Technology Assessment Tool Development

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Phase 2

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FOREWORD

The study entitled, "Technology Assessment Tool Development, Phase 2," was performed by Boeing Missiles and Space, Huntsville for the George C. Marshall Space Flight Center (MSFC). The activities were carried out during the period of June 1992 through November 1992. Boeing's Project Manager was Irwin E. Vas and the MSFC Contracting Officer's Technical Representative was C. Frederick Huffaker. Technical support was provided by J. McGhee.

ABSTRACT

A technology assessment tool has been developed to predict the characteristics of performance, cost, and schedule of a Lunar Transport System (LTS) and the risks associated with these characteristics and their prediction methodology. The primary purpose of this tool is to provide Project Managers and Technologists with a quick complete evaluation of the effect of an advanced technology on a LTS. The current tool is in a developmental stage and will be further improved as the scope of the assessment broadens. The current model has focused development on the advancement of Cryogenic Fluid Management and Propulsion. The working model, while in need of further upgrades and additions, has proven to be a useful tool to perform technology trades and performance sizing. The Technology Assessment Tool is designed to be hosted on a personal computer and operates readily on either IBM or MacIntosh format.

ABBREVIATIONS AND ACRONYMS

Al	Aluminum
Al-Li	Aluminum-Lithium
CFM	Cryogenic Fluid Management
CH_4	Methane
CRV	Crew Return Vehicle
delta-V	Velocity Change (m/s or km/s)
DOD	Department of Defense
ETO	Earth-to-Orbit
g	Acceleration in Earth Gravities (acceleration 9.80665m/s^2)
GR-3.0D	Ground Return 3-stage Dual crew module
HLLV	Heavy Lift Launch Vehicle
HLV	Heavy Launch Vehicle
IMLEO	Initial Mass in Low Earth Orbit
Isp	Specific Impulse (=thrust/mass flow rate)0
ISTS	Integrated Space Transportation System
kg	Kilograms
kg/m^2	Kilograms per square meter
kg/m^3	Kilograms per cubic meter
Klb	1000 pounds
km	Kilometers
km/sec	Kilometers/second
kWe	Kilo-watt Electric
lb	pounds
LEO	Low Earth Orbit
LEV	Lunar Excursion Vehicle
LH_2	Liquid Hydrogen
LLO	Low Lunar Orbit
LO_2	Liquid Oxygen
LOI	Lunar Orbit Injection
LOR	Lunar Orbit Rendezvous
LTS	Lunar Transfer System
LTV	Lunar Transfer Vehicle
m	Meters
m/sec, m/s	meters/second
mt	metric ton (1000 kg)
MLI	Multi-Layer Insulation
MSFC	Marshall Space Flight Center

ABBREVIATIONS AND ACRONYMS (Concluded)

N	Newton, Kilogram-Meters per Second Squared
n. mi	nautical mile
NASA	National Aeronautics and Space Administration
NASP	National AeroSpace Plane
NLS	National Launch System
P	Pressure
Pa	Pascals
PCM	Parametric Cost Model
psi	Pounds per Square Inch
q	Heat Flux (Watt per Square Centimeter)
Q	Heat Flux (Joules per Square Centimeter), Radiation Quality Factor
RCS	Reaction Control System
s, sec	Seconds
SSME	Space Shuttle Main Engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
STME	Space Transportation Main Engine
t	Metric Tons (1000kg)
TEI	Trans-Earth Injection
TLI	Trans-Lunar Injection
TPS	Thermal Protection System
T/W	Thrust to Weight Ratio

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APPENDICES

FIGURES

1. SUMMARY

A tool has been developed which has the capability to assess the value of a technology as it applies to the Integrated Space Transportation System (ISTS). Based upon inputs on technology advancements and mission needs provided by the Technologist and Program Manager, a methodology is derived to quantify the risks in performance, mission, schedule and cost. This technique also has the capability to provide an evaluation of innovative and evolutionary technologies and their impacts on mission goals. A technology ranking can therefore be derived. In the initial phase of the study, a preliminary modeling tool was developed and applied to evaluate the features of two technologies (Cryogenic Fluid Management and Propulsion) as they relate to a single alternative of the ISTS comprised of an HLLV core, lunar transportation system and chemical/LH_2 return. For the lunar transportation system, three modes were examined, and for the "return" system, two propulsion alternatives (cryogenic and storable) were evaluated. In this, the second phase of the study, the modeling tool has been further developed with the addition of more sophisticated evaluation routines. During the second phase of this study, the effort has primarily been focused on Cryogenic Fluid Management, specifically the evaluation of multi-layer insulation (MLI) and fluid venting (low-g vs. propulsive settling).

Current accomplishments of this project have been the development of a methodology to assess the characteristics of a technology system. The primary questions of what technologies to assess and how to assess technology have been identified and approached. A demonstrative model for proof of concept of assessment methodology has been developed. Capability to perform this type of task for a large development project has been demonstrated on a limited scale.

Future accomplishments of this project will be in refining the work already started. Broadening of project scope to include a high precision model of applicable known technologies is a long term goal. In the near future the addition of a more precise cost and mass model is desired. Also the constant evolution of the methodology and assessment process is desired.

The technology assessment tool has been developed to provide a rapid quantitative evaluation of various technologies. For ease of operation and widespread compatibility, it has been designed to operate on a desktop computer. Enhancements to the tool may be performed readily, as information on developing and proposed technologies becomes available. Thus, the tool is flexible and can stay abreast with current technology.

2. INTRODUCTION

Over the past 3 decades, considerable efforts have been expended in space exploration by NASA. On the 20th anniversary of the Apollo Moon landing, of 20 July 1969, President Bush announced his Space Exploration Initiative. A series of short-term studies were conducted to examine the architecture options for lunar and Mars exploration. The architecture has been defined in several manners. In the STCAEM study (ref. 1), the architecture has been defined in terms of the principal transportation propulsion system utilized. More recently, the Stafford Commission has proposed a series of architectures which are in terms of functional elements to meet mission goals (ref. 2). The system and subsystem performance required to fulfill the mission is intimately connected with the technologies required to fulfill the task. The evaluation of these technologies to comprise the system has been performed through use of trade and technology studies which lead to a selection of technologies to meet the program and mission goals and which are based on the judgment of importance, availability, and risks. These technology assessments are performed for specific applications, take a significant time to complete, and have limited flexabilities with parameters which are not quantified. Redirection of the technology is neither easy nor cost effective because of the extent of the work that has already been completed.

There have been a few studies aimed primarily at technology assessment and evaluation. Some of these have been very well done, and have provided valuable guidance to technology priorities and performance goals. However, they have tended to take too long, especially when time to initiate them is included, and usually have not included quantitative risk assessments. It is impractical to issue a study contract every time a technology assessment is needed; usually the assessment is needed on a short time scale entirely incompatible with the process of procuring and conducting a study.

The study reported herein was a continuation of an effort to assess the feasibility of constructing a technology assessment methodology based on algorithms which could be implemented on a commercial spread-sheet for desktop computers. If this could be done, it would be possible to create a methodology that could be used to perform a technology assessment in a day or two, after estimates of technology performance are obtained from technology experts. The motivation is to enable transportation technology assessments to be performed by NASA technology and program managers as needed.

The algorithms in question are often used in conceptual design studies for estimating mass and performance of transportation systems. In addition, probabilistic algorithms were proposed that would permit rapid graphical display of the performance and cost risks associated with typical uncertainties in performance achieved by technology advancements and in weights and performance estimates for conceptual and preliminary designs of space transportation systems.

Since the approach was novel and experimental, it was decided to explore and demonstrate its feasibility thorough small initial steps with a large measure of review, introspection, and evaluation of progress. This was done by two purchase-order contracts; key features of the approach have been demonstrated and useful results obtained relative to lunar transportation options. We have shown that a spread-sheet methodology can rapidly generate representative performance and cost estimates, including system performance comparisons over a range of technology performance and risk scattergraphs. Assessment of a range of cryogenic fluid management technology levels showed that a cryogenic lander and return stage for lunar transportation could deliver major performance advantages with technology advancements already demonstrated in laboratory tests, and that risks are modest.

As we have observed during the past few years, with very rapid development in technology, there is a need to examine the value of the technology to meet the mission goals in a rapid method and to assess its capabilities in a quantitative fashion. The current study is a second phase of an effort to develop a tool to quantify the risks associated with performance, mission, schedule, and cost by which technologies may be assessed based upon input provided by the Technologist and the Program Manager from which a decision could be based and quantified.

3. BACKGROUND

Technology assessments have generally been carried out for specific applications. These have been brought about by input from the Technologist, Program Manager, and a variety of Delphi-type teams. This process, even though thorough, is time consuming and, in fact, quite rigid in exploring and attempting to optimize upon a single system or system derivative. As a significant time is required to complete the assessment, there is a possibility that the conclusions may be outdated by the time the process has run its course. With programs as broad as those under the NASA sponsorship, the value placed upon a specific technology is difficult to quantify within the guidelines of the entire organization. With the inclusion of DOD within some NASA programs, this becomes increasingly difficult to accomplish.

The objective of the current study is to develop a technology evaluation aid which provides a rapid, flexible method for comparison and analysis of technology with respect to architectures, mission and program needs. The technique would also provide an evaluation of innovative and evolutionary technologies to determine pertinent issues, trends, and program implementation strategies. From this, one can obtain the ingredients for a quantitative hierarchy ranking of the technologies. The outcome is the understanding of the implication of the technologies and capabilities to meet mission requirements.

4. METHODOLOGY

In the first phase of this study, Reference 3, the purpose was to develop an assessment tool and to demonstrate it on technologies as they relate to an Integrated Space Transportation System (ISTS). The system tool was used to evaluate the technology based upon performance, payoff, cost, risk, and schedule. The tool was prototyped to demonstrate practical applications for a specific total transportation system mission. Initial implementation was illustrated on a desktop computer using a spreadsheet format.

The ISTS model for technology assessment was developed to use the Microsoft Excel spreadsheet on the Macintosh operating system. This tool can be readily transported with some manipulation to another operating system running Excel, or other fully functional spreadsheet. The spreadsheet format was chosen for its simplicity of user interface as opposed to its reduced processing speed. The choice of spreadsheet implementation allows a new user to become proficient in using the tool with minimal instructional or operational time. This benefit is a major advantage over a high-level programming language which requires a user to possess proficient programming skills for operation. The disadvantage of slower execution speed may become more apparent as the tool grows in complexity, at which time the need to upgrade to a more efficient compiler may be deemed necessary. However, hardware advances may also offset the software deficiency.

The Integrated Space Transportation System is only one of several parts of the NASA mission. It is comprised of three major elements: earth-to-orbit transportation system, the space transportation vehicle, and the return vehicle (fig. 4-1). In each of the major elements of the ISTS, several options are available. The current analysis has considered one option in each major element, namely, the heavy lift launch vehicle core, lunar transportation system, all chemical and chemical/LH_2 return. Even though the methodology is generic and may be applied to any system, the focus will be in these particular areas.

The issues that are examined in the technology assessment are illustrated in figure 4-2. Cost is tied directly to the risk analysis and schedule. It is essential to identify the payoff for developing a specific technology versus an alternative and using it for an identified mission program. The leveraging of this technology to other programs is also to be considered. The payoff depends upon a technology performance and systems model and is considered with the risk involved in technology development (fig. 4-3).

