat, experiment and theory indicate that for a one-dimensional

flow $M^*=1$, i.e. the Mach number of the throat is "sonic". As a result, the mass flow rate equation now takes the form of equation 21:

$$MF = (SHR/R)^{1/2} P_c A^* / (T_c^{1/2}) ((SHR + 1)/2)^{-(SHR+1)/2} (SHR-1)$$

This relates the mass flow rate of the rocket to the chemistry of the propellant system.

Going back to the previous mass flow equation, we generalize it for any area and Mach number. This provides the equation:

$$MF = (SHR/R)^{1/2} P_c A M / T_c^{1/2} (1 + ((SHR - 1)/2) M^2)^{-(SHR+1)/2} (SHR-1)$$

Equating the two forms and solving for area ratio provides the following:

$$A/A^* = ((SHR+1)/2)^{-(SHR+1)/2} (SHR-1) / M (1 + ((SHR-1)/2) M^2)^{-(SHR+1)/2} (SHR-1)$$

Rearranged this becomes:

$$A_e = A^* (1 + ((SHR-1)/2) M_e^2)^{(SHR+1)/2} (SHR-1) / M_e ((SHR+1)/2)^{(SHR+1)/2} (SHR-1)$$

For isentropic flow, M_e can be found in terms of P_e/P_c . From the energy equation ($h_c + u_c^2/2 = h_e + u_e^2/2$), assuming $u_c^2/2 \ll h_c$ and constant specific heats, the following equation is provided:

$$T_c = T_e + u_e^2/2C_p$$

Since $u_e = M_e((SHR)RT_e)^{1/2}$, then

$$T_c = T_e + M_e^2(SHR)RT_e/C_p = T_e(1 + ((SHR-1)/2)M_e^2)$$

and $T_e/T_c = (1 + ((SHR-1)/2)M_e^2)^{-1}$

Now we examine the isentropic relation between pressure and temperature:

$$P_e/P_c = (T_e/T_c)^{SHR/(SHR-1)} = (1 + ((SHR-1)/2)M_e^2)^{-SHR/(SHR-1)}$$

As a result:

$$P_e = P_c(1 + ((SHR-1)/2)M_e^2)^{-SHR/(SHR-1)}$$

$$M_e = ((2/(SHR-1))((P_e/P_c)^{-((SHR-1)/SHR)} - 1))^{1/2}$$

Mass flow rate, exhaust velocity, nozzle exit area, and nozzle exit pressure have all been related to the chemistry of the propellant. This only leaves the ambient pressure parameter in equation 19 which is provided at the start of a test. As a result, thrust has now been related to the energy conversion process of the rocket. Equation 12 relates specific impulse to thrust and the weight flow rate of the propellant. Equation 13 relates the effective exhaust velocity to thrust and mass flow rate of the propellant. Equation 24 relates the characteristic exhaust velocity to chamber pressure, throat area, and the weight flow of the propellant. Equation 27 relates the thrust coefficient to thrust, chamber pressure, and throat area. As a result, all performance parameters have been related to the energy conversion process.

Analysis of the energy conversion process has indeed been the basis for identifying true rocket performance standards, as stated earlier.

The thrust chamber nozzle stagnation pressure is not a measurable parameter, however. Instead, it is calculated from the thrust chamber injector pressure using equation 14. This equation is for isentropic processes only and was obtained from reference [6]. Typical values for the injector Mach number were obtained from reference [8] and were determined to be 0.15 to 0.45. Since an injector Mach number of 0.15 gives the highest value for the chamber nozzle stagnation pressure, it is used in all calculations made by the expert system. This is to assure that the largest pressure value possible is used in the checks against safety limits. The actual value may be lower, but choosing the higher number adds a little extra safety margin.

APPENDIX B

ROCKET ENGINE TEST SIMULATION

B.1 Simulated Data Files Used

To show the expert system in full operation, a rocket engine test simulation was created. The user follows the same procedure, described in chapter two, when operating the expert system, except the measured test parameters it obtains are from three pre-constructed ASCII data files instead of an operating rocket test. The files are copied into the main ASCII file and the expert system draws its information from one by one each time as the user requests to continue monitoring the test. The fourth time such a request is made, the first file will again be copied. This loop through the three files of information will continue until the user indicates test monitoring should be discontinued. The archive files constructed will consist of the three data files with current times being the last three times monitoring took place. In other words, if the user monitors the test nine times, the clock time of each of the last three monitoring runs will be the stored times of the three archive files.

B.2 Initial Design Parameter Entry

Expert system operation is begun by typing "VPX ROCKET". This tells the operating system to execute the VP-EXPERT software using the file named "ROCKET". First, a title page appears indicating the name and author of this expert system (Figure 14). The user is instructed to press any key to start the system and therefore does so. Next, the rocket engine design parameters are requested. This is done by providing a listing of the parameters such as nozzle area ratio, oxidizer to fuel ratio, thrust, exhaust gas specific heat ratio, propellant molecular weight, nominal on desired specific impulse, and ambient pressure, and their default values (Figure 15). To modify these values for the current rocket, the user clicks the mouse on the field surrounding each number and types the new value. Pressing "return" causes the field boundary to return with the new value inside. The user may modify these values as many times as desired. When all modifications have been completed, the user is instructed to click the mouse on the field at the bottom of the screen labeled "CONTINUE". For this simulation, the values displayed in figure 16 were utilized.

B.3 Safety Limit Entry

ROCKET ENGINE PERFORMANCE OPTIMIZATION EXPERT SYSTEM
BY PETER A. MONTGOMERY

PRESS ANY KEY TO BEGIN

Figure 14. Expert system title Page

ROCKET ENGINE DESIGN PARAMETERS

AREA RATIO: 7

O/F: 2.23

THRUST (LB): 200000

SPECIFIC HEAT RATIO: 1.2

PROPELLANT MOLECULAR WEIGHT: 32

SPECIFIC IMPULSE (SEC): 257.68

AMBIENT PRESSURE (PSIA): 14.7

MAKE ALL NECESSARY CHANGES AND CLICK HERE WHEN DONE

Figure 15. Design Parameter Default
Values

ROCKET ENGINE DESIGN PARAMETERS

AREA RATIO: 6.53
O/F: 2.35
THRUST (LB): 210000
SPECIFIC HEAT RATIO: 1.2
PROPELLANT MOLECULAR WEIGHT: 32
SPECIFIC IMPULSE (SEC): 257.68
AMBIENT PRESSURE (PSIA): 14.7

MAKE ALL NECESSARY CHANGES AND CLICK HERE WHEN DONE

Figure 16. User Defined Designed
Parameters

Now the expected performance parameters of the rocket are calculated, using equations 1 through 8 in Appendix A, and stored for later recall. The next screen that appears is the first of two safety limit parameter screens (Figure 17). All default values are zero. The user defines these values by clicking on each field surrounding the numbers with the mouse, typing the new value, and hitting "return" to return the field to normal with the new value displayed. This may be done as many times as desired until the user is satisfied with the defined values. For this simulation, the values displayed in figure 18 were utilized. Clicking the mouse on the field marked "CONTINUE" at the bottom of the screen, defines the values and calls up the next safety limit screen (Figure 19). For this simulation, the values displayed in figure 20 were utilized.

One final pre-test user input is requested upon completion of the safety limit definitions. This input is the handling of the gas generator fuel and oxidizer flow. The user is asked if this flow is dumped into the nozzle (Figure 20). If the user answer is yes, then the measured values of gas generator fuel and oxidizer flow are added to those of

CURRENT ROCKET ENGINE PARAMETER LIMITS

MINIMUM ENVIRONMENTAL TEMPERATURE (DEG. F): 0

MAXIMUM ENVIRONMENTAL TEMPERATURE ALLOWED (DEG. F): 0

FUEL PUMP INLET PRESSURE LIMIT (PSIA): 0

FUEL PUMP INLET TEMPERATURE LIMIT (DEG. F): 0

FUEL PUMP OUTLET PRESSURE LIMIT (PSIA): 0

FUEL PUMP OUTLET TEMPERATURE LIMIT (DEG. F): 0

OXIDIZER PUMP INLET PRESSURE LIMIT (PSIA): 0

OXIDIZER PUMP INLET TEMPERATURE LIMIT (DEG. F): 0

OXIDIZER PUMP OUTLET PRESSURE LIMIT (PSIA): 0

OXIDIZER PUMP OUTLET TEMPERATURE LIMIT (DEG. F): 0

CLICK FIELDS TO MODIFY LIMIT VALUES CLICK HERE WHEN DONE

Figure 17. Safety Limits - Screen 2

CURRENT ROCKET ENGINE PARAMETER LIMITS

MINIMUM ENVIRONMENTAL TEMPERATURE (DEG. F): 0

MAXIMUM ENVIRONMENTAL TEMPERATURE ALLOWED (DEG. F): 130

FUEL PUMP INLET PRESSURE LIMIT (PSIA): 250

FUEL PUMP INLET TEMPERATURE LIMIT (DEG. F): 100

FUEL PUMP OUTLET PRESSURE LIMIT (PSIA): 1124

FUEL PUMP OUTLET TEMPERATURE LIMIT (DEG. F): 110

OXIDIZER PUMP INLET PRESSURE LIMIT (PSIA): 250

OXIDIZER PUMP INLET TEMPERATURE LIMIT (DEG. F): -230

OXIDIZER PUMP OUTLET PRESSURE LIMIT (PSIA): 1195

OXIDIZER PUMP OUTLET TEMPERATURE LIMIT (DEG. F): -200

CLICK FIELDS TO MODIFY LIMIT VALUES CLICK HERE WHEN DONE

Figure 18. User - Defined Safety Limits - Screen 1

CURRENT ROCKET ENGINE PARAMETER LIMITS

THRUST CHAMBER NOZZLE STAGNATION PRESSURE LIMIT (PSIA): 0
THRUST CHAMBER INJECTOR PRESSURE LIMIT (PSIA): 0
THRUST CHAMBER TEMPERATURE LIMIT (DEG. F): 0
TURBOPUMP SPEED LIMIT (RPM): 0
TURBINE EXIT STATIC PRESSURE (PSIA): 0
GAS GENERATOR CHAMBER PRESSURE LIMIT (PSIA): 0
GAS GENERATOR CHAMBER TEMPERATURE LIMIT (DEG. F): 0
TURBOPUMP LUBRICANT CONSUMPTION LIMIT (GPM): 0
THRUST CHAMBER COOLING SYSTEM FLOW RATE LIMIT (LB. SEC): 0

CLICK FIELDS TO MODIFY LIMIT VALUES CLICK HERE WHEN DONE

Figure 19. Safety Limits Defaults- Screen 2

CURRENT ROCKET ENGINE PARAMETER LIMITS

THRUST CHAMBER NOZZLE STAGNATION PRESSURE LIMIT (PSIA):	1000
THRUST CHAMBER INJECTOR PRESSURE LIMIT (PSIA):	1200
THRUST CHAMBER TEMPERATURE LIMIT (DEG. F):	2500
TURBOPUMP SPEED LIMIT (RPM):	4320
TURBINE EXIT STATIC PRESSURE (PSIA):	52.5
GAS GENERATOR CHAMBER PRESSURE LIMIT (PSIA):	800
GAS GENERATOR CHAMBER TEMPERATURE LIMIT (DEG. F):	1000
TURBOPUMP LUBRICANT CONSUMPTION LIMIT (GPM):	7
THRUST CHAMBER COOLING SYSTEM FLOW RATE LIMIT (LB/SEC):	400

CLICK FIELDS TO MODIFY LIMIT VALUES

CLICK HERE WHEN DONE

Figure 20. User - Defined Safety Limits - Screen 2

the thrust chamber fuel and oxidizer flow to obtain the total flow rate through the rocket. If the answer is no, then the gas generator flow is not included in the total flow rate calculation. For this simulation, the user answered no.

B.5 Expert System Main Menu

At this point, the actual rocket engine test would normally begin. For this simulation, it is assumed that this has indeed happened and the test-time-elapsed clock is started. The main expert system operations control menu now appears on the screen as well (Figure 21). If the user is unclear about the meaning of each menu choice and wishes an overall operations summary, clicking the mouse on the light-blue title word "EXPERT" will provide one. This is done and first page of the summary replaces the main menu in the display window (Figure 22). Clicking the mouse on the light-blue "PAGE2" causes the window to now display the second page of the summary (Figure 23). Clicking the mouse on the light-blue "RETURN" causes the window to redisplay the main menu. The user may also click on the light-blue words "UPDATE", "PERFORMANCE", and "ROCKET" to obtain explanations of the listed menu option they are part of (See Figures 24, 25, and 26)

IS THE GAS GENERATOR FUEL AND OXIDIZER FLOW DUMPED INTO THE NOZZLE (Y OR N)?

Figure 21. Gas Generator Flow Handling Inquiry

```

#####
#####ROCKET ENGINE EXPERT SYSTEM OPERATIONS#####
#####UPDATE ROCKET TEST PARAMETER FILE#####
#####DISPLAY CURRENT ROCKET TEST PARAMETER VALUES#####
#####COMPARE TEST PARAMETERS TO EXPECTED PERFORMANCE#####
#####COMPARE TEST PARAMETERS TO ROCKET ENGINE DATABASE#####
#####PERFORMANCE OPTIMIZATION ADVICE#####
#####ROCKET ENGINE LIFE-HISTORY DATABASE OPERATIONS#####
#####END EXPERT SYSTEM OPERATIONS#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION#####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS#####
#####

```

Figure 22. Expert System Main Menu

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
MAIN MENU OPERATIONS SUMMARY
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
. THE FIRST MENU SELECTION CAUSES THE CURRENT PARAMETER
VALUES FROM AN ONGOING ROCKET ENGINE TEST TO BE READ
IN, FROM A CONSTANTLY UPDATED ASCII FILE, AND COMPARED
TO THE USER-DEFINED SAFETY LIMITS.
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
. THE SECOND MENU SELECTION CAUSES THE CURRENT ROCKET
PARAMETER DATA SET TO BE DISPLAYED FOR REVIEW ALONG
WITH ANY RECOMMENDATIONS MADE SO FAR.
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
. THE THIRD MENU SELECTION CAUSES A COMPARISON OF THE
CURRENT MEASURED PERFORMANCE VALUES TO THE EXPECTED
PERFORMANCE VALUES AND THE CALCULATION OF THE RATIO
OF THE MEASURED TO EXPECTED (IDEAL) VALUES.
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CLICK HERE FOR NEXT PAGE:      PAGE2
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION.
CLICK THE BLUE TITLE WORD 'EXPERT' FOR A SUMMARY OF MENU OPERATIONS.
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

```

Figure 23. Main Menu Operations Survey- Page 1