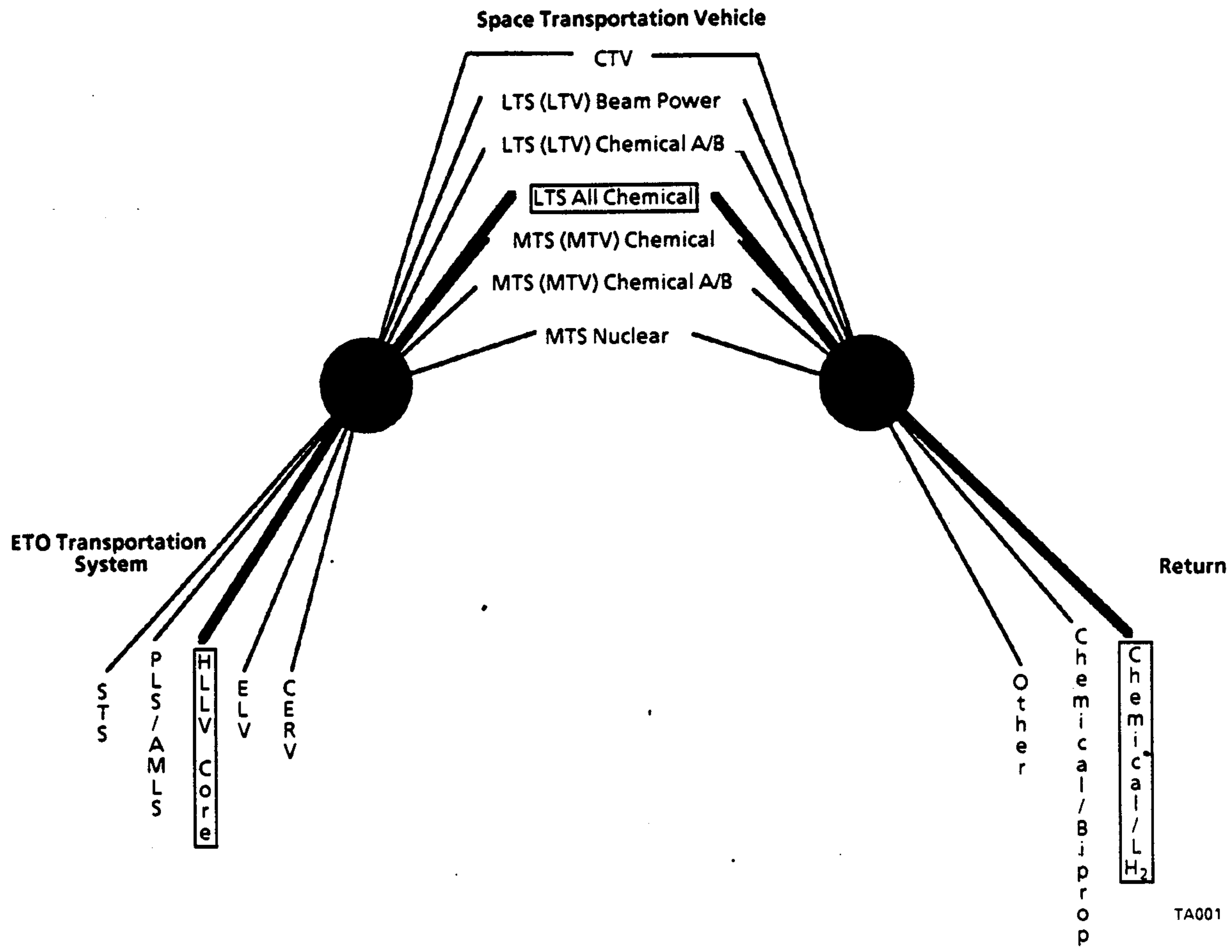


Figure 4-1. Integrated Space Transportation System

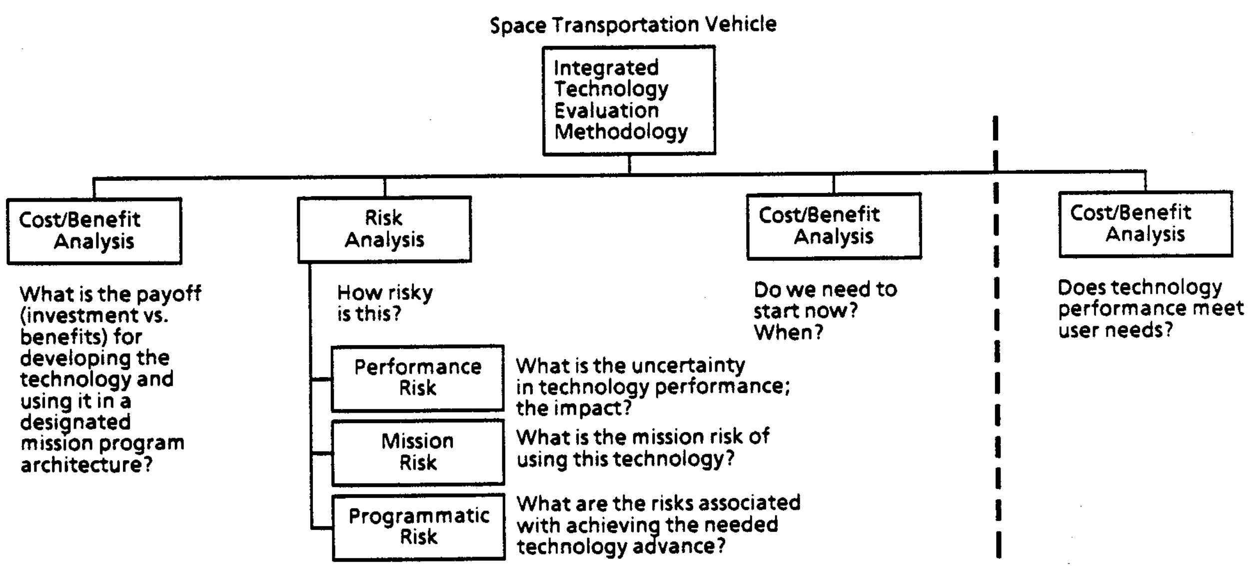


Figure 4-2. Technology Evaluation Methodology Overview

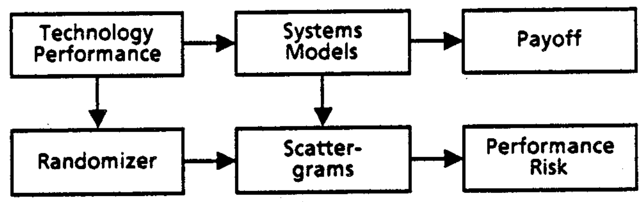


Figure 4-3. Overall Method for Payback

Commencing with technology advancements and mission needs information which is provided by Technologists as well as Program Managers and Senior Staff, the cost benefit methodology then goes through a cycle which includes vehicle and propulsion system performance, vehicle costs, and eventually gets to a cost benefit ratio (fig. 4-4). For each of these characteristics, such as vehicle performance model, the range of characteristics of vehicle performance is provided by the Technologist and verified through the Delphi process. The spreadsheet model itself can be updated as required by new inputs provided by experts.

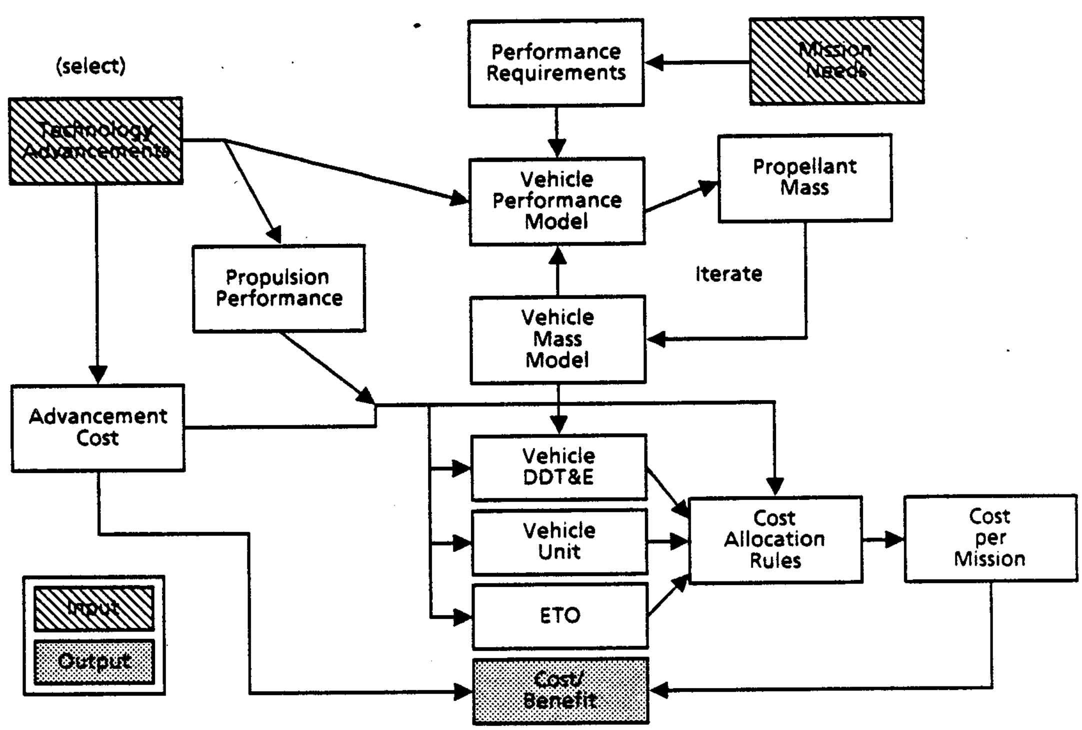


Figure 4-4. Cost/Benefit Methodology

Risk analysis is carried out for performance, mission and programmatic risk. In all cases, quantification of characteristics of the performance is provided by the Technologist and the Program Managers. The performance risk depends upon technology performance and system models based upon inputs provided by the experts. The results are obtained relating the performance to the cost or any other desired parameter. Initial mass in low-earth orbit is one of the parameters utilized to gage performance.

In a similar fashion, mission risk is identified and quantified relating to this specific technology to meet mission demands. Changes or improvements in the technology would impact the mission risks of the technology. A risk is also associated with the technology as it relates to the entire program. Again, modifications to the technology would impact the program achieving its goals within a stated time.

Funding for technologies to achieve certain mission goals or performance characteristics is generally specified early in the program. Changes in schedule of mission goals would impact the total cost of the program. Such changes should be implemented as soon as possible. The schedule analysis would identify the impacts of these changes in the program's lifetime.

The methodology developed considers current as well as innovative technologies, the issues, the trends for these technologies and provides the implications and quantitative priorities of these technologies as they relate to mission goals. These results come about following a combination of the inputs of the Technologists, Experts, Program Managers, and those directly involved with the program.

5. ASSESSMENT FLOW METHODOLOGY

In the process of developing the Technology Assessment Tool, we have experimented with the methodologies necessary to gather and process the information necessary for calculation. Our goal is to develop a uniform methodology to approach the problem of assessment and to document this process for a single technology and limited parameters; so that as we further the technology assessment process and incorporate other avenues, whether they be differing technologies or missions or other parameters, we will have a defined method to deal with the new problem. During the first two phases of this project, we have determined the need for a use of a Delphi-type process to be incorporated in conjunction with an Analytical Hierarchy Process (AHP).

Our main focus during Phase 2 towards developing the methodology has been setting up the decision hierarchy by breaking the decision problem into a hierarchy of interrelated decision elements. This seemingly simple task can become somewhat formidable when applied to a project the size and complexity of a space transport vehicle. By focusing on a single limb of this hierarchy tree, we can attempt to devise and explain a simplified flow diagram.

For our project, we selected to focus on cryogenic fluid technology and, even more specifically, the MLI technology as it applies to the performance of the space transportation vehicle.

On the bottom level of figure 5-1, we see the three levels of technology that we have chosen to define: current, intermediate, and advanced. Technology, as it refers to multi-layer insulation, generally means varying thicknesses of MLI that can be applied to a cryogenic tank. Currently, only three layers or approximately 1/16 in of MLI may be applied to the wall of a tank. For an intermediate technology application, we used 1 in of MLI, and 2 in for advanced technology . On the second level, we see MLI and its sister technologies: vapor-cooled shields and foam. A little regression shows that the sister technologies can also be broken out into technology levels which can be defined. Stepping up to the third level shows these technologies to be part of the insulation system. At this level, we also find sister cryogenic systems that can be broken out into similar branches. The next level requires analysis to be performed to determine how the chosen system (insulation) effects certain aspects of performance such as IMLEO, sensitivity, mission flexibility, safety.

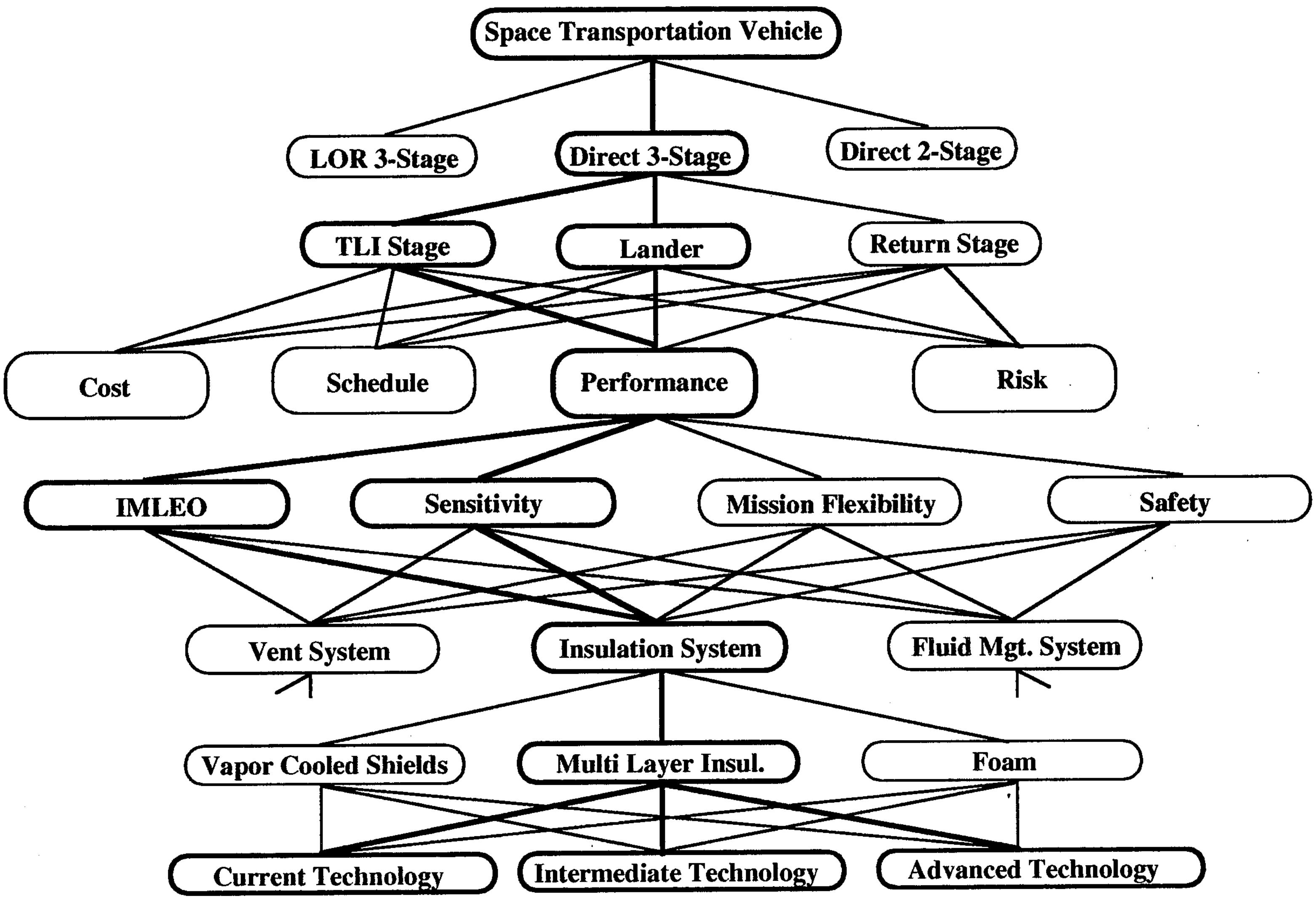


Figure 5-1. Simplified Performance Flow Diagram

The Insulation Systems' effect on IMLEO can be ascertained in the assessment tool by use of Lockheed's boiloff equation to determine technology's specific performance and general geometry to assess the mass requirement. By use of the generalized rocket equation, we can determine the overall performance parameter effect on IMLEO. Sensitivity can be determined using the probability routine incorporated into the assessment tool. Mission Flexibility and Safety will be discussed at a later date.

The next level shows us that these parameters are all related to the assessment parameter performance. Also on a level equal to performance are the assessment parameters cost, schedule, and risk. Each of these parameters have a similar breakout as the simplified version shown for performance. These assessment parameters can all be related to the next level which incorporates the vehicle stages. For this model, the Direct 3-Stage vehicle is represented; it should be noted that each of the staged mission modes has a similar breakout. These mission modes can then be grouped under the heading Space Transportation Vehicle.

Thus, by carefully following the decision flow from the bottom to the top, one is able to ascertain the effect of a single technology (MLI) on the complete Space Transportation Vehicle. Using this same methodology, a similar breakout can be designed for various technologies and assessment parameters.

We have now demonstrated the basic methodology that will be used to incorporate new technologies and parameters into the assessment tool. The analytical models used to assess the technology will be designed in accordance with this flow model.

6. ASSESSMENT TOOL UPGRADES

During the initial phase of the the Technology Assessment project, a tool was developed which, while being very useful and adaptable, was basic in its application capability. A major portion of the Phase 2 project has been to provide updates to the tool. The upgrades include both programming improvements geared toward more accurate solution processes and user interface improvements which make the tool more versatile and applicable toward increasingly specific technology trades.

The first desired upgrade performed was the addition of more selectability in the propulsion-type specification, so that mission portions can be more independently defined. The Technology Assessment Tool developed in the first phase of the project only allowed for a global selection in propulsion type for the lunar transport vehicle. An upgrade was performed during Phase 2 to allow the user to independently select the fuel type and technology level for each engine cluster, thus determining the Isp for each stage of each mission mode. Now, each engine cluster can be independently defined as opposed to all engine clusters having identical characteristics.