```

#####
#####
#####
#####
#####
THE FOURTH MENU SELECTION CAUSES A COMPARISON OF THE
CURRENT MEASURED PERFORMANCE VALUES TO THE MINIMUM
AND MAXIMUM PERFORMANCE VALUES OF THE ROCKET DATABASE.
#####
#####
THE FIFTH MENU SELECTION CAUSES ADVICE TO BE PROVIDED
TO EXPLAIN WHY THE MEASURED TO EXPECTED PERFORMANCE
RATIOS ARE NOT EQUAL TO ONE. SUGGESTIONS FOR IMPROVING
PERFORMANCE ARE PROVIDED AS WELL.
#####
#####
THE SIXTH MENU SELECTION CAUSES A DATABASE OPERATIONS
MENU TO BE DISPLAYED. FROM THIS, CHOICES MAY BE MADE
TO ADD THE CURRENT DATA SET TO THE ROCKET DATABASE,
DELETE A SELECTED DATA SET FROM THE DATABASE, OR REVIEW
THE DATA SETS CURRENTLY IN THE DATABASE.
#####
#####
CLICK HERE TO GO BACK TO THE MAIN MENU: RETURN
#####
#####
CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION
#####
CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS
#####

```

Figure 24. Main Menu Operations Survey- Page 2

```

#####
#####
#####
#####
##### THIS MENU OPTION ALLOWS FOR THE INPUT OF ROCKET ENGINE #####
##### TEST INFORMATION FROM AN ASCII FILE CONSTANTLY MODIFIED #####
##### BY THE ONGOING TEST. IN THE PROCESS OF READING THIS #####
##### INFORMATION, VALUES ARE CHECKED AGAINST THE USER DEFINED #####
##### SAFETY LIMITS AND DISPLAYED IN RED WHEN EXCEEDING THESE. #####
##### THE LIMIT VALUE AND AN OPTION TO IGNORE DATA FROM #####
##### A CHANNEL RETURNING BAD DATA ARE PROVIDED AT THE BOTTOM #####
##### OF THE SCREEN. ONCE THE INFORMATION FOR A PARTICULAR #####
##### POINT IN TIME IS OBTAINED, IT MAY BE USED IN THE #####
##### OTHER MENU OPERATIONS. #####
#####
##### PRESS ANY KEY TO RETURN TO MAIN MENU #####
#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION#####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS#####
#####

```

Figure 25. Menu Option One Explanation

```

#####
#####
#####
#####
##### THIS MENU OPTION PROVIDES ADVICE AS TO WHICH TEST #####
##### PARAMETERS TO CHANGE TO IMPROVE PERFORMANCE OF THE #####
##### ROCKET. THESE RECOMMENDATIONS ARE BASED ON INFORMATION #####
##### GATHERED ON THE MOST IMPORTANT TEST PARAMETERS TO #####
##### CONSIDER WHEN OPTIMIZING PERFORMANCE. BEFORE ANY #####
##### ADVICE IS GIVEN, THE PARAMETERS IT WILL AFFECT ARE #####
##### CHECKED FOR NEARNESS TO THEIR SAFETY LIMITS AND THE #####
##### VALIDITY OF ANY INPUT CHANNELS INVOLVED IS CONSIDERED #####
##### AS WELL. #####
#####
#####
#####
#####
#####
#####
##### PRESS ANY KEY TO RETURN TO THE MAIN MENU #####
#####
#####
#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION#####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS#####
#####
#####

```

Figure 26. Menu Option Five Explanation

respectively).

B.6 Monitoring Data Set 1

With the test operating the user selects the first menu option, "UPDATE ROCKET TEST PARAMETER FILE", to monitor the test. For this simulation, the first data file is now copied into the expert system ASCII test parameter file. Display of the monitoring screen is then begun. The first measured parameter, environmental temperature, is compared to its safety limit and immediately is flagged as exceeding displayed in red, as is the safety limit and the ignore-channel inquiry shown at the bottom of the screen (Figure 27). Since this value is clearly unreasonable, the user concludes the channel is malfunctioning is indicates it should be ignored in future monitoring for this particular test. The parameter display continues, once the user makes this indication, and the fuel pump inlet and outlet temperatures are flagged in a similar manner. These channels will not be ignored in future monitoring. The resulting monitor screen for the first simulated data set is displayed in figure 28. Note the nominal values for the important performance parameters are displayed next to the measured values for easy reference.