A similar upgrade was performed on the mission duration aspects of the technology assessment tool. In the earlier work, the Technology Assessment Tool had the ability to set the mission duration globally for the the entire lunar transport vehicle. In order to perform a reasonable trade for cryogenic technology, a more specific application of mission duration is necessary. This is due to the sensitivity of cryogenic boiloff to time. In order to fulfill this critical need, the tool was updated during Phase 2, so that each stage of each mission mode could have the mission duration independently defined.

Another input problem approached in Phase 2 was the addition of selectability in the Cryogenic Fluid Management variables. During Phase 2 of the technology assessment, the user is given the ability to select Multi-Layer Insulation thickness for each tank set on each mission mode. This selection ability allows the user to optimize the MLI for each stage of the vehicle rather than optimize the vehicle globally. Thus, the user can produce results that more closely reflect the value of MLI.

The ETO vehicle was given a major renovation during the Phase 2 portion of the technology assessment project. The Phase 1 ETO model was an IBM PC/AT based C++ model. The compiler, while possessing significantly faster processing capability, was far less user friendly than the preferred compiler Excel. Therefore, the ETO model was converted from a C++ standard program to an Excel spreadsheet program.

During the Phase 1 portion of the the project, the ETO model was based on a cryogenic core with two cryogenic boosters. For the Phase 2 portion of the project, an ETO model based on a cryogenic core with F-1A boosters was added. Both of these ETO models are currently available on Excel spreadsheet format. An accompanying macro sheet is also available for performing sensitivity trades using the ETO models.

A major product of the technology tool (aside from direct technology trades) is the ability to assess vehicle sensitivities to technologies. A major update to the probability (sensitivity) generator for the tool has been performed in Phase 2. During Phase 1, probability generation was possible with only Isp, primary & secondary structures, and ETO transportation cost, with full probability control on only Isp. During Phase 2, full probability generation control has been developed for the Isp of each stage of the vehicles (assuming the mission modes have similar propulsion stages), for primary and secondary structures, for ETO transportation cost, for settling delta velocity, and for MLI insulation thickness. Probability generation can be performed for a single or any combination of items using the accompanying probability macro.

For Phase 2 of the project, the sensitivity generation of the ETO model has also been expanded. In the ETO model, full probability generation is now available for the ascent delta velocity, booster Isp, core Isp, booster engine thrust-to-weight ratio, core engine thrust-to-weight ratio, booster engine cost, core engine cost, and shroud weight. As in the lunar transport vehicle model, probability generation can be performed for a single item or any combination of items using the accompanying probability macro.

It should be noted that each of the upgrades to the Phase 1 tool that were performed under Phase 2 are complete upgrades designed to improve the program-to-user interface of the program. It may be deemed necessary to provide further upgrades to the tool as work progresses.

7. LUNAR TRANSPORTATION SYSTEM

The purpose of technology assessment is to meet the trade study needs of the space exploration community. In order to meet this need, a modeling of currently discussed alternatives should be represented in the Technology Assessment Tool. The purpose of this task is to determine and select likely lunar mission transportation alternatives.

With the help of the COTR, the following Lunar Transport System (LTS) scenario(s) have been selected to be represented in the Technology Assessment Tool. The first portion of the lunar mission is the launch to Low Earth Orbit (LEO) of the Space Transport System (STS). This will be the mission of a Heavy Lift Launch Vehicle (HLLV), which will be modeled generically enough that as part of the Technology Assessment, a trade will be performed to determine the most viable HLLV configuration.

The next portion of the lunar mission is the transportation from LEO to the Moon and back to Earth. This portion will probably require a closely integrated vehicle which will be discussed as a mission "mode" of travel. Currently, there are three distinct mission modes receiving dominate attention in the space exploration community. These three modes: 3-stage Lunar Orbit Rendezvous (LOR) (formerly referred to as Ground Return 3-stage dual crew module, GR-3.0D, during Phase 1 of the Technology Assessment), 3-stage Direct and 2-stage Direct, will be represented in the Technology Assessment Tool. Trade studies will be performed to aid the process of selecting a mission mode.

ETO. This is the primary launch vehicle which will be used to project the Lunar Transport Vehicle (LTV) into LEO. This vehicle consists of a cryogenic core and two chemical boosters. The core of the vehicle is powered by four Space Shuttle Main Engines (SSMEs) in the baseline configuration, the number of engines and operating characteristics being an input variable in the modeling program. Each booster is powered by three F-1A engines in the baseline configuration, the number of engines being variable in the program. This four plus six configuration is designed for a total lift to earth orbit of approximately 150 tons of payload.

Three-Stage LOR

<u>First Stage - Trans-Lunar Injection (TLI) Stage.</u> This stage consists of a large engine, or cluster of engines, and the fuel required to propel the vehicle from LEO to Low Lunar Orbit (LLO). This stage is discarded as the vehicle enters LLO after its function has been performed.

Second Stage - Lunar Orbit Injection/Trans-Earth Injection (LOI/TEI) Stage. This stage consists of a single or set of engines and fuel as required to perform Lunar Orbit Injection (LOI) and perform the Trans-Earth Injection (TEI). This stage houses the transfer crew module and is fully capable of independent action unlike the first stage. The second stage remains in LLO, with the transfer crew module manned by a single crewmember, while the other crewmembers perform the lunar surface mission. After the lunar surface mission is complete, the entire crew returns to the second stage for the TEI phase of the mission. The second stage returns the transfer crew module to LEO. After performing the TEI the second stage is discarded in LEO (where it can be reused) or in a trajectory towards the sun for disposal.

Transfer Crew Module. This is the crew module that the crew utilizes during the launch to LEO, the TLI, the LOI, and the ground return portions of the lunar mission. This module is based in the second stage of the LTV until the ground return portion of the mission is performed. It provides suitable life support and shielding for up to six crewmembers. During the lunar surface phase of the mission, this module supports the single crewmember left in lunar orbit. Upon return to Earth, this crew module provides aerobraking and thermal protection during reentry, and protection during splashdown.

Third Stage - Lunar Excursion Vehicle (LEV). This stage consists of a single or set of engines and fuel as required to perform the lunar descent/ascent. The third stage houses the lunar excursion crew module and is fully capable of independent action. The LEV performs the lunar landing, remains on the lunar surface during the lunar surface mission, and then performs the lunar ascent to LLO where it docks with the second stage to transfer crew from the lunar excursion crew module to the transfer crew module. After crew transfer back to the second stage, the third stage is discarded in LLO.

Lunar Excursion Crew Module. The lunar excursion crew module is used in conjunction with the third stage of the LTV. This crew module is fixed to the third stage and is capable of supporting up to five crewmembers during the lunar descent/ascent and the lunar surface portion of the mission. The lunar excursion crew module is expended in LLO with the third stage after the crew is transferred to the transfer crew module and second stage.

Three-Stage Direct

First Stage - Trans-Lunar Injection (TLI) Stage. This stage consists of a large engine or cluster of engines and the fuel required to propel the vehicle from LEO to LLO. This stage is discarded as the vehicle enters LLO after its function has been performed.

Second Stage - Lunar Lander. This stage consists of a single or set of engines and fuel as required to perform the lunar descent. The second stage houses the transfer crew module and third stage and is fully capable of independent action. Upon lunar ascent the second stage is discarded on the surface of the Moon.

Third Stage - Lunar Ascent/Trans-Earth Injection (TEI) Stage. This stage consists of a single or set of engines and fuel, as required, to perform lunar ascent and TEI. The third stage houses the transfer crew module and is fully capable of independent action. After performing the TEI, the third stage is discarded in LEO (where it can be reused) or a trajectory towards the sun for disposal.

Transfer Crew Module. This is the crew module that the crew utilizes during the launch to LEO, the TLI, the lunar landing, TEI, and the ground return portions of the lunar mission. This module is based in the third stage of the LTV until the ground return portion of the mission is performed. It provides suitable life support and shielding for up to six crewmembers. Upon return to Earth, this crew module provides aerobraking and thermal protection during reentry, and protection during splashdown.

Two-Stage Direct

First Stage - Trans-Lunar Injection (TLI) Stage. This stage consists of a large engine or cluster of engines and the fuel required to propel the vehicle from LEO to LLO. This stage is discarded as the vehicle enters LLO after its function has been performed.

Second Stage - Lunar Lander/Lunar Ascent/Trans-Earth Injection (TEI) Stage. This stage consists of a single or set of engines and fuel, as required, to perform the lunar descent, lunar ascent, and TEI. The second stage houses the transfer crew module and is fully capable of independent action. After performing the TEI, the second stage is discarded in LEO (where it can be reused) or a trajectory towards the sun for disposal.

Transfer Crew Module. This is the crew module that the crew utilizes during the launch to LEO, the TLI, the lunar landing, TEI, and the ground return portions of the lunar mission. This module is based in the second stage of the LTV until the ground return portion of the mission is performed. It provides suitable life support and shielding for up to six crewmembers. Upon return to Earth, this crew module provides aerobraking and thermal protection during reentry, and protection during splashdown.

For each technology considered, a methodology is established to assess the capability of the technology. The initial step was to establish the methodology for the specific technology in identifying the aspects of that technology that directly affect the technology assessment. These aspects include, but are not limited to, performance, cost, risk, and schedule. It is also necessary to identify the variables associated with these specific technologies with respect to the design outputs in performance, cost, risk, and schedule. The next step is to determine the relationships between the mission characteristics, technology variables, and the technology assessment. The technology assessment contains the relationships that apply to the variables and integrates them to other technologies and into one another where applicable.

For the technologies, a preliminary evaluation or ranking of the technologies options was given relating to the technology readiness and capability. For simplicity, the ranking has been refined to three major categories based on current development status of the technology which are:

a. Low implementation or current technology
b. Medium implementation or intermediate technology
c. High implementation or advanced technology.

Current technology is that technology which is available at the current time without any major development cost. It utilizes existing hardware or is fully documented (technology readiness level 6 or higher). An example may be an RL-10 engine. Intermediate technology is that technology which, while not currently available, will become available in the near future. It has been demonstrated at the component or subsystem level and is potentially acceptable for full-scale development risk. An example may be aluminum-lithium tanks. Advanced technology is that technology which is currently being proposed for further study; however, it will not be available in the near term. In this case, the principles have been demonstrated in laboratory tests or analogous applications. It is unlikely to be accepted for full-scale development prior to a technology advancement program. An example may be cryo-fluid management with multiple vapor-cooled shields.

8. APPLICATION

8.1 LUNAR TRANSPORTATION SYSTEM VARIABLES

As indicated in section 4, the methodology will be illustrated on specific elements of the Integrated Space Transportation System. For the ETO transportation system, the heavy lift launch vehicle is taken as the example. For the lunar transportation system, there are three mission profiles that may be considered, as shown in figure 8-1.

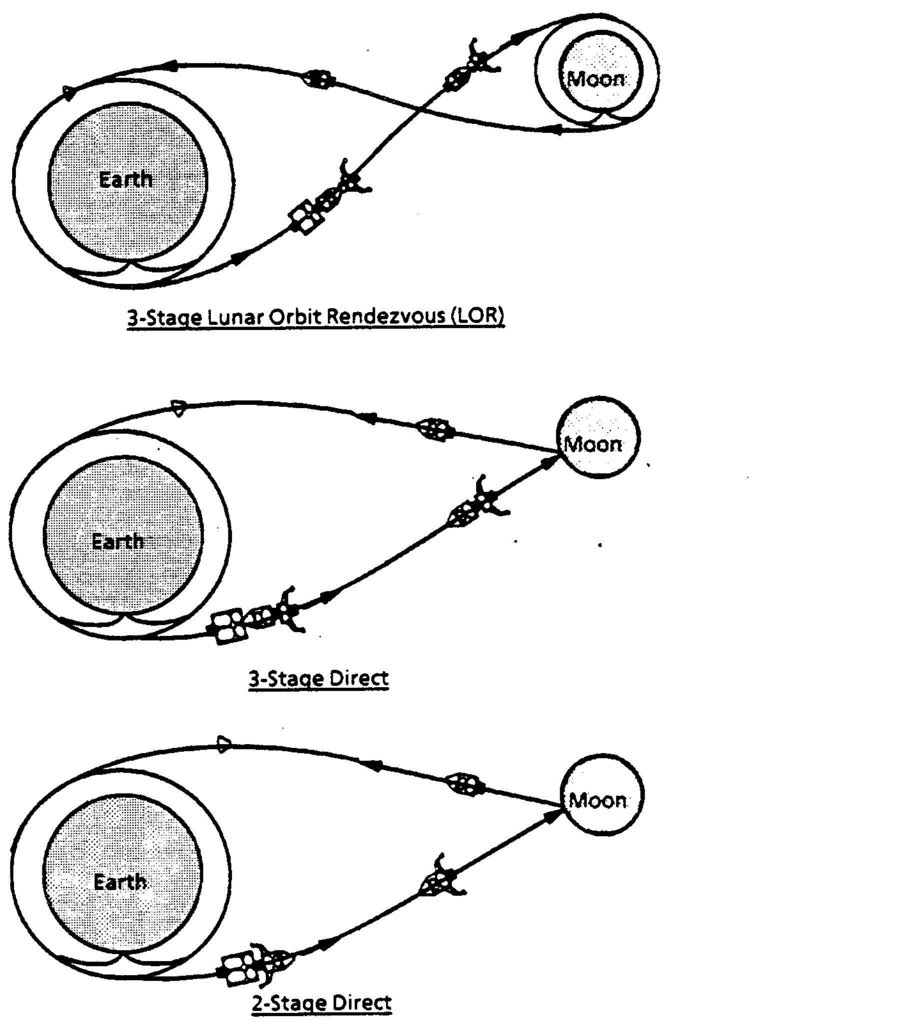


Figure 8-1. LTS Mission Profiles

Three missions were examined for the lunar transportation system: a Lunar Orbit Rendezvous 3-stage (LOR 3-stage), a direct 3-stage and a direct 2-stage. The differences between these three modes are shown in figure 8-2.

The methodology will be illustrated for the direct 3-stage with the range of characteristics shown in figure 8-3. The specific lunar transportation system input data

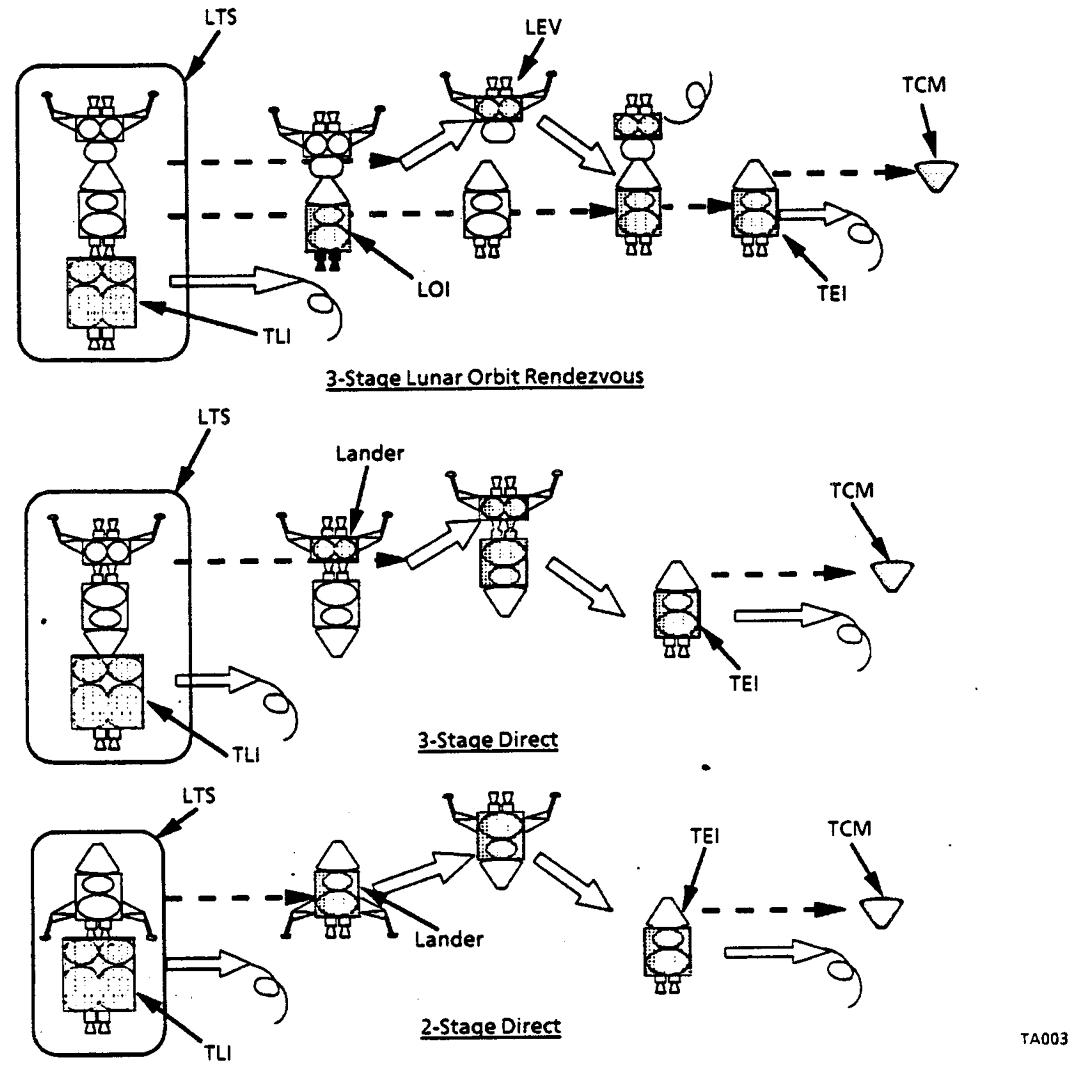


Figure 8-2. Lunar Modes Diagrams

is provided for each of the technologies (fig. 8-4). The complete spreadsheet for the LTS and "Return" vehicles is given in appendix A.

Settling Delta Velocity	0 - 340	Benefit of zero-g fluid venting
Multilayer Insulation Thickness	3 layers - 2 inches	Benefit of insulation technology
Return Fuel	Cryo - Storable	Cryogenic vs. Storable propulsion

Figure 8-3. Trade Parameters 3-Stage Direct

In addition, a selection is made for the level of the technology. For this particular illustration, we will assume as a baseline that the technology level for each of the technologies is zero ("0") or current technology. The value utilized for the weights and characteristics of the technologies is provided by experts in the field and available

	Current	Intermediate	Advanced
Main Prop.	Throttle RL-1	Operated RL-1	New engine
Tanks	Aluminum	Al-Li	Al-composite
Avionics	IUS/Centaur	SSF-class	Advanced
Elect. Power	.o.a. fuel cell	N/A	Adv. fuel cells
RCS	Bi-props	N/A	Integrated cryo
Pri & Sec St.	Aluminum	Al-Li	Composites

Specify Technology Level

Tanks	0	0 = Current
Avionics	0	1 = Intermediate
Elect. Power	0	2 = Advanced
RCS	*0*	
Pri & Sec St.	*0*	

INPUT

1. Vehicle Structures	
Legs fract.	0.04
Density alum. 2219 (kg/r)	2850
Density Al-Li 2195 (kg/rr.	2713
Density Comp. (kg/M³)	1852.5
Stress Alum. 2219 (PSI)	38,000
Stress Al-Li 2195 (PSI)	50,000
Stress Composite (PSI)	114,000
2. Cryo Fluid Mgmt.	
Ullage Factor	5%
Residuals Factor	2%
Mixture Ratio LO_2/LH_2	6
Mixture Ratio LO_2/CH_4	3
Mixture Ratio Storable	1.6
Density LO_2 (kg/m³)	1141
Density LH_2 (kg/m³)	71
Density CH_4 (kg/m³)	423
Storable Fuel (kg/m³)	800
Storable Oxidizer (kg/m³)	1500
Tank Pressure (PSIg)	35
3. Cryo Eng./Prop.	
Zero-base Isp LO_2/LH_2	450
Intermediate Isp LO_2/LH_2	465
Advanced Isp LO_2/LH_2	478
Zero-base Isp LO_2/CH_4	350
Intermediate Isp LO_2/CH_4	365
Advanced Isp LO_2/CH_4	380
Zero-base Isp Storable	320
Intermediate Isp Storable	340
Advanced Isp Storable	380
4. Veh. Avionics/Software	
kiloWatts	2
5. Aerobrake	
Brake fract.	0.2
6. Crew Modules & Sys.	
Transfer cab mass	8,263
LTV cab mass	8,000
LEV cab mass	4,000
7. ECLS	
8. Vehicle Assembly	
9. Orbit Launch & Checkout	
10. Vehicle Flight Ops.	
Basic Mission Requirements	
Earth G	9.80665
PI	3.1415927
Payload Del'd (kg)	5000
Payload Ret'd (kg)	0
% growth	12
LOR 3-Stage dVs	
TLI dV	3204
LOI dV	900
Landing dV	2100
Ascent dV	2000
TEI dV	1120
Post-Aero dV	300
Dir. Exp. dVs	
Booster dV	3084
Finite dV	100
Return dV	2850
TLI dV	3084
Landing dV	2950

INPUT (contd)

10. Vehicle Flight Ops.	
Cost	
ETO Transportation Cost	2500
LTV Dev Cost, $/kg	140000
LTV Unit Cost $/kg	20000
# missions amortize dev	10
Effective # of vehicle ret	3

LOR 3-Stage

OUTPUT

Mass Statement	
TEI	
TEI dry mass	13,408
TEI propellant	5,751
TEI mass	19,159
LEV	
LEV Orbit burnout mass	8,949
LEV ascent propellant	8,137
Ascent mass	17,086
Landing mass	23,243
Landing legs mass	930
LEV descent propellant	22,585
LEV gross	45,828
LEV total propellant	30,722
LEV inert mass	6,106
LOI	
MLOI	64,988
LOI propellant	21,586
LOI mass	86,573
LOI/TEI propellant	27,883
TLI	
TLI propellant	104,315
TLI mass	201,501
LTV	
LTV total propellant	162,374
LTV inert mass	21,197
IMLEO	**201,501**
Cost	
Dev cost	3,098
Dev cost per mission	310
Unit cost per mission	148
ETO cost per mission	504
Total cost per mission	**961**

Mass Ratios

TLI	2.07
LOI	1.33
Landing	1.95
Ascent	1.89
TEI	1.43
Post-aero	1.10

Specify Engine Fuel Type

	ISP		
TLI	450	1	1 = LH_2/LO_2
LOI/TEI	320	3	2 = CH_4/LO_2
LEV	320	3	3 = Storable

Specify Eng. Technology Level

TLI	0	0 = Current
LOI/TEI	0	1 = Intermediate
LEV	0	2 = Advanced

Mission Duration (days)

TLI	1
LOI/TEI	45
LEV Descent	4
LEV Ascent	45

Insulation Thickness (m)

	TLI	LOI/TEI	LEV
O_2 tank	0.001524	0.05	0.05
H_2 tank	0.001524	0.05	0.05

Insulation Area Density (kg/m²)

	TLI	LOI/TEI	LEV
O_2 tank	0.0493776	1.62	1.62
H_2 tank	0.0493776	1.62	1.62

Figure 8-4a. LTS Spreadsheet Example

	Probability Generator 1 = ON, 0 = OFF	Sigma Value (Actual)	Max. Sigma 1, 2, or 3	Value Returned
1st Stage Isp	0	3	3	450
2nd Stage Isp	0	3	3	450
3rd Stage Isp	0	3	3	320
PRI & SEC STR.	0	35	3	
ETO TRANS COST	0	250	3	2500
Settling dV	0	56.67	3	170
Insulation Thickness	0	0.00017	3	0.05

Direct 3-Stage

OUTPUT

Mass Ratio for Return	2.48
Propellant for Return	20176
Total Return Dry Mass	13367
Ascent Mass	33543
Landing Stage Inert Mass	10767
Landing Legs Mass	1842
Landed Mass	46049
TLI topoff & Landing Dv	3120
Mass Ratio for Landing	2.03
Landing Propellant Mass	48356
Total Lander Propellant I	68532
Total Boost Payload	94404
Boost Mass Ratio	2.06
Boost Inert Mass	12257
Boost Burnout Mass	106661
Boost Propellant	116162
Total Initial Mass	**222823**
Dev Cost	3481
Dev Cost per Mission	348
Unit Cost per Mission	166
ETO Cost per Mission	557
Total Cost per Mission	**1071**

Direct 2-Stage

OUTPUT

Mass Ratio for Return	2.48
Propellant for Return	28503
Total Return Dry Mass	19261
Ascent Mass	47765
Landing Stage Inert Mass	10998
Landing Legs Mass	2199
Landed Mass	54963
TLI topoff & Landing Dv	2950
Mass Ratio for Landing	2.56
Landing Propellant Mass	85749
Total Lander Propellant I	116537
Total Boost Payload	142997
Boost Mass Ratio	2.06
Boost Inert Mass	18283
Boost Burnout Mass	161280
Boost Propellant	174014
Total Initial Mass	**335294**
Dev Cost	4407
Dev Cost per Mission	441
Unit Cost per Mission	210
ETO Cost per Mission	838
Total Cost per Mission	**1489**

Specify Engine Fuel Type

	ISP	
Boost	450	1
Lander	450	1
Return	320	3

1 = LH_2/LO_2
2 = CH_4/LO_2
3 = Storable

Specify Engine Fuel Type

	ISP	
Boost	450	1
Lander	320	3

1 = LH_2/LO_2
2 = CH_4/LO_2
3 = Storable

Specify Eng. Technology Level

Boost	0
Lander	0
Return	0

0 = Current
1 = Intermediate
2 = Advanced

Specify Eng. Technology Level

Boost	0
Lander	0

0 = Current
1 = Intermediate
2 = Advanced

Mission Duration (days)

Boost	1
Lander	4
Return	45

Mission Duration (days)

Boost	1
Lander	45

Insulation Thickness (m)

	Boost	Lander	Return
O_2 tank	0.001524	0.05	0.05
H_2 tank	0.001524	0.05	0.05

Insulation Thickness (m)

	Boost	Lander
O_2 tank	0.05	0.05
H_2 tank	0.05	0.05

Insulation Area Density (kg/m²)

	Boost	Lander	Return
O_2 tank	0.0493776	1.62	1.62
H_2 tank	0.0493776	1.62	1.62

Insulation Area Density (kg/m²)

	Boost	Lander
O_2 tank	1.62	1.62
H_2 tank	1.62	1.62

Figure 8-4b. LTS Spreadsheet Example

background information. The output information provided in the figure is calculated based upon the input information, mission duration, and the technology level. The relationships between the output and input are provided in appendix B. For purposes of discussion, we will illustrate the calculations for change in velocity due to propulsive settling required for fluid venting (settling delta velocity or settling dV) as it applies to the lander and consider the major output values of the initial mass into low-earth orbit and the total cost of mission. These items are flagged on figure 8-4. With a zero technology level of Isp, with a 45-day stay time, the IMLEO value is approximately 223 metric tons, and the total cost per mission is close to \$1071M. This value is also obtained with the primary and secondary structure weight at a level zero, as well as the ETO transportation cost. Using a random number generator with a 3-sigma deviation, Gaussian distributions for two areas, settling dV and ETO transportation costs, are generated, and the technology assessment tool can calculate a mission cost as a function of IMLEO. This is illustrated in figure 8-5. The range in the mission cost varied from about \$910M to \$1270M with IMLEO varying from about 212 to 234 metric tons. The average of all the data is approximately \$1090M per mission at an IMLEO of 223 metric tons. An illustration for different mission modes is shown in appendix C.