```

#####
#####
#####
#####
##### THIS OPTION PROVIDES FOR THE STORAGE OF THE CURRENT #####
##### DATA AND RECOMMENDATIONS FOR THIS TEST IN A DATABASE #####
##### OF ALL ROCKET TESTS CONSIDERED BY THE SYSTEM SO FAR. #####
##### EXAMINATION OF THESE OTHER TESTS IS ALSO AN OPTION HERE #####
##### AS WELL AS THE REMOVAL OF A SELECTED CASE FROM THE #####
##### DATABASE FOR WHATEVER REASON. #####
#####
##### PRESS ANY KEY TO RETURN TO THE MAIN MENU #####
#####
#####
#####
#####CLICK THE LIGHT-BLUE WORDS OF ANY MENU OPTIONS FOR MORE INFORMATION#####
#####CLICK THE BLUE TITLE WORD "EXPERT" FOR A SUMMARY OF MENU OPERATIONS#####
#####

```

Figure 27. Menu Option six Explanation

CURRENT ROCKET ENGINE TEST DATA
CURRENT TIME: 02:18:47 TEST TIME ELAPSED: 19.3 SEC
ENV. TEMP. (DEG. F): 150*

MAX. TEMP.: 100 DEG. F IGNORE THIS CHANNEL (Y OR N)?

Figure 28. Outside- Limit Parameter Warning

B.7 Monitoring Data Set 2

After examining the values, the user indicates a wish to continue monitoring and the simulated second data set is read into the ASCII file. Notice now that the environmental temperature is not flagged, since ignoring the channel was chosen previously, the fuel pump inlet and outlet temperatures have dropped below their safety limitations, and the turbopump lubricant consumption and thrust chamber temperature have now climbed beyond their safety limitations (Figure 29). Both channels were decided not to be ignored in future monitoring.

B.8 Monitoring Data Set 3

After examining these value, the user again indicates a wish to continue monitoring and the simulated third data set is read into the ASCII file. Notice now that the turbopump lubricant consumption was adjusted below its safety limit and how the coolant flow rate was increased to decrease the thrust chamber temperature below its limit (Figure 30). All parameter values are within their limitations at this time. At this point the user may continue the simulation, if desired, by indicating a wish to continue monitoring. If this is done, the first

CURRENT ROCKET ENGINE TEST DATA

CURRENT TIME: 02:16:47	TEST TIME ELAPSED: 18.5 SEC		
ENV. TEMP. (DEG. F): 150*	COOLANT FLOW RATE (LB/S): 284.67		
GG PRESSURE (PSIA): 671.49	TC FUEL FLOW (LB/S): 289.23		
GG TEMPERATURE (DEG. F): 900	TC OXID FLOW (LB/S): 654.95		
TB STATIC PRESS. (PSIA): 34.96	GG FUEL FLOW (LB/S): 13.21		
TP SPEED (RPM): 4100	GG OXID FLOW (LB/S): 4.59		
TP LUB. CONSUM. (GPM): 6	FUEL QUANTITY (LB): 40000		
FP INLET PRESS. (PSIA): 77	OXID QUANTITY (LB): 90000		
FP INLET TEMP. (DEG. F): 277.29*		MEASURED	NOMINAL
FP OUTLET PRESS. (PSIA): 538.95			
FP OUTLET TEMP. (DEG. F): 120*	NOZ EA PRESS (PSIA): 14.5	14.7	
OP INLET PRESS. (PSIA): 180	MEAS. THRUST (LB): 209790	210000	
OP INLET TEMP. (DEG. F): -290	NOZ ST PRESS. (PSIA): 635.36	650.76	
OP OUTLET PRESS. (PSIA): 765.07	O/F RATIO: 2.26	2.35	
OP OUTLET TEMP. (DEG. F): -295	TOT. FLOW RT (LB/S): 944.16	814.96	
TC INJECTOR PRESS. (PSIA): 653.03	SPEC. IMPULSE (SEC): 222.09	237.58	
TC TEMPERATURE (DEG. F): 2490	EFF EXH. VEL. (FT/S): 7151.55	8237.2	
	THRUST COEFF: 1.55	1.55	
	CHAR EXH VEL (FT/S): 4661.91	5294.8	

PRESS 'PRINT SCREEN' FOR HARD COPY.....CONTINUE MONITORING TEST (Y OR N)??

Figure 29. Data Set 1 Monitor

CURRENT ROCKET ENGINE TEST DATA

CURRENT TIME: 02:21:14	TEST TIME ELAPSED: 285.0 SEC
ENV. TEMP. (DEG. F): 150	COOLANT FLOW RATE (LB/S): 284.67
GG PRESSURE (PSIA): 671.49	TC FUEL FLOW (LB/S): 257.39
GG TEMPERATURE (DEG. F): 900	TC OXID FLOW (LB/S): 625.23
TB STATIC PRESS. (PSIA): 34.96	GG FUEL FLOW (LB/S): 17.21
TP SPEED (RPM): 4100	GG OXID FLOW (LB/S): 4.59
TP LUB. CONSUM. (GPM): 10*	FUEL QUANTITY (LB): 40000
FP INLET PRESS. (PSIA): 57	OXID QUANTITY (LB): 90000
FP INLET TEMP. (DEG. F): 60	
FP OUTLET PRESS. (PSIA): 256	MEASURED NOMINAL
FP OUTLET TEMP. (DEG. F): 30	
OP INLET PRESS. (PSIA): 65	NOZ EX PRES (PSIA): 14.5 14.7
OP INLET TEMP. (DEG. F): -290	MEAS. THRUST (LB): 209700 210000
OP OUTLET PRESS. (PSIA): 310	NOZ ST PRESS. (PSIA): 875.86 880.56
OP OUTLET TEMP. (DEG. F): -265	O/F RATIO: 2.41 2.35
TC INJECTOR PRESS. (PSIA): 653.03	TOT. FLOW RT (LB/S): 880.21 814.96
TC TEMPERATURE (DEG. F): 2550*	SPEC. IMPULSE (SEC): 238.07 257.68
	EFP EXH. VEL. (FT/S): 7665.97 9097.2
	THRUST COEFF: 1.33 1.35
	CHAR EXH VEL (FT/S): 4997.05 5394.8

PRESS *PRINT SCREEN* FOR HARD COPY/*****CONTINUE MONITORING TEST (Y OR N)***

Figure 30. Data Set 2 Monitor

data set is again read in and the comments of the past two paragraphs and this one will still apply. This loop may be done as long as desired. For this simulation test, the user discontinues monitoring at this point and the main menu is once again displayed (Figure 21).

B.9 Recalling Archive Files

Now the test is assumed to be completed, and a post-test analysis may now begin. Selecting the second option of the menu, labeled "DISPLAY CURRENT ROCKET TEST PARAMETER VALUES", the list of all current archive files for the test is displayed in order of the time they were stored (Figure 31). The user decides to examine the last archive made, and therefore selects number three. The archive file is now retrieved from storage, and its storage time and the elapsed time of the test when stored are displayed along with all parameter values and any performance evaluation recommendations made so far (Figure 32). Since post-test analysis has just begun, the recommendations section is currently left blank. The user is also reminded that pressing the "PRINT SCREEN" key will provide a hard copy of the archive file information. After reviewing the values, the user selects any key, and the main menu is again displayed (Figure 21).

CURRENT ROCKET ENGINE TEST DATA

CURRENT TIME: 02:24:57	TEST TIME ELAPSED: 508.1 SEC	
ENV. TEMP. (DEG. F): 62	COOLANT FLOW RATE (LB/S): 325.68	
GG PRESSURE (PSIA): 671.49	TC FUEL FLOW (LB/S): 267.59	
GG TEMPERATURE (DEG. F): 900	TC OXID FLOW (LB/S): 623.23	
TB STATIC PRESS. (PSIA): 34.96	GG FUEL FLOW (LB/S): 13.21	
TP SPEED (RPM): 4100	GG OXID FLOW (LB/S): 4.59	
TP LUB. CONSUM. (GPM): 5	FUEL QUANTITY (LB): 40000	
FP INLET PRESS. (PSIA): 57	OXID QUANTITY (LB): 90000	
FP INLET TEMP. (DEG. F): 60		
FP OUTLET PRESS. (PSIA): 256	MEASURED NOMINAL	
FP OUTLET TEMP. (DEG. F): 80		
OP INLET PRESS. (PSIA): 65	NOZ EX PRES (PSIA): 14.5 14.7	
OP INLET TEMP. (DEG. F): -290	MEAS. THRUST (LB): 209700 210000	
OP OUTLET PRESS. (PSIA): 310	NOZ ST PRESS. (PSIA): 635.86 653.36	
OP OUTLET TEMP. (DEG. F): -285	O/F RATIO: 2.32 2.35	
TC INJECTOR PRESS. (PSIA): 653.03	TOT. FLOW RT (LB/S): 890.81 814.96	
TC TEMPERATURE (DEG. F): 2000	SPEC. IMPULSE (SEC): 235.40 257.68	
	EFF EXH. VEL. (FT/S): 7579.91 8297.2	
	THRUST COEFF: 1.53 1.53	
	CHAR EXH VEL (FT/S): 4941.16 5394.8	

PRESS 'PRINT SCREEN' FOR HARD COPY/CONTINUE MONITORING TEST (Y OR N)???

////////////////////////////////////

Figure 31. Data Set 3 Monitor

CURRENT ROCKET ENGINE TEST ARCHIVES

FILE: 1	TIME STORED: 02:16:47
FILE: 2	TIME STORED: 02:21:14