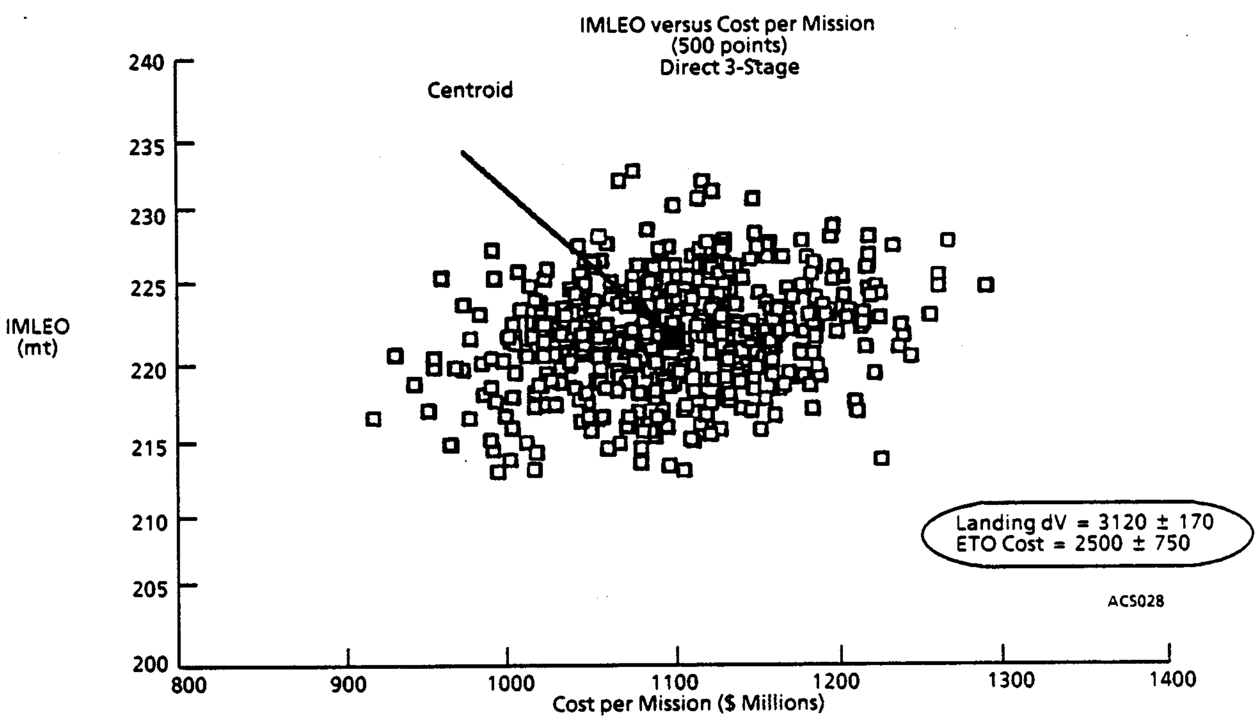
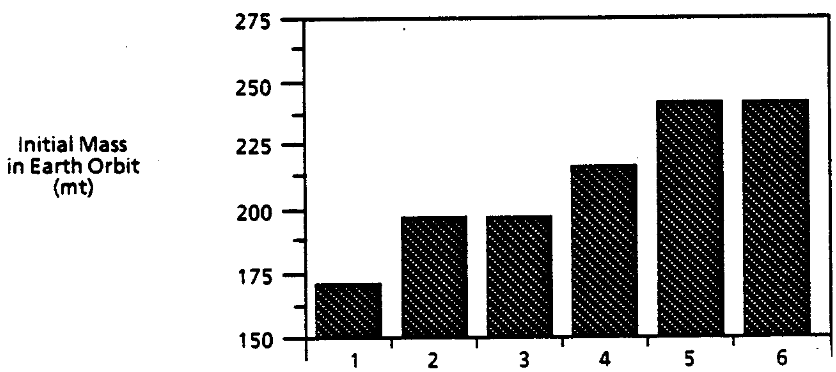


Figure 8-5. Lunar Landing Delta-V Sensitivity (as Related to Fluid Settling)

To perform the analysis (fig. 8-6), a range of characteristics was required for the lunar transportation system. These are illustrated in figure 8-3. The settling delta velocity for the lander was varied from 0 with zero-g fluid venting, to 340 seconds, a value that represents a constant acceleration of 1×10^{-4} g for the lander mission duration (4 days). The cryogenic fluid management also has features relating to the insulation characteristics, the thickness of the total number of layers of insulation that are utilized in the packaging and storage of the cryogenic fluids. For this study, the thickness of the MLI was varied from 2 in or 0.05 meters (m) to 3 layers or 0.001524 m. The mission duration is important as it relates to cryogenic fluid management and the storability of the fuel. The mission duration for the vehicle was broken up as it applies to each mission stage. The TLI or boost duration is set at 1 day, the Lander duration at 4 days, and the LOI/TEI, LEV, and Return stage at 45 days as the nominal case. The last element that was examined was the fuel type of the Lander, LOI/TEI, LEV, and Return stages. The fuel for these stages was varied and traded between storable propellant and cryogenic propellant. The technology level for these engine technologies remained current.



	Booster		Lander		Return
	Insul.	Settling	Insul.	Settling	
1	2"		2"		Cryo
2	2"		3 layers	X	Cryo
3	3 layers	X	3 layers	X	Cryo
4	2"		2"		Stor
5	2"		3 layers	X	Stor
6	3 layers	X	3 layers	X	Stor

Figure 8-6. Cryogenic Trade Study

With the ETO vehicle, several items are considered. The SSME core engines have a range in a specific impulse of 417.1 to 442.9 secs, representing a variance of 3% from a nominal value of 430. The F-1A booster engines have a range in specific impulse of 295.8 to 314.2 secs, representing a variance of 3% from a nominal value of 305. The thrust-to-weight ratio for SSME core engines has a range of 68.4 to 75.6, representing a variance of 5% from a nominal value of 72. The thrust-to-weight ratio for F-1A booster engines has a range of 76 to 84, representing a variance of 5% from a nominal value of 80. The cost per engine for both the SSME and F-1A has a range of \$16 to 24M, representing a variance of 20% from a nominal value of \$20M per engine. The shroud weight of the ETO vehicle was also allowed to vary from 26980 to 29820 lbs, representing a variance of 5% from a nominal value of 28400 lbs.

8.2 HIGH TECHNOLOGY AND LOW TECHNOLOGY CRYOGENIC APPLICATIONS

Cryogenic technology for this study was focused on two independent areas. The first area examined was insulation of cryogenic tanks for storage duration. For this study, we chose MLI as our insulation medium and varied the thickness as a technology variation. Currently, the state of the art in MLI application is three layers of insulation which when applied is a mere 0.001524 m thick. For advanced or high technology MLI application, we made the assumption that technology would be developed to allow a usable MLI thickness application of 2 in or 0.05 m. Using the Lockheed equation, we where able to calculate the boiloff of the fuel and oxidizer tanks, and knowing the density of MLI, the added mass of the insulation on the vehicle was calculated.

The second area studied was fluid venting. In a zero-G environment, venting of boiloff vapor on a cryogenic tank can be very difficult. The desire is to vent off vapor while retaining the mass that is still liquid. In the lack of gravitational force, it is somewhat difficult to separate the vapor from the liquid so that it can be removed. The current state-of-the-art method is to introduce an artificial G-force by means of a small, steady propulsive force to the vehicle. While under influence of this G-force, the liquid, being denser than the vapor, will settle at one end of the tank allowing the vapor to be vented on the other. The expense of this settling force can be calculated as a change in velocity or delta Velocity; therefore, this expense of the settling is referred to as settling delta velocity (or settling dV). For our current technology case, we assumed a required settling force of $1x10^{-4}g$ for the mission duration. For advanced or high technology, we assumed technology has been developed that allowed for the separation of the vapor from the liquid without producing a settling dV and paying the associated performance penalty for the added dV.

The results which were calculated for the 3-stage direct mission are shown in figure 8-6. The payoff for using advanced cryogenic technology was considerable, whether one chose to use a cryogenic or storable stage on the return portion of the mission. The calculated mass for an advanced technology all cryogenic was approximately 175 metric tons in Low Earth Orbit (LEO). The same vehicle, using low technology and performing the same mission, had a LEO mass of approximately 195 metric tons. The high-technology vehicle with storable return stage had a calculated LEO mass of approximately 220 metric tons. While a similar vehicle, using low technology and performing the same mission, had a LEO mass of approximately 247 metric tons. Also, one can see for the booster stage that high technology vs. low only has a limited effect. This is due to the fact that the booster stage was modeled with a mission of only 1 day, and thus produced little boiloff and little settling dV in the low technology. It is apparent that advanced technology has a higher payoff as mission duration increases.

8.3 SETTLING DELTA VELOCITY SENSITIVITY

During this portion of the task, a study was done to examine the sensitivity of the mission cost and performance to a deviation in required velocity change needed for fluid settling. The settling dV was varied according to a probability based on a Gaussian distribution. A 3-sigma distribution was selected and landing plus settling dV was allowed to vary between 2950 and 3290 m/sec. This distribution produced a variation in LEO mass from 212 to 234 metric tons for the 3-stage direct mission (fig. 8-5). Similar plots for 3-stage LOR and 2-stage direct can be found in the appendix. This plot is useful in showing the performance and cost risk associated with settling dV.

8.4 ETO SENSITIVITIES

Several probability plots were generated to examine the sensitivity of the ETO vehicle to certain variables. The variables selected for examination were Isp of the booster and core engines, engine and shroud mass, and engine cost.

For the first trade sensitivity, the F-1A engines in the booster and the SSME engines in the core were given an Isp variation of ± 3%. The booster engines were hence given an Isp range of 296 to 314. Similarly, the core engines were given an Isp range of 417 to 443. The results of this distribution are shown in figure 8-7. The sensitivity of the ETO model to Isp variation is strictly in performance output. The payload varied from 275000 to 325000 lbs with a constant mission cost of approximately $610M. Thus, a ±3% change in engine Isp results in approximately a 9% variation in performance. As one would expect, the sensitivity of the ETO model to performance is quite high.

For the second study, the inert masses of the ETO model were varied in a test of sensitivity. To perform this test, the thrust-to-weight ratio (T/W) of engines and the shroud weight were allowed to vary ±5%. The booster engines varied in T/W from 76 to 84. The core engines varied in T/W from 68.4 to 75.6. The shroud weight was allowed to vary from 26980 to 29820 lbs. For this sensitivity, the payload varied from 295700 to

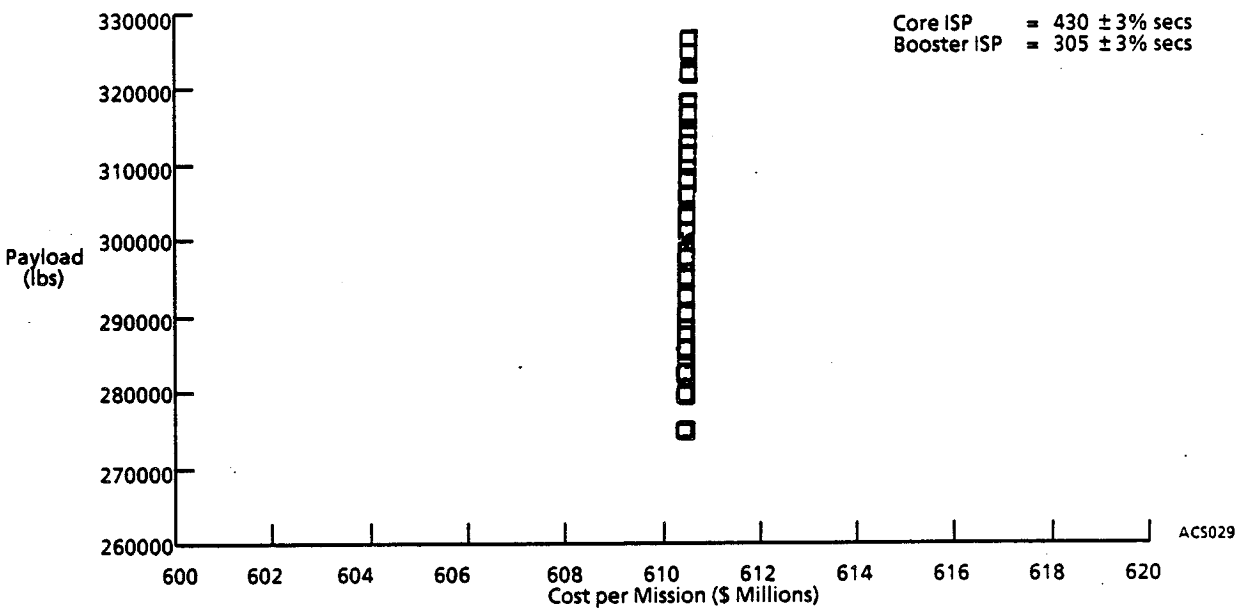


Figure 8-7. Payload versus Mission Cost, ETO Booster

301800 lbs, and the accompanying mission cost varied from $607.3 to $613.5M (fig. 8-8). A 5% variation in these inert weights results in a 1% variation in both cost and performance.

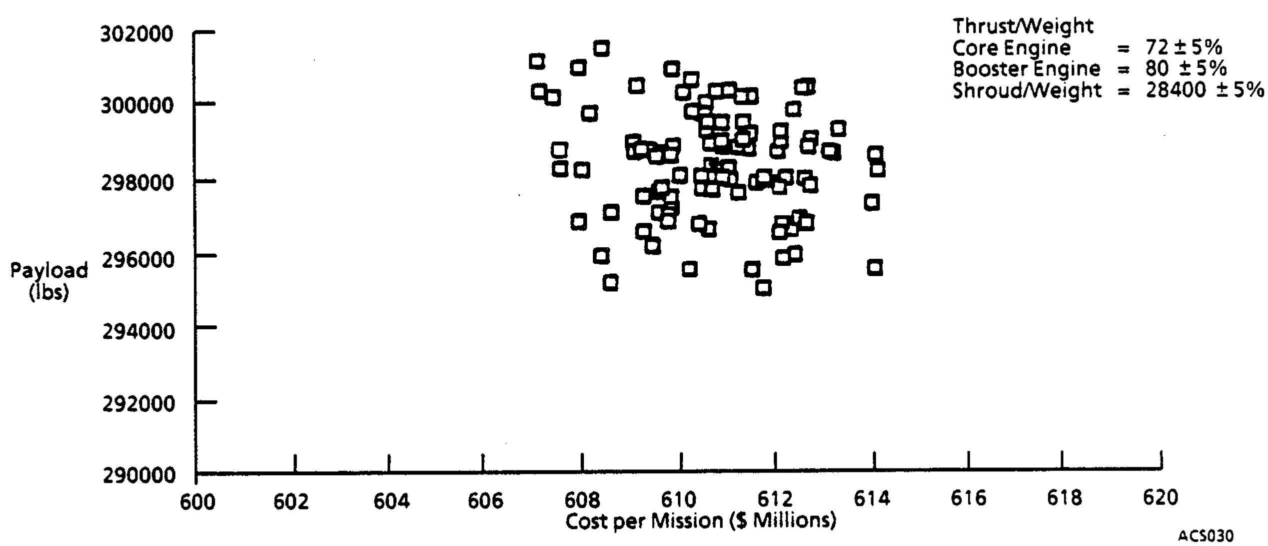


Figure 8-8. Payload versus Mission Cost, ETO Booster

For the third trade, the engine cost was allowed to vary 20%. A large variation was allowed for engine cost due to the lack of information on the cost of engines for this application. Both the core and booster engines in the model were given a cost variation from $16 to $24M. The calculation indicates that the mission cost varies from $590 to $641M (fig. 8-9). Thus a 20% variation in engine cost resulted in a 10% variation in mission cost.

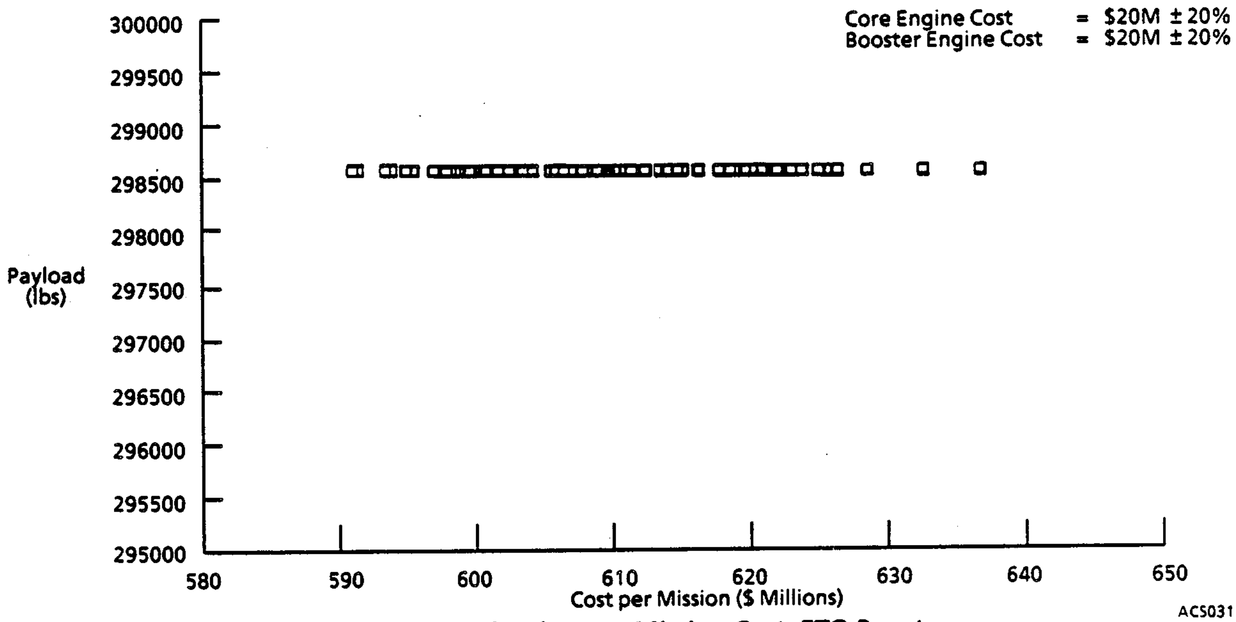


Figure 8-9. Payload versus Mission Cost, ETO Booster

Finally, all the aforementioned parameters were allowed to vary as previously described. This resulted in a variation in payload from 272000 to 321000 lbs (±9%), and a variation in mission cost from $579 to $640M (±5.5%) (fig. 8-10). These results indicate that the variation of unrelated variables in the model do have a combined effect on the output of the model. Even though there are significant variations in each of the parameters, amounting to ±56% total, the combined effect is not additive due to the interrelationship of the parameters.

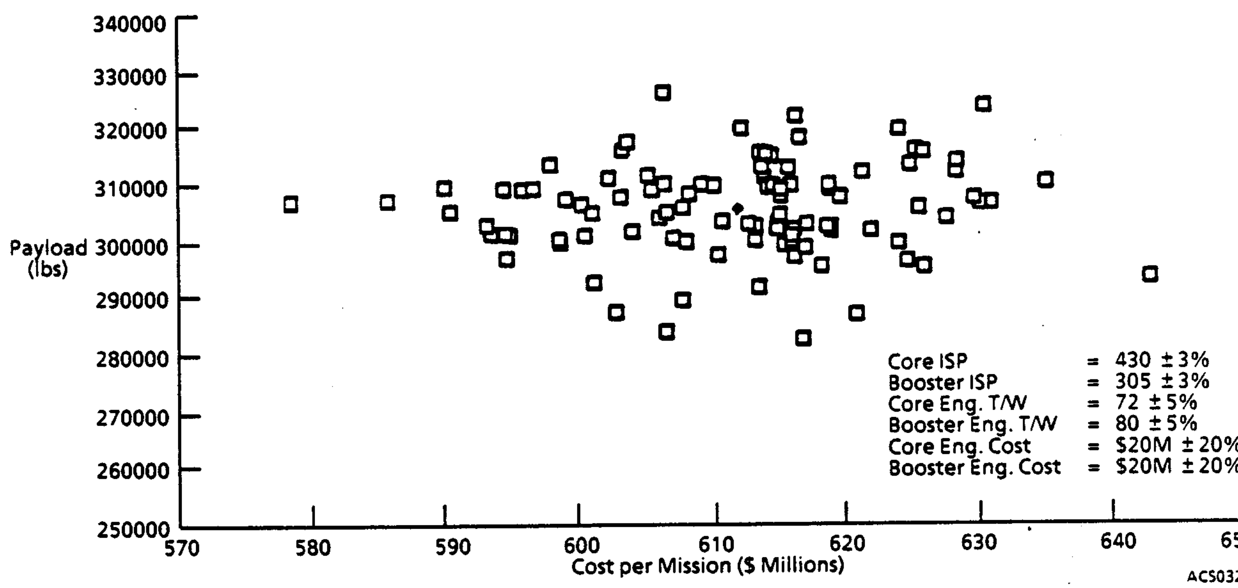


Figure 8-10. Payload versus Mission Cost, ETO Booster

9. CONCLUDING REMARKS

A tool has been developed that assists in the assessment of technologies based upon performance, payoff, cost, risk, and schedule. A decision theory approach has been incorporated in the methodology which provides a quantitative assessment of the technologies. In the project, an assessment was made of two technologies, cryogenic fluid management and propulsion, for a single alternative of the integrated space transportation system composed of an HLLV core, lunar transportation system, and chemical/LH_2 return. The technology assessment was made for three levels of technology, current, intermediate and advanced, using inputs provided by experts in the field, the Technologist and the Program Manager. Rapid response to changes in the input characteristics of the technology requirements has been illustrated of this tool with a spreadsheet format using a desktop computer. Enhancements to the tool may be provided for a broader range of technologies and applications, improved quantification of the technology-level capabilities, and updated relationships for technology performance which would lead to a more rigorous Technology Assessment Tool.

REFERENCES

1. "Space Transfer Concepts and Analysis for Exploration Missions," Final Report Phase 1, Boeing Defense and Space Group, Huntsville D615-10030-2, Mar, 1991.

2. "America at the Threshold," The Synthesis Group on America's Space Exploration Initiative, May, 1991.

3. "Technology Assessment Tool Development," Phase 1 Final Report, Boeing Defense & Space Group, Huntsville, D615-14111, February, 1992.

Appendix A

Assessment Tool Spreadsheets

App. A

Multi-Mode

	Current	Intermediate	Advanced
[redacted]	hrottle RL-1	prated RL-1	New engine
Tanks	Aluminum	Al-Li	Al-composite
Avionics	IUS/Centaur	SSF-class	Advanced
Elect. Powe	o.a. fuel cel	N/A	dv. fuel cells
RCS	Bi-props	N/A	ntegrated cryo
Pri & Sec S	Aluminum	Al-Li	Composites

Specify Technology Level

Tanks	0	0=Current
Avionics	0	1=Intermediate
Elect. Powe	0	2=Advanced
RCS	*0*	
Pri & Sec S	*0*	

INPUT	
1. Vehicle Structures	
Legs fract.	0.04
Density Alum. 2219(kg/r	2850
Density Al-Li 2195(kg/m	2713
Density Comp. (kg/m^3)	1852.5
Stress Alum. 2219(PSI)	38,000
Stress Al-Li 2195 (PSI)	50,000
Stress Composite (PSI)	114,000
2. Cryo Fluid Mgmt.	
Ullage Factor	5%
Residuals Factor	2%
Mixture Ratio LO2/LH2	6
Mixture Ratio LO2/CH4	3
Mixture Ratio Storable	1.6
Density LO2 (kg/m^3)	1141
Density LH2 (kg/m^3)	71
Density CH4 (kg/m^3)	423
Storable Fuel (kg/m^3)	800
Storable Oxidizer (kg/m'	1500
Tank Pressure (PSIg)	35
3. Cryo Eng./Prop.	
Zero-base Isp LO2/LH2	450
Intermediate Isp LO2/LF	465
Advanced Isp LO2/LH2	478
Zero-base Isp LO2/CH4	350
Intermediate Isp LO2/CF	365
Advanced Isp LO2/CH4	380
Zero-base Isp Storable	320
Intermediate Isp Storable	340
Advanced Isp Storable	380
4. Veh. Avionics/Software	
kiloWatts	2
5. Aerobrake	
Brake fract.	0.2

LOR 3-Stage

OUTPUT	
Mass Statement	
TEI	
TEI dry mass	13,408
TEI propellant	5,751
TEI mass	19,159
LEV	
LEV Orbit burnout mass	8,949
LEV ascent propellant	8,137
Ascent mass	17,086
Landing mass	23,243
Landing legs mass	930
LEV descent propellant	22,585
LEV gross	45,828
LEV total propellant	30,722
LEV inert mass	6,106
LOI	
MLOI	64,988
LOI propellant	21,586
LOI mass	86,573
LOI/TEI propellant	27,883
TLI	
TLI propellant	104,315
TLI mass	201,501
LTV	
LTV total propellant	162,374
LTV inert mass	21,197
IMLEO	201,501
Cost	
Dev cost	3,098
Dev cost per mission	310
Unit cost per mission	148
ETO cost per mission	906
Total cost per mission	1,363

6. Crew Modules & Sys.	
Transfer cab mass	8,263
LTV cab mass	8000
LEV cab mass	4000
7.ECLS	
8. Vehicle Assembly	
9. Orbit Launch & Checkout	
10. Vehicle Flight Ops.	
Basic Mission Requirements	
Earth G	9.80665
PI	3.1415927
Payload Del'd (kg)	5000
Payload Ret'd (kg)	0
% growth	12
LOR 3-Stage dV's	
TLI DV	3204
LOI DV	900
Landing DV	2100
Ascent DV	2000
TEI DV	1120
Post-Aero DV	300
Dir. Exp. dV's	
Booster DV	3084
Finite DV	100
ReturnDV	2850
TLI Delta V	3084
Landing Delta V	2950
Cost	
ETO Transportation Cost	4495
LTV Dev Cost, $/kg	140000
LTV Unit Cost, $/kg	20000
# missions amortize dev	10
Effective # of vehicle re	3

2250

Mass Ratios

TLI	2.07
LOI	1.33
Landing	1.95
Ascent	1.89
TEI	1.43
Post-aero	1.10

Specify Engine Fuel Type

	ISP		
TLI	450	[illegible]	1= LH2/LO2
LOI/TEI	320	[illegible]	2= CH4/LO2
LEV	320	[illegible]	3= Storable

Specify Engine Technology Level

TLI	0	0=Current
LOI/TEI	0	1=Intermediate
LEV	0	2=Advanced

Mission Duration (days)

TLI	3
LOI/TEI	45
LEV Descent	4
LEV Ascent	45

Insulation Thickness (m)

	TLI	LOI/TEI	LEV
O2 tank	0.001524	0.001524	0.001524
H2 tank	0.001524	0.001524	0.001524

Insulation Area Density (kg/m^2)

	TLI	LOI/TEI	LEV
O2 tank	0.0493776	0.0493776	0.0493776
H2 tank	0.0493776	0.0493776	0.0493776

Multi-Mode

	Probability Generator 1=ON, 0=OF	Sigma Value (Actual)	Max. Sigma 1,2, or 3	Value Returned
1st Stage Isp	0	3	3	450
2nd Stage Isp	0	3	3	450
3rd Stage Isp	0	3	3	320
PRI & SEC STR.	0	35	3	
ETO TRANS COST	0	250	3	4495
Settling dv	0	11	3	340
Insulation Thickness	0	0.00017	3	0.001524

Direct 3-Stage

OUTPUT	
Mass Ratio For Return	2.48
Propellant for Return	20233
Total Return Dry Mass	13405
Ascent Mass	33638
Landing Stage Inert Mas:	11089
Landing Legs Mass	1858
Landed Mass	46444
TLI topoff + Landing Dv	3290
Mass Ratio for Landing	2.11
Landing Propellant Mass	55166
Total Lander Propellant I	75400
Total Boost Payload	101610
Boost Mass Ratio	2.06
Boost Inert Mass	13053
Boost Burnout Mass	114663
Boost Propellant	124848
Total Initial Mass	**239512**
Dev Cost	3640
Dev Cost per Mission	364
Unit Cost per Mission	173
ETO Cost per Mission	1077
Total Cost per Mission	**1614**

Specify Engine Fuel Type

	ISP		
			1= LH2/LO2
Boost	450	[redacted]	2= CH4/LO2
Lander	450	[redacted]	3= Storable
Return	320	[redacted]	

Specify Engine Technology Level

Boost	0	0=Current
Lander	0	1=Intermediate
Return	0	2=Advanced

Direct 2-Stage

OUTPUT	
Mass Ratio For Return	2.48
Propellant for Return	28503
Total Return Dry Mass	19261
Ascent Mass	47765
Landing Stage Inert Mas:	10998
Landing Legs Mass	2199
Landed Mass	54963
TLI topoff + Landing Dv	2950
Mass Ratio for Landing	2.56
Landing Propellant Mass	85749
Total Lander Propellant I	116537
Total Boost Payload	142997
Boost Mass Ratio	2.06
Booster Inert Mass	18283
Booster Burnout Mass	161280
Booster Propellant	174014
Total Initial Mass	**335294**
Dev Cost	4407
Dev Cost per Mission	441
Unit Cost per Mission	210
ETO Cost per Mission	1507
Total Cost per Mission	**2158**

Specify Engine Fuel Type

	ISP		
			1= LH2/LO:
Boost	450	[redacted]	2= CH4/LO:
Lander	320	[redacted]	3= Storable

Specify Engine Technology Level 0=Current

Boost	0	1=Intermedi
Lander	0	2=Advanced

Mission Duration (days)

10/8/92 10:3(

Mission Duration (days)

Boost	[illegible]
Lander	[illegible]
Return	[illegible]

Insulation Thickness (m)

	Boost	Lander	Return
O2 tank	[illegible]	[illegible]	[illegible]
H2 tank	[illegible]	[illegible]	[illegible]

Insulation Area Density (kg/m^2)

	Boost	Lander	Return
O2 tank	0.0493776	0.0493776	0.04938
H2 tank	0.0493776	0.0493776	0.04938

Multi-Mode

Boost	[illegible]
Lander	[illegible]

Insulation Thickness (m)

	Boost	Lander
O2 tank	[illegible]	[illegible]
H2 tank	[illegible]	[illegible]

Insulation Area Density (kg/m^2)

	Boost	Lander
O2 tank	1.62	0.0493776
H2 tank	1.62	0.0493776

LOR 3-Stage

INERT WEIGHT CALCULATION				
	TLI	LOI/TEI	LEV	Dev. Cost
Pri & Sec S	2,628	1,252	1,303	0
Tanks	3,218	276	341	0
Avionics	0	600	600	0
Elect. Powe	0	686	686	0
RCS	0	678	270	0
Main Prop.	3,629	1,336	1,422	0
Ident. inert mass	9,475	4,829	4,622	0

Direct 2-Stage

INERT WEIGHT CALCULATION			
	Lander	Booster	Dev.Cost
Pri & Sec S	2848	3882	0
Tanks	1154	6007	0
Avionics	600		0
Elect. Powe	686		0
RCS	536	714	0
Main Prop.	3996	5720	0
Ident. inert mass	9820	16324	0

Direct 3-Stage

	Ascent/Retur	Lander	TLI	Dev.Cost
Pri & Sec S	1114	1743	2997	0
Tanks	209	1703	3850	0
Avionics	600			0
Elect. Powe	686			0
RCS	238	346	562	0
Main Prop.	1107	2155	4245	0
Ident. inert mass	3954	5947	11655	0

Note: Inert masses calculated in these sections are based on scaling equations relating inert mass to a percentage of fuel mass and do not currently follow a design calculation format, therefore these calculations are not discussed in Appendix B.