FILE: 3	TIME STORED: 02:24:57

TYPE THE NUMBER OF THE FILE YOU WISH TO EXAMINE

Figure 32. Archive File List

B.10 Expected Performance Comparison

Now the user selects the third menu option, labeled "COMPARE TEST PARAMETERS TO EXPECTED PERFORMANCE", and again must choose an archive file to be examined (Figure 31). The third file is again selected, and a table of measured, expected, and measured to expected ratios for thrust, specific impulse, effective exhaust velocity, thrust coefficient, characteristic exhaust velocity, oxidizer to fuel ratio, and propellant weight flow rate is displayed (Figure 33). After reviewing the values, the user selects any key, and the main menu is again displayed (Figure 21).

B.11 Database Extreme Value Comparison

Now the user selects the fourth menu option, labeled "COMPARE TEST PARAMETERS TO ROCKET ENGINE DATABASE", and again must choose an archive file to be examined (Figure 31). Once again the third file is selected, and a table of database minimums, database maximums, the current archive files test values, and out-of-range asterisk flags for thrust, specific impulse, oxidizer to fuel ratio, fuel weight flow rate, oxidizer weight flow rate, and total propellant weight

CURRENT ROCKET ENGINE TEST DATA

STORAGE TIME:	02:24:57	TIME ELAPSED (SEC):	508.1
ENV. TEMP. (DEG. F):	62	COOLANT FLOW RATE (LB/S):	325.88
GG PRESSURE (PSIA):	671.49	TC FUEL FLOW (LB/S):	267.59
GG TEMPERATURE (DEG. F):	900	TC OXID FLOW (LB/S):	623.23
YB STATIC PRESS. (PSIA):	34.96	GG FUEL FLOW (LB/S):	10.21
TP SPEED (RPM):	4100	GG OXID FLOW (LB/S):	4.39
TP LUB. CONSUM. (GPM):	5	NOZ EX STAT PRESS. (PSIA):	14.5
FP INLET PRESS. (PSIA):	57	MEASURED THRUST (LB):	209700
FP INLET TEMP. (DEG. F):	60	FUEL QUANTITY (LB):	40000
FP OUTLET PRESS. (PSIA):	256	OXID QUANTITY (LB):	90000
FP OUTLET TEMP. (DEG. F):	80	TC NOZ STAG PRESS. (PSIA):	635.86
OP INLET PRESS. (PSIA):	65	O/F RATIO:	2.32
OP INLET TEMP. (DEG. F):	-290	TOTAL FLOW RATE (LB/S):	990.91
OP OUTLET PRESS. (PSIA):	310	SPECIFIC IMPULSE (SEC):	235.40
OP OUTLET TEMP. (DEG. F):	-285	EFF. EXH. VEL. (FT/S):	7579.91
TC INJECTOR PRESS. (PSIA):	653.03	THRUST COEFF:	1.53
TC TEMPERATURE (DEG. F):	2000	CHAR. EXH. VEL. (FT/S):	4941.16

RECOMMENDATIONS SINCE START OF TEST:

PRESS 'PRINT SCREEN' FOR HARD COPY

PRESS ANY KEY TO RETURN TO THE MAIN MENU

Figure 33. Selected Archive Parameter Values

flow rate is displayed (Figure 34). After reviewing the values, the user selects any key, and the main menu is again displayed (Figure 21).

B.12 Performance Evaluation

Now the user selects the fifth menu option, labeled "PERFORMANCE OPTIMIZATION ADVICE", and again must choose an archive file to be examined (Figure 31). Once again the third file is selected. Now the user is asked to define an acceptable percentage for measured to expected performance ratios to be used in determining the advice to be given. The default value is 92% (Figure 35). This value may be modified by clicking the mouse on the field containing the percentage. The user does this and changes the value to 99% (Figure 36). Once satisfied with this definition, the user clicks the mouse on the field labeled "Begin Optimization" at the bottom of the screen. Having done this, the first performance evaluation recommendation is made. The measured to expected specific impulse ratio is determined to be below 99%. As a result, the measured, expected, and ratio values are displayed along with the statement that the thrust coefficient and characteristic exhaust velocity must be optimized to maximize specific impulse (Figure 37).

MEASURED/EXPECTED PERFORMANCE VALUE COMPARISON			
PARAMETER	MEASURED PERF.	EXPECTED PERF.	MEAS/EXP RATIO
THRUST (LB)	209700	210000	0.99
SPECIFIC IMPULSE (SEC)	235.40	237.68	0.91
EFFECTIVE EXHAUST VELOCITY (FT/S)	7579.91	8297.2	0.91
THRUST COEFFICIENT	1.53	1.53	0.99
CHARACTERISTIC EXHAUST VELOCITY (FT/S)	4941.16	5394.8	0.91
O/F RATIO	2.32	2.35	0.99
PROP. FLOW (LB/S)	890.91	814.96	1.09
PRESS ANY KEY TO RETURN TO MAIN MENU			

Figure 34. Expected Performance Value Comparison

CURRENT TEST AND DATABASE PERFORMANCE PARAMETER COMPARISON

PERFORMANCE PARAMETERS	DATABASE MINIMUMS	DATABASE MAXIMUMS	CURRENT TEST	OUT OF RANGE INDICATOR
THRUST (LB)	200	209700	209700	*
SPECIFIC IMPULSE (SEC)	0.25	1626.01	235.40	
O/F RATIO	0.83	2.41	2.32	
FUEL FLOW (LB/S)	67	400	250.80	
OXID FLOW (LB/S)	56	627.82	627.82	*
TOTAL FLOW (LB/S)	123	898.62	890.81	

PRESS ANY KEY TO RETURN TO MAIN MENU

Figure 35. Database Extreme Performance Value
Comparison

ACCEPTABLE PERCENTAGE FOR MEASURED/EXPECTED PERFORMANCE RATIOS: 92 %

Figure 36. Default Acceptable Ratio Percentage

ACCEPTABLE PERCENTAGE FOR MEASURED/EXPECTED PERFORMANCE RATIOS: 99 %

Figure 37. User-Defined Acceptable Ratio Percentage

The user presses any key, as instructed, and the thrust coefficient is now evaluated. The measured, expected, and ratio values are displayed along with the statement that this parameter is optimized under these conditions, since the ratio is acceptable (Figure 38).

The user presses any key, as instructed, and the characteristic exhaust velocity is now evaluated. The measured, expected, and ratio values are displayed followed by the statement that poor combustion and injector performance are responsible for the low ratio value. This advice is to increase characteristic exhaust velocity by increasing the propellant mass flow rate, and thus the injector and nozzle stagnation pressure. It also notes that the safety limits for injector pressure, nozzle stagnation pressure, and chamber temperature, since temperature increases with pressure, must be watched closely and displays these values and their limits (Figure 39).

The user presses any key, as instructed, and the oxidizer to fuel flow ratio is now evaluated. The measured, expected, and ratio of measured to expected values are displayed along with the statement that this parameter is optimized under these conditions, since the ratio is acceptable (Figure 40). Pressing any key, the user is returned to the

PARAMETER: SPECIFIC IMPULSE

VALUES: EXPECTED SPECIFIC IMPULSE (GEO): 257.68
MEASURED SPECIFIC IMPULSE: 235.40
MEASURED/EXPECTED RATIO: 0.91

ADVICE: THIS RATIO IS AN INDICATION OF THE ROCKET ENGINE PERFORMANCE.
IT IS LOWER THAN THE USER-DEFINED ACCEPTABLE PERCENTAGE WHICH IMPLIES THAT
THE ROCKET ENGINE IS NOT PERFORMING AS WELL AS IT SHOULD. AS A RESULT, THE
SPECIFIC IMPULSE SHOULD BE OPTIMIZED TO OPTIMIZE THE ROCKET PERFORMANCE.
SPECIFIC IMPULSE REACHES ITS OPTIMUM WHEN THE CHARACTERISTIC EXHAUST VELOCITY
AND THE THRUST COEFFICIENT ARE OPTIMIZED. KEEPING THIS IN MIND, THIS
OPTIMIZATION WILL NOW BEGIN.

PRESS ANY KEY TO CONTINUE

Figure 38, Specific Impulse Evaluation Advice

PARAMETER: THRUST COEFFICIENT

VALUES: EXPECTED THRUST COEFFICIENT: 1.53
MEASURED THRUST COEFFICIENT: 1.53
MEASURED/EXPECTED RATIO: 0.99

UNDER CURRENT CONDITIONS, THE THRUST COEFFICIENT IS AT ITS OPTIMUM LEVEL.

PRESS ANY KEY TO CONTINUE

Figure 39, Thrust Coefficient Evaluation

Advice

PARAMETER: CHARACTERISTIC EXHAUST VELOCITY

VALUES: EXPECTED CHARACTERISTIC EXHAUST VELOCITY (FT/S): 5094.8
 MEASURED CHARACTERISTIC EXHAUST VELOCITY (FT/S): 4941.16
 MEASURED/EXPECTED RATIO: 0.91

ADVICE: THIS RATIO IS AN INDICATION OF COMBUSTION AND INJECTOR PERFORMANCE. IT IS LOWER THAN THE USER-DEFINED ACCEPTABLE PERCENTAGE WHICH IMPLIES POOR COMBUSTION AND POOR INJECTOR PERFORMANCE. TO INCREASE THE CHARACTERISTIC EXHAUST VELOCITY, A GREATER PROPELLANT MASS FLOW RATE MUST BE USED TO INCREASE THE INJECTOR PRESSURE AND THUS THE NOZZLE STAGNATION PRESSURE. LIMITS THAT NEED TO BE CONSIDERED WHEN DOING THIS ARE THE INJECTOR PRESSURE LIMIT, THE NOZZLE STAGNATION PRESSURE LIMIT, AND THE THRUST CHAMBER TEMPERATURE LIMIT. THE LAST CONSIDERATION MUST BE MADE DUE TO THE FACT THAT AN INCREASE IN THE CHAMBER PRESSURE WILL BRING ABOUT AN INCREASE IN CHAMBER TEMPERATURE. AS A RESULT, THE VALUES OF THE MENTIONED PARAMETERS AND THEIR LIMITS ARE DISPLAYED BELOW.

NOTE: THRUST CHAMBER NOZZLE STAGNATION PRESSURE: 600.96
 THRUST CHAMBER PRESSURE LIMIT: 1000
 THRUST CHAMBER INJECTOR PRESSURE: 600.03
 THRUST CHAMBER INJECTOR PRESSURE LIMIT: 1200
 THRUST CHAMBER TEMPERATURE: 2000
 THRUST CHAMBER TEMPERATURE LIMIT: 2500
 PRESS ANY KEY TO CONTINUE

Figure 40. Characteristic Exhaust Velocity

Evaluation Advice

main menu (Figure 21).

B.13 Database Operations

B.13.1 Database Additions

Now the user selects the sixth menu option, labeled "ROCKET ENGINE LIFE-HISTORY DATABASE OPERATIONS", and again must choose an archive file to be examined (Figure 31). Once again the third file is selected. This file had a reasonably good agreement with expected values so the user now decides to include it in the database as part of this rocket engine's life history. Selecting the first option of the database operations menu (Figure 41), labeled "ADD NEW ROCKET ENGINE TEST CASES TO DATABASE", all parameter values of the file, as well as the recommendations made for improving performance, are displayed for review before database addition (Figure 42). After examining the list, the user presses any key and must now decide if the reviewed file should indeed be added, or if the operation should be aborted (Figure 44). The user chooses to continue the addition process.

Next a name for the engine tested is requested. This must not be identical to any other names currently in the database, and all spaces]

[illegible]

Figure 41. Database Operations Menu

ROCKET ENGINE NUMBER		TEST DATA
ENGINE:		
ENV. TEMP.:	62 DEG. F	COOLANT FLOW RATE: 325.68 LB/S
GG PRESSURE:	671.49 PSIA	TC FUEL FLOW: 267.59 LB/S
GG TEMPERATURE:	900 DEG. F	TC OXID FLOW: 623.23 LB/S
TS STATIC PRESS.:	34.965 PSIA	GG FUEL FLOW: 15.212 LB/S
TP SPEED:	4100 RPM	GG OXID FLOW: 4.5979 LB/S
TP LUB. CONSUM.:	5 GPM	NOZ E1 STAT PRESS.: 14.5 PSIA
FP INLET PRESS.:	57 PSIA	MEASURED THRUST: 209700 LB
FP INLET TEMP.:	60 DEG. F	FUEL QUANTITY: 40000 LB
FP OUTLET PRESS.:	256 PSIA	OXID QUANTITY: 90000 LB
FP OUTLET TEMP.:	80 DEG. F	TC NOZ STAG PRESS.: 635.8618 PSIA
OP INLET PRESS.:	65 PSIA	O/F RATIO: 2.329048
OP INLET TEMP.:	-390 DEG. F	TOTAL FLOW RATE: 890.319946 LB/S
OP OUTLET PRESS.:	310 PSIA	SPECIFIC IMPULSE: 255.401108 SEC
OP OUTLET TEMP.:	-285 DEG. F	SFF. EXH. VEL.: 7579.916016 FT/SEC
TC INJECTOR PRESS.:	653.03 PSIA	THRUST COEFF: 1.534033
TC TEMPERATURE:	2000 DEG. F	CHAR. EXH. VEL.: 4941.162598 FT/SEC
RECOMMENDATIONS SINCE START OF TEST: INCREASE_PROPELLANT_FLOW		
GOOD_NOZZLE_DESIGN GOOD_FUEL_PUMP_INLET_PRESSURE		
INCREASE_INJECTOR_PRESSURE GOOD_OXID_PUMP_INLET_PRESSURE		
PRESS ANY KEY TO CONTINUE		

Figure 41. File Parameter Value Review

ROCKET ENGINE NUMBER TEST DATA

ENGINE:

ENV. TEMP.:	62 DEG. F	COOLANT FLOW RATE:	325.68 LB/S
GG PRESSURE:	671.49 PSIA	TC FUEL FLOW:	267.39 LB/S
GG TEMPERATURE:	900 DEG. F	TC OXID FLOW:	623.23 LB/S
TB STATIC PRESS.:	34.966 PSIA	GG FUEL FLOW:	10.212 LB/S
TP SPEED:	4100 RPM	GG OXID FLOW:	4.5979 LB/S
TP LUB. CONSUM.:	5 GPM	NOZ 8' STAT PRESS.:	14.5 PSIA
FP INLET PRESS.:	57 PSIA	MEASURED THRUST:	209700 LB
FP INLET TEMP.:	60 DEG. F	FUEL QUANTITY:	40000 LB
FP OUTLET PRESS.:	256 PSIA	OXID QUANTITY:	90000 LB
FP OUTLET TEMP.:	80 DEG. F	TC NOZ STAG PRESS.:	635.8618 PSIA
OP INLET PRESS.:	55 PSIA	O/F RATIO:	2.729048
OP INLET TEMP.:	-290 DEG. F	TOTAL FLOW RATE:	890.319946 LB/S
OP OUTLET PRESS.:	310 PSIA	SPECIFIC IMPULSE:	235.401108 SEC
OP OUTLET TEMP.:	-285 DEG. F	EFF. EXH. VEL.:	7579.916016 FT/SEC
TC INJECTOR PRESS.:	653.03 PSIA	THRUST COEFF:	1.504035
TC TEMPERATURE:	2000 DEG. F	CHAR. EXH. VEL.:	4941.162598 FT/SEC

RECOMMENDATIONS SINCE START OF TEST: INCREASE_PROPELLANT_FLOW
GOOD_NOZZLE_DESIGN GOOD_FUEL_PUMP_INLET_PRESSURE
INCREASE_INJECTOR_PRESSURE GOOD_OXID_PUMP_INLET_PRESSURE
PRESS ANY KEY TO CONTINUE

Figure 43. File Parameter Value Review

TYPE 'B' TO BEGIN OR ANY CHARACTER TO ABORT:

Figure 44. Begin Operation Inquiry

must be underlined. The best way to avoid name duplication is by adding the test number at the end of the name. Since no two tests have the same number, for the same engine, this problem may be avoided. Seven lines of brief comments about the test are also allowed for. The user therefore selects a name and five comments (Figure 45). Pressing "return" after the last comment causes the file to be added to the database and the database operations menu to be redisplayed (Figure 42).

B.13.2 Database Case Information Recall

Now the user decides to examine the cases currently in the database, including the most recent addition. Selecting the third option, labeled "DISPLAY ROCKET ENGINE TEST CASES IN DATABASE" scrolled, if there are too many cases to be displayed on a single screen, by clicking the mouse on the "Next Page" and "Previous Page" fields, and options for "First Page" and "Last Page" display are available if needed. Another option is to click the mouse on the case name to obtain the up to seven lines of brief comments about each test. The user now does this for the case recently added (Figure 45). To return to the case list, the user clicks the mouse on the "Current Page" field. The list is once again displayed (Figure 46).

TYPE THE NAME OF THE ENGINE USED FOR THIS TEST CASE
 (NO MORE THAN 40 CHARACTERS AND ALL SPACES MUST BE UNDERLINED)
 (NOTE: TWO ENGINE NAMES MAY NOT BE EXACTLY ALIKE.)
 H_1A_VERSION_2_TEST_10

TYPE ANY COMMENTS FOR THIS TEST CASE IN THE NEXT SEVEN LINES.
 (EACH LINE SHOULD ONLY BE 40 CHARACTERS LONG. HIT RETURN FOR UNNEEDED LINES.)

TYPE LINE NUMBER ONE: SLIGHT VIBRATIONAL PROBLEM
 TYPE LINE NUMBER TWO: REASONABLE NOISE LEVELS
 TYPE LINE NUMBER THREE: CHAMBER TEMPERATURE FLUCTUATIONS
 TYPE LINE NUMBER FOUR: FUEL PUMP PRESSURE VARIATIONS
 TYPE LINE NUMBER FIVE: GOOD THEORETICAL PERFORMANCE COMPARISON
 TYPE LINE NUMBER SIX:
 TYPE LINE NUMBER SEVEN:

Figure 45. Case Identifier and Comment Entry