Multi-Mode

Material Stress Factor	38000
Material Density	2850

LOR 3-Stage

	TLI	LOI/TEI	LEV
Oxidizer Density	1141	1500	1500
Fuel Density	71	800	800
Mixture Ratio	6	1.6	1.6
Oxidizer Insul. Dens.	0.0493776	0	0
Fuel Insul. Dens.	0.0493776	0	0
Oxygen "Q"	4.37604699	4.37604699	4.37604699
Fuel "Q"	4.98760729	4.98760729	4.98760729

Fuel Volumes		**Tank Thickness**	
LOI/TEI Oxidizer Vol.	12.01	LOI/TEI Oxidizer Thickn	1.35E-03
LOI/TEI Fuel Vol.	14.08	LOI/TEI Fuel Thickness	1.42E-03
LEV Ascent Oxidizer Vc	3.51	LEV Ascent Oxidizer Thi	1.30E-03
LEV Ascent Fuel Vol.	4.11	LEV Ascent Fuel Thickne	1.30E-03
LEV Descent Oxidizer V	9.73	LEV Descent Oxidizer Th	1.30E-03
LEV Descent Fuel Vol.	11.40	LEV Descent Fuel Thickn	1.33E-03
TLI Oxidizer Vol.	82.28	TLI Oxidizer Thickness	2.56E-03
TLI Fuel Vol.	220.38	TLI Fuel Thickness	3.56E-03
Tank Diameters		**Tank Mass**	
LOI/TEI Oxidizer DIA.	2.09	LOI/TEI Oxidizer	127.13
LOI/TEI Fuel DIA.	2.21	LOI/TEI Fuel	148.98
LEV Ascent Oxidizer DI	1.39	LEV Ascent Oxidizer	53.87
LEV Ascent Fuel DIA.	1.46	LEV Ascent Fuel	59.88
LEV Descent Oxidizer D	1.95	LEV Descent Oxidizer	106.39
LEV Descent Fuel DIA.	2.06	LEV Descent Fuel	120.67
TLI Oxidizer DIA.	3.98	TLI Oxidizer	870.87
TLI Fuel DIA.	5.52	TLI Fuel	2332.55
Tank Cylinder Length		**Tank Insulation**	
LOI/TEI Oxidizer length	2.09	LOI/TEI Oxidizer	0.00
LOI/TEI Fuel length	2.21	LOI/TEI Fuel	0.00
LEV Ascent Oxidizer ler	1.39	LEV Ascent Oxidizer	0.00
LEV Ascent Fuel length	1.46	LEV Ascent Fuel	0.00
LEV Descent Oxidizer le	1.95	LEV Descent Oxidizer	0.00
LEV Descent Fuel lengtl	2.06	LEV Descent Fuel	0.00
TLI Oxidizer length	3.98	TLI Oxidizer	4.90
TLI Fuel length	5.52	TLI Fuel	9.46
Surface Area		**Boiloff**	
LOI/TEI Oxidizer	27.54	LOI/TEI Oxidizer	6206.75
LOI/TEI Fuel	30.61	LOI/TEI Fuel	3696.36
LEV Ascent Oxidizer	12.12	LEV Ascent Oxidizer	2730.74
LEV Ascent Fuel	13.47	LEV Ascent Fuel	1626.26
LEV Descent Oxidizer	23.93	LEV Descent Oxidizer	479.41
LEV Descent Fuel	26.60	LEV Descent Fuel	285.50
TLI Oxidizer	99.33	TLI Oxidizer	497.51
TLI Fuel	191.58	TLI Fuel	514.09

Direct Expendable 2-Stage	Booster	Lander		
Oxidizer Density	1141	1500		
Fuel Density	71	800		
Mixture Ratio	6	1.6		
Insulation Density	1.62	0		
	1.62	0		
	0.13338191	4.37604699		
	0.15202227	4.98760729		
Fuel Volumes			**Tank Thickness**	
Lander Oxidizer Vol.	50.20		Lander Oxidizer Thicknes	2.17E-03
Lander Fuel Vol.	58.83		Lander Fuel Thickness	2.29E-03
Booster Oxidizer Vol.	137.26		Booster Oxidizer Thickne	3.04E-03
Booster Fuel Vol.	367.63		Booster Fuel Thickness	4.22E-03
Tank Diameters			**Tank Mass**	
Lander Oxidizer Dia	3.37		Lander Oxidizer	531.32
Lander Fuel Dia	3.56		Lander Fuel	622.65
Booster Oxidizer Dia	4.72		Booster Oxidizer	1452.75
Booster Fuel Dia	6.55		Booster Fuel	3891.05
Tank Cylinder Length			**Tank Insulation**	
Lander Oxidizer length	3.37		Lander Oxidizer	0.00
Lander Fuel length	3.56		Lander Fuel	0.00
Booster Oxidizer length	4.72		Booster Oxidizer	226.34
Booster Fuel length	6.55		Booster Fuel	436.53
Surface Area			**Boiloff**	
Lander Oxidizer	71.45		Lander Oxidizer	16104.48
Lander Fuel	79.42		Lander Fuel	9590.83
Booster Oxidizer	139.72		Booster Oxidizer	21.33
Booster Fuel	269.46		Booster Fuel	22.04

Direct Expendable 3-Stage	Booster	Lander	Return		
Oxidizer Density	1141	1141	1500 kg/m^3		
Fuel Density	71	71	800 kg/m^3		
Mixture Ratio	6	6	1.6		
Oxidizer Insulation Dens	0.0493776	0.0493776	0 kg/m^2		
Fuel Insulation Dens.	0.0493776	0.0493776	0 kg/m^2		
Oxygen "Q"	4.37604699	4.37604699	4.37604699		
Fuel "Q"	4.98760729	4.98760729	4.98760729		
Fuel Volumes				**Tank Thickness**	
Ascent/Return Ox.	8.72			Ascent/Return Ox.	1.30E-03
Ascent/Return Fuel	10.21			Ascent/Return Fuel	1.30E-03
Lander Ox.	43.51			Lander Ox.	2.07E-03
Lander Fuel	116.55			Lander Fuel	2.88E-03
TLI Ox.	98.48			TLI Ox.	2.72E-03
TLI Fuel	263.76			TLI Fuel	3.78E-03
Tank Diameters				**Tank Mass**	
Ascent/Return Ox.	1.88			Ascent/Return Ox.	98.87
Ascent/Return Fuel	1.98			Ascent/Return Fuel	109.90
Lander Ox.	3.22			Lander Ox.	460.55
Lander Fuel	4.47			Lander Fuel	1233.55

TLI Ox.	4.22	TLI Ox.	1042.29
TLI Fuel	5.86	TLI Fuel	2791.68
Tank Cylinder Length		**Tank Insulation**	
Ascent/Return Ox.	1.88	Ascent/Return Ox.	0.00
Ascent/Return Fuel	1.98	Ascent/Return Fuel	0.00
Lander Ox.	3.22	Lander Ox.	3.21
Lander Fuel	4.47	Lander Fuel	6.19
TLI Ox.	4.22	TLI Ox.	5.53
TLI Fuel	5.86	TLI Fuel	10.66
Surface Area		**Boiloff**	
Ascent/Return Ox.	22.24	Ascent/Return Ox.	5012.06
Ascent/Return Fuel	24.72	Ascent/Return Fuel	2984.87
Lander Ox.	64.96	Lander Ox.	1301.39
Lander Fuel	125.28	Lander Fuel	1344.77
TLI Ox.	111.97	TLI Ox.	560.82
TLI Fuel	215.96	TLI Fuel	579.51

PROBABILITY CURVE GENERATION

SERIAL #

32415.423

2nd ISP		3rd ISP		settling dv		Insulation Thk	
9770.7713	9770	15941.905	15941	5805.3219	5805	2187.1415	2187
8366.6207	8366	13429.457	13429	2433.3936	2433	1838.6668	1838
3703.4272	3703	5499.765	5499	499.24048	499	201.64345	201
5875	5875	1701	1701	24047	24047	10749	10749
16074	16074	8964	8964	24914	24914	13396	13396
24270	24270	26970	26970	24306	24306	3908	3908
	0.524848		0.2413932		0.4180672		0.926005

1st Isp		PRI & SEC STR.		ETO COST	
8546.6077	8546	19554.029	19554	17997.922	17997
827.82985	827	9378.5797	9378	10758.772	10758
823.5207	823	838.88919	838	7249.7637	7249
8454	8454	14144	14144	20318	20318
21148	21148	8494	8494	3662	3662
18862	18862	21412	21412	21850	21850
	0.5991243		0.4536727		0.5126531

	RANDOM #		SIGMA	RANDOM # LIMITATIONS Max	Min
ISP	0.5991243	0.7157417	0.2732764	0.9987	0.0013
2nd Isp	0.524848	0.8028988	0.0747463	0.9987	0.0013
3rd Isp	0.2413932	1.1921946	-0.720234	0.9987	0.0013
PRI&SEC STR.	0.4536727	0.8890328	-0.116748	0.9987	0.0013
ETO COST	0.5126531	0.817408	0.0420817	0.9987	0.0013
Settling DV	0.4180672	0.9338699	-0.213616	0.9987	0.0013
Insulation Thk	0.926005	0.2772647	1.4137796	0.9987	0.0013

Binary Table

Tanks	1	0	0
Avionics	1	0	0
Elect. Pwr	1	0	0
RCS	1	0	0
Pri&Sec Str	1	0	0

LOR 3-Stage

Specify Fuel Type

TLI	1	0	0
LOI/TEI	0	0	1
LEV	0	0	1
	Specify Tech Level		
TLI	1	0	0
LOI/TEI	1	0	0
LEV	1	0	0

Direct 3-Stage

	Specify Fuel Type		
Boost	1	0	0
Lander	1	0	0
Return	0	0	1
	Specify Tech Level		
Boost	1	0	0
Lander	1	0	0
Return	1	0	0

Direct 2-Stage

	Specify Fuel Type		
Boost	1	0	0
Lander	0	0	1
	Specify Tech Level		
Boost	1	0	0
Lander	1	0	0

	Random # Generator 1=On, 0=Off	Sigma Value (Actual)	Max. Sigma 1,2 or3	Returned Value
Ascent Dv	0	96	3	9600.00
Booster Isp	0	4	3	414.00
Core Isp	0	4	3	430.00
Boost Eng T/W	0	0.7	3	72.00
Core Eng T/W	0	0.7	3	72.00
Bst. Eng. Cost	0	0.2	3	20.00
Core Eng. Cost	0	0.2	3	20.00
Shroud Wt.	0	280	3	28400.00

INPUTS	
Ascent Dv (m/sec)	9600
Booster Isp (vac.)	414
Booster Isp (s/l)	381
Core Isp (vac.)	430
Core Isp (s/l)	358
Booster Mixture Ratio	6
Core Mixture Ratio	6
Boost Eng. Thrust (vac)	560000
Core Eng. Thrust (vac)	583000
Core Eng. Throttle set%	75%
Number of boost eng.	12
Number of Core eng.	4
Boost Eng T/W	72
Core Eng T/W	72
Bst. Eng. Cost ($M each)	20
Core Eng. Cost($M each)	20
Shroud weight	28400
# of Boost Tanks	2
Boost prop cap per tank	1.69E+06
Core prop capacity	1.69E+06
Stg. 1 LOX Tank factor	0.0087
Stg. 1 LH2 Tank factor	0.136
Stg. 1 Proport. factor	0.00212
Stg. 1 2/3 pwr factor	1.535
Stg. 1 Fixed Mass	2250
Stg. 1 Contigency %	10%
Stg. 1 Residuals %	1%
Stg. 2 LOX Tank factor	0.0098
Stg. 2 LH2 Tank factor	0.103
Stg. 2 Proport. factor	0.00168
Stg. 2 2/3 pwr factor	1.588
Stg. 2 Fixed Mass	3000
Stg. 2 Contigency %	10%
Stg. 2 Residuals %	0.90%
Tank Cost factor	0.011075
Tank Materials factor	1
Tank Cost Exponent	0.732
Structures Cost factor	0.004546
Struct. Materials factor	1
Struct. Cost Exponent	0.837
Avionics Cost factor	0.021924
Avionics Tech. factor	1
Avionics Cost Exponent	0.927
G	9.80665

OUTPUT	
Boost Thrust	6720000.00
Core Thrust	2332000.00
Liftoff Thrust	8125873.41
Payload	279773.31
Thrust Ratio	0.26
Isp Ratio	0.96
Mean Isp	417.21
Boost Thrust fract.	0.79
q factor	0.80
Boost Eng. Instl. Wt.	186666.67
Core Eng. Instl. Wt.	64777.78
Booster Prop. Wt.	1690000.00
Boost Prop. Wt.	4226972.13
Booster Inert	199919.91
Booster Structure Cost	599.76
Core Boost Prop	846972.13
Cost	80.00
Core Inert	160641.92
Core Structure Cost	240.96
Core Phase Prop.	843027.87
Total Core Prop.	1690000.00
Core Sep Wt	1283443.09
Liftoff Wt	5938655.05
Boost mr	3.47
Boost dv	5089.72
Core dv	4510.28
Core mr	2.91
Core Burnout Wt.	440415.23
Payload	279773.31
Total Cost	711.93
Cost Per Pound	2544.68

Booster Inert	
LOX Load	1448571.43
Fuel Load	241428.57
LOX Tank	12602.57
Fuel Tank	32834.29
Proport. Mass	3582.80
Scaled Mass	21778.74
Ident Mass	166381.74
Contingency Mass	16638.17
Residual Mass	16900.00
Cost Mass	183019.91
LOX Tank Cost	11.11
Fuel Tank Cost	22.40
Proport Cost	4.29
Scaled Cost	19.43
Engines Cost	120.00
Engine Inst. Cost	36.78
Fixed Mass Cost	28.08
*Cost	122.10
Contingency Cost	10.14
*Cost	252.24

Core Inert	
LOX Load	1448571.43
Fuel Load	241428.57
LOX Tank	14196.00
Fuel Tank	24867.14
Proport. Mass	2839.20
Scaled Mass	22530.71
Ident Mass	132210.84
Contingency Mass	13221.08
Residual Mass	15210.00
Cost Mass	145431.92
LOX Tank Cost	12.13
Fuel Tank Cost	18.28
Proport Cost	3.53
Scaled Cost	19.99
Engines Cost	80.00
Engine Inst. Cost	27.09
Fixed Mass Cost	36.66
*Cost	117.68
Contingency Cost	9.77
*Cost	207.46