```

#####
C
C1 H_1A_VERSION_2_TEST_10
C2 H_1C_TEST_NUMBER_ONE
C3
C4
C5
C6
C7
C8
C9
C10
C11
C12
C13
C14
C15
C16
C17
C18
C19
C20
#####
CLICK ENG NAME AND SEL_CASE BUTTON FOR TEXT AND NUM INFO #####
#####

```

Figure 46. Case List Display

```

#####
/
/1 SLIGHT VIBRATIONAL PROBLEM          #####
/2 REASONABLE NOISE LEVELS            #####
/3 CHAMBER TEMPERATURE FLUCTUATIONS   #####
/4 FUEL PUMP PRESSURE VARIATIONS       #####
/5 GOOD THEORETICAL PERFORMANCE COMPARISON #####
/6                                     #####
/7                                     #####
/8                                     #####
/9                                     #####
/10                                    #####
/11                                    #####
/12                                    #####
/13                                    #####
/14                                    #####
/15                                    #####
/16                                    #####
/17                                    #####
/18                                    #####
/19                                    #####
/20                                    #####
#####
CLICK ENG NAME AND SEL CASE BUTTON FOR TEXT AND NUM INFO #####
#####

```

Figure 47. Case One Comments

To obtain the numerical information and performance recommendations made for a case, the user selects the "Select Case Number" field. Doing so, the number of the case to be examined is requested (Figure 48). The user selects case number one and the desired information is displayed (Figure 49). Pressing any key returns the user to the archival file retrieval list. To return to the database operation menu, the user may press the "Ctrl-Break" (Figure 41). The user has now completed the post-test analysis of the tested rocket and added to its life history. Selecting the final option of the menu returns the user to the main menu (Figure 21). Selecting the final option there causes the temporary files to be deleted and the expert system is exited.

B.14 Applications with Simulated Data

This completes a typical run through using the constructed expert system. As a real world application, operating the expert system with simulated data files could be a way of training those who will watch the monitors during a test. A variety of problems could be introduced into carefully thought out data files, or files from previous tests, to teach operators trends in test problems to watch for, procedures for

```

ENTER THE CASE NUMBER: 11
TYPE 'A' FOR DISPLAY OF ALL CASES

```

Figure 48. Detailed Case Information Selection

ROCKET ENGINE NUMBER 1 TEST DATA

```

ENGINE:                H_1A_VERSION_2_TEST_10
ENV. TEMP.:            62 DEG. F      COOLANT FLOW RATE: 325.68 LB/S
GG PRESSURE:           671.49 PSIA    TC FUEL FLOW:      267.59 LB/S
GG TEMPERATURE:       900 DEG. F     TC OXID FLOW:      623.23 LB/S
TB STATIC PRESS.:     34.966 PSIA    GG FUEL FLOW:      13.212 LB/S
TP SPEED:              4100 RPM       GG OXID FLOW:      4.5979 LB/S
TP LUB. CONSUM.:       5 GPM          NOZ EX STAT PRESS.: 14.5 PSIA
FP INLET PRESS.:       57 PSIA        MEASURED THRUST:   209700 LB
FP INLET TEMP.:        60 DEG. F     FUEL QUANTITY:     40000 LB
FP OUTLET PRESS.:      255 PSIA       OXID QUANTITY:     90000 LB
FP OUTLET TEMP.:       80 DEG. F     TC NOZ STAG PRESS.: 635.86 PSIA
OP INLET PRESS.:       65 PSIA       O/F RATIO:         2.32904
OP INLET TEMP.:        -290 DEG. F   TOTAL FLOW RATE:   890.819 LB/S
OP OUTLET PRESS.:      310 PSIA      SPECIFIC IMPULSE:   235.401 SEC
OP OUTLET TEMP.:       -235 DEG. F   EFF. EXH. VEL.:    7579.915016 FT/SEC
TC INJECTOR PRESS.:    653.03 PSIA   THRUST COEFF:      1.5340
TC TEMPERATURE:       2000 DEG. F    CHAR. EXH. VEL.:   4941.162596 FT/SEC

RECOMMENDATIONS SINCE START OF TEST: INCREASE_PROPELLANT_FLOW
GOOD_NOZZLE_DESIGN                  GOOD_FUEL_PUMP_INLET_PRESSURE
INCREASE_INJECTOR_PRESSURE          GOOD_OXID_PUMP_INLET_PRESSURE
PRESS ANY KEY TO CONTINUE

```

Figure 49. Case Number One Information

when safety limits are exceeded, etc. As a result, this exercise for demonstration of the expert systems capabilities has shown yet another application for the system. All computer files utilized in the operation of the expert system are listed in figure 50 in regard to their affiliation with the main expert system control menu.

FIGURE 50. EXPERT SYSTEM FILES

FILES FOR ROCKET DESIGN AND SAFETY LIMIT PARAMETERS

Basica.Exe	Basic compiler
Clock.	Computer internal clock time
Eperf.	Stores user-defined performance values
Expperf.	Stores all measured/calculated performance
Values	
GGflag.	Flags the gas generator flow when dumped into the nozzle
Limits.	Stores all safety limits
Perf.	Stores all calculated performance values
Performance.bas	Calculates performance values
Rocket.kbs	Obtains rocket design and safety limit parameters from user
Starter.kbs	Starts elapsed test time count
Timekeep.bas	Retrieves internal clock time
Control.kbs	Main menu display and options

FILES FOR EXPERT SYSTEM CONTROL MENU

Blank.kbs	Dummy file for exit routine
Clean.bat	Deletes unwanted temporary files
Control.kbs	Main menu display and options
Link.kbs	Checks for reading errors
Kill.bat	Deletes all temporary files
Menus.txt	Stores hypertext for all menus
Mlink.kbs	Returns user to test monitor routine
Rmenus.txt	Hypertext for various main menu options
Rlink.kbs	Returns user to the main menu
Timepass.bas	Calculates time passed
Timer.	Stores current time
Tpass.	Stores elapsed time
Tstart.	Stores test starting time

FILES FOR "UPDATE ROCKET TEST PARAMETER FILE" MENU OPTION

Basica.exe	Basic compiler
Calc.bas	Calculates performance parameters

Data.	Stores ASCII file data	138
Data1.	Stores archive file data 1	
Data2.	Stores archive file data 2	
Data3.	Stores archive file data 3	
Lk2.kbs	Checks for input errors	
Mclean.bat	Deletes unwanted temporary files	
Mon2.kbs	Monitors second simulation file	
Mon3.kbs	Monitors third simulation file	
Monitor.kbs	Monitors first simulation file	
Sim.bat	Calls first set of simulated data	
Sim2.bat	Calls second set of simulated data	
Sim3.bat	Calls third set of simulated data	
TD1.bat	Copies first set of simulated data into ASCII file	
TD2.bat	Copies second set of simulated data into ASCII file	
TD3.bat	Copies third set of simulated data into ASCII file	
Tdata1.	Stores simulated data 1	
Tdata2.	Stores simulated data 2	
Tdata3.	Stores simulated data 3	
Timepass.bas	Calculates time passed	
Timer.	Stores current time	
Tpass.	Stores elapsed time	
Tstart.	Stores test starting time	

FILES FOR "DISPLAY CURRENT ROCKET TEST PARAMETER VALUES" MENU OPTION

Data_display.kbs	Displays selected archive file parameter values
Timepass.bas	Calculates time passed
Timer.	Stores current time
Tpass.	Stores elapsed time
Tstart.	Stores test starting time

FILES FOR "COMPARE EXPECTED VALUES TO MEASURED VALUES" MENU OPTION

Compare.kbs	Compares expected values to measured values
TD1.bat	Copies first set of simulated data into ASCII file

TD2.bat	Copies second set of simulated data into ASCII file
TD3.bat	Copies third set of simulated data into ASCII file
Tdata1.	Stores simulated data 1
Tdata2.	Stores simulated data 2
Tdata3.	Stores simulated data 3
Expperf.	Stores expected performance parameter values
Test_data.	Stores retrieved archive file information

FILES FOR "COMPARE TEST PARAMETERS TO ENGINE DATABASE" MENU OPTION

Dbcomp.kbs	Compares database extremes to measure values
TD1.bat	Copies first set of simulated data into ASCII file
TD2.bat	Copies second set of simulated data into ASCII file
TD3.bat	Copies third set of simulated data into ASCII file
Tdata1.	Stores simulated data 1
Tdata2.	Stores simulated data 2
Tdata3.	Stores simulated data 3

FILES FOR "PERFORMANCE OPTIMIZATION ADVICE" MENU OPTION

Optimize.kbs	Provides advice for performance optimization
TD1.bat	Copies first set of simulated data into ASCII file
TD2.bat	Copies second set of simulated data into ASCII file
TD3.bat	Copies third set of simulated data into ASCII file
Tdata1.	Stores simulated data 1
Tdata2.	Stores simulated data 2
Tdata3.	Stores simulated data 3
Expperf.	Stores expected performance parameter values
Test_data.	Stores retrieved archive file information

FILES FOR "ROCKET ENGINE LIFE-HISTORY DATABASE OPERATIONS"

MENU OPTION

Basica.exe	Basic compiler
Casenum.txt	Stores case number listings
Case_info.dbf	Stores numerical information
Case_info.	Stores hyertext information
CL.dbf	Back up for case_info.dbf
Clink.kbs	Returns user to main menu
Combine.bat	Combines files into one hypertext file
Counter.	Stores number of cases in database
CSE.kbs	Adds test case to database
Dbout.kbs	Displays database information for a selected case
DE.kbs	Deletes a selected database case
Delchange.bat	Copies backup into current file
Extremes.dbf	Stores minimum and maximum parameter values
Header.bas	Creates hypertext headers
Htext.bas	Stores hypertext test comments
Hypermaker.bas	Modifies parameter format for hypertext
MCL.dbf	Master case information database file
MPL.dbf	Master page information database file
Page.bas	Stores engine name and summary comments in database
Pagehdr.	Stores page name in hypertext format form
Pagemaker.bas	Modifies page name for hypertext format
Page_list.	Stores case information by page name for scrolling purposes
Page_list.dbf	Stores page name and test cases within the database
PL.dbf	Backup page list database
Perror.	Checks database selection for error
Scroll.	Flags current scrolling screen
Testcase.txt	Stores the numerical and graphical information for a selected archive

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Peter Andrew Montgomery was born April 7, 1967 in Spokane, Washington. He attended the Boston University College of Engineering and graduated in May 1988, with the degree Bachelor of Science in Aerospace Engineering. While at B.U., Peter worked for two years as a teaching assistant in the area of expert systems and their applications to aerospace engineering. He also developed an interest in the area of propulsion, and undertook a senior project in ion propulsion systems.

To pursue his goal of designing rocket propulsion systems, Peter decided to seek an advanced engineering degree. He entered the University of Tennessee Space Institute in September 1988 and accepted a graduate research assistantship with the Arnold Engineering Development Center (AEDC)/Sverdrup Analysis Branch. The topic of his research was an expert system for propulsion test facility diffuser design. The approach taken to solve this problem was the same approach utilized, however, in solving the problem of this thesis. In May 1990, Peter completed his studies at the Space Institute and received a Master of Science degree in Aerospace Engineering.

Peter is currently considering an offer by the University of

Washington Department of Aeronautics and Astronautics to continue his studies and work towards a Ph.D. in Aerospace Engineering. He plans to get more into research work, especially related to rocket propulsion.