PROBABILITY CURVE GENERATION

SERIAL #

33861.5994

Ascent Dv		Booster Isp		Core Isp	
521.702262	521	19998.1474	19998	7500.35375	7500
272.583128	272	6901.19672	6901	4242.03507	4242
154.051496	154	2469.9905	2469	3193.66873	3193
28553	28553	29530	29530	11202	11202
16510	16510	6286	6286	3048	3048
26180	26180	26324	26324	28356	28356
	0.35143801		0.05111632		0.40578418

Boost Eng. T/W		Core Eng. T/W		Boost Eng. Cost	
2472.79758	2472	7452.46805	7452	30559.9755	3055
1002.95905	1002	5046.20568	5046	19241.0261	19241
580.269027	580	980.719705	980	12747.7853	12747
29215	29215	2994	2994	7832	7832
20974	20974	20240	20240	9586	9586
7814	7814	15290	15290	18388	18388
	0.91492242		0.2709833		0.18144752

Core Eng. Cost		Shroud Wt	
31588.7832	3158	18786.7644	18786
580.481439	580	63.0906632	63
396.977164	396	12.1579732	12
25445	25445	3892	3892
8938	8938	10836	10836
6796	6796	2040	2040
	0.3596647		0.55339722

	RANDOM #		SIGMA	RANDOM # LIMITS MAX	MIN
Ascent Dv	0.35143801	1.02260547	-0.3980142	0.9987	0.0013
Booster Isp	0.05111632	1.72442786	-1.6198253	0.9987	0.0013
Core Isp	0.40578418	0.94970198	-0.247263	0.9987	0.0013
Boost Eng. T/W	0.91492242	0.29818787	1.33867811	0.9987	0.0013
Core Eng. T/W	0.2709833	1.14267147	-0.6301714	0.9987	0.0013
Boost Eng. Cost	0.18144752	1.30644128	-0.9177732	0.9987	0.0013
Core Eng. Cost	0.3596647	1.0112285	-0.3749626	0.9987	0.0013
Shroud Wt.	0.55339722	0.76920688	0.15108309	0.9987	0.0013

	Random # Generator 1=On, 0=Off	Sigma Value (Actual)	Max. Sigma 1,2 or3	Returned Value
Ascent Dv	0	96	3	9200.00
Booster Isp	0	3.05	3	305.00
Core Isp	0	4.3	3	430.00
Boost Eng T/W	0	1.33	3	80.00
Core Eng T/W	0	1.2	3	72.00
Bst. Eng. Cost	0	1.33	3	20.00
Core Eng. Cost	0	1.33	3	20.00
Shroud Wt.	0	473.33	3	28400.00

INPUTS	
Ascent Dv (m/sec)	9200
Booster Isp (vac.)	305
Booster Isp (s/l)	265
Core Isp (vac.)	430
Core Isp (s/l)	358
Booster Mixture Ratio	2.3
Core Mixture Ratio	6
Boost Eng. Thrust (vac)	1522000
Core Eng. Thrust (vac)	583000
Core Eng. Throttle set%	75%
Number of boost eng.	6
Number of Core eng.	4
Boost Eng T/W	80
Core Eng T/W	72
Bst. Eng. Cost ($M each)	20
Core Eng. Cost($M each)	20
Shroud weight	28400
# of Boost Tanks	2
Boost prop cap per tank	2.63E+06
Core prop capacity	1.69E+06
Stg. 1 LOX Tank factor	0.013
Stg. 1 LH2 Tank factor	0.014
Stg. 1 Proport. factor	0.00212
Stg. 1 2/3 pwr factor	1.535
Stg. 1 Fixed Mass	2250
Stg. 1 Contigency %	10%
Stg. 1 Residuals %	1.50%
Stg. 2 LOX Tank factor	0.0098
Stg. 2 LH2 Tank factor	0.103
Stg. 2 Proport. factor	0.00168
Stg. 2 2/3 pwr factor	1.588
Stg. 2 Fixed Mass	3000
Stg. 2 Contigency %	10%
Stg. 2 Residuals %	0.90%
Tank Cost factor	0.011075
Tank Materials factor	1
Tank Cost Exponent	0.732
Structures Cost factor	0.004546
Struct. Materials factor	1
Struct. Cost Exponent	0.837
Avionics Cost factor	0.021924
Avionics Tech. factor	1
Avionics Cost Exponent	0.927
G	9.80665

OUTPUT	
Boost Thrust	9132000.00
Core Thrust	2332000.00
Liftoff Thrust	9875886.24
Payload	298723.26
Thrust Ratio	0.19
Isp Ratio	0.71
Mean Isp	319.95
Boost Thrust fract.	0.84
q factor	0.88
Boost Eng. Instl. Wt.	228300.00
Core Eng. Instl. Wt.	64777.78
Booster Prop. Wt.	2630000.00
Boost Prop. Wt.	5974563.84
Booster Inert	244280.35
Booster Structure Cost	732.84
Core Boost Prop	714563.84
Cost	80.00
Core Inert	160641.92
Core Structure Cost	240.96
Core Phase Prop.	975436.16
Total Core Prop.	1690000.00
Core Sep Wt	1434801.34
Liftoff Wt	7926325.87
Liftoff T/W	1.25
Boost mr	4.06
Boost dv	4397.27
Core dv	4802.73
Core mr	3.12
Core Burnout Wt.	459365.18
Payload	298723.26
Total Cost	610.30
Cost Per Pound	2043.02

Booster Inert	
LOX Load	1833030.30
Fuel Load	796969.70
LOX Tank	23829.39
Fuel Tank	11157.58
Proport. Mass	5575.60
Scaled Mass	29246.84
Ident Mass	186209.40
Contingency Mass	18620.94
Residual Mass	39450.00
Cost Mass	204830.35
LOX Tank Cost	17.72
Fuel Tank Cost	10.17
Proport Cost	6.21
Scaled Cost	24.87
Engines Cost	60.00
Engine Inst. Cost	43.53
Fixed Mass Cost	28.08
*Cost	130.58
Contingency Cost	10.84
*Cost	201.42

Core Inert	
LOX Load	1448571.43
Fuel Load	241428.57
LOX Tank	14196.00
Fuel Tank	24867.14
Proport. Mass	2839.20
Scaled Mass	22530.71
Ident Mass	132210.84
Contingency Mass	13221.08
Residual Mass	15210.00
Cost Mass	145431.92
LOX Tank Cost	12.13
Fuel Tank Cost	18.28
Proport Cost	3.53
Scaled Cost	19.99
Engines Cost	80.00
Engine Inst. Cost	27.09
Fixed Mass Cost	36.66
*Cost	117.68
Contingency Cost	9.77
*Cost	207.46

PROBABILITY CURVE GENERATION

SERIAL #

33869.6382

Ascent Dv		Booster Isp		Core Isp	
23047.6965	23047	7792.10386	7792	21896.7899	21896
11703.0268	11703	485.120714	485	14167.9734	14167
8600.17777	8600	168.892825	168	5000.7866	5000
6067	6067	596	596	21129	21129
14832	14832	22872	22872	14804	14804
9424	9424	28560	28560	2664	2664
	0.00061517		0.71622656		0.27436301

Boost Eng. T/W		Core Eng. T/W		Boost Eng. Cost	
22766.3936	22766	22426.6282	22426	4815.78509	4815
14094.1856	14094	4073.74366	4073	831.270332	831
6344.7252	6344	1193.71718	1193	25.1003273	25
18554	18554	20952	20952	6102	6102
2248	2248	4254	4254	21836	21836
19310	19310	21238	21238	4250	4250
	0.32395499		0.53294939		0.06224364

Core Eng. Cost		Shroud Wt	
25667.6187	25667	30992.7746	3099
574.115263	574	27290.8688	27290
86.2739025	86	1102.51766	1102
52	52	15356	15356
7906	7906	1410	1410
14620	14620	5768	5768
	0.74472402		0.74406027

	RANDOM #		SIGMA	RANDOM # LIMITS MAX	MIN
Ascent Dv	0.50061517	0.83181584	0.00979002	0.9987	0.0013
Booster Isp	0.71622656	0.57771856	0.59238234	0.9987	0.0013
Core Isp	0.27436301	1.13723488	-0.6200919	0.9987	0.0013
Boost Eng. T/W	0.32395499	1.06167353	-0.4757857	0.9987	0.0013
Core Eng. T/W	0.53294939	0.79330248	0.09642409	0.9987	0.0013
Boost Eng. Cost	0.06224364	1.66634298	-1.5177786	0.9987	0.0013
Core Eng. Cost	0.74472402	0.54290107	0.67404844	0.9987	0.0013
Shroud Wt.	0.74406027	0.54372166	0.67211311	0.9987	0.0013

Appendix B

Equations for Assessment Tool

EQUATIONS for Lunar Orbit Rendevous

Final dry mass =	(LTV Inert Mass)+(LTV cab mass)+(payload returned)
TEI propellant =	(mass ratio -- 1)(Final dry mass)
TEI mass =	(Final dry mass)+(TEI propellant)
LEV orbit burnout mass =	(LEV inert mass)-(landing legs mass)+(LEV cab mass)+(payload returned)
LEV Ascent propellant =	(mass ratio -- 1)(LEV orbit burnout mass)
Ascent mass =	(LEV orbit burnout mass)+(ascent propellant)-(Landing legs mass)
Landing mass =	(ascent mass)+(payload delivered)--(payload returned)+(Landing legs mass)
Landing legs mass =	(landing legs fraction)(landing mass)
LEV descent propellant =	(mass ratio -- 1)(landing mass)
LEV gross =	(landing mass)+(LEV propellant)
LEV total propellant =	(LEV descent propellant)+(LEV ascent propellant)
LEV inert mass =	(1 + %growth)(LEV ident. inert mass)
MLOI =	(TEI mass)+(gross LEV mass)--(payload returned)
LOI propellant =	(mass ratio -- 1)(LOI mass)
LOI mass =	(MLOI)+(LOI propellant)
LOI/TEI propellant =	(LOI propellant)+(TEI propellant)
TLI propellant =	(mass ratio -- 1)((LOI mass)+(TLI Inert Mass))
TLI mass =	(LOI mass)+(TLI propellant)+(TLI Inert Mass)
LTV propellant =	(TLI propellant)+(LOI propellant)+(TEI propellant)+(LEV total propellant)
LTV inert mass =	(1 + %growth)((TLI ident. inert mass)+(LOI/TEI ident.inert mass) +(LEV ident. inert mass))
IMLEO =	(LTV inert mass)+(LTV total propellant)+(LTV cab mass)+(Landing mass)

Dev cost = $(\text{LTV dev cost})\left[\dfrac{(\text{LTV inert mass})+(\text{Landing legs mass})}{10^6}\right] + \text{Dev.cost}$

Dev cost per mission = $\dfrac{\text{Dev cost}}{\text{\# missions to amortize dev cost}}$

Unit cost per mission = $(\text{LTV unit cost})\left[\dfrac{(\text{LTV inert mass})+(\text{Landing legs mass})}{(10^6)\ (\text{Effective \# of vehicle reuse})}\right]$

ETO cost per mission = $\dfrac{(\text{IMLEO})(\text{ETO transportation cost})}{10^6}$

Total cost per mission = (Dev cost per mission)+(Unit cost per mission)+(ETO cost per mission)

EQUATIONS for the Direct Mission Spreadsheet

Mass ratio for return = $e^{\frac{\text{Return } \Delta V}{g * Isp}}$

Propellent for return = (mass ratio for return - 1)(Total return dry mass)

Total return dry mass = (landing stage inert mass)+(Mass of crew return vehicle)+(payload returned)

Landing stage inert mass = (1+ %growth)(lander ident. inert mass)

Landing legs mass = (Landing legs fraction)(Landed mass)

Landed mass = (Return propellant)+(Tot. return dry mass)+(Landing legs mass) -- (payload returned)+(mass of crew return vehicle)

TLI topoff + Landing delta V = (TLI ΔV)+(Landing ΔV)--(Booster ΔV)

Mass ratio for landing = $e^{\left(\frac{\text{TLI topoff + Landing } \Delta V}{g * Isp}\right)}$

Landing propellant mass = (Mass ratio for landing -- 1)(Landed mass)

Total lander propellant mass = (Propellant for return)+(Landing propellant mass)

Total boost payload = (Tot. lander propellant mass)+(Landing stage inert mass) + (Landing legs mass)+(Mass of crew return vehicle) + (Payload delivered)

Boost mass ratio = $e^{\frac{\text{Booster } \Delta V}{g * Isp}}$

Booster inert mass = (1+ %growth)(Booster ident. inert mass)

Booster burnout mass = (Booster inert mass)+(Tot. boost payload)

Booster propellant = (Boost mass ratio -- 1)(Booster burnout mass)

Total initial mass = (Booster burnout mass)+(Booster propellant)

Dev cost = $\text{(LTV dev cost)}\left[\frac{\text{(Booster inert mass)+(Landing legs mass)}}{10^6}\right]$

Dev cost per mission = $\frac{\text{Dev cost}}{\text{\# missions to amortize dev cost}}$

Unit cost per mission = $\text{(LTV unit cost)}\left[\frac{\text{(Booster inert)+(Landing stage inert)+(Landing legs)}}{10^6}\right]$

ETO cost per mission = $\frac{\text{(Tot. initial mass)(ETO transportation cost)}}{10^6}$

Total cost per mission = (Dev cost per mission)+(Unit cost per mission)+(ETO cost per mission)

	A	B
1	LENGTH	CALCULATES FUEL TANK LENGTH
2	=ARGUMENT("VOL")	Input tank volume
3	=ARGUMENT("PI")	Input pi (3.14......)
4	=ARGUMENT("DIA")	Input tank diameter
5	=(4*(VOL-(1/6)*PI*(DIA^3)))/(PI*(DIA^2))	
6	=RETURN(A5)	
7		
8	DIAMETER	CALCULATES FUEL TANK DIAMETER
9	=ARGUMENT("VOL")	Input tank volume
10	=ARGUMENT("PI")	Input pi
11	=(VOL/(5*PI/12))^(1/3)	Calculates DIAMETER=LENGTH
12	=RETURN(A11)	
13		
14	H2TANK	CALCULATES HYDROGEN VOLUME
15	=ARGUMENT("PROPMASS")	Input propellant mass
16	=ARGUMENT("MIXRAT")	Input mixture ratio
17	=ARGUMENT("ULLAGE")	Input ullage
18	=ARGUMENT("LH2DEN")	Input Liquid Hydrogen density
19	=((PROPMASS)*(1/(1+MIXRAT))*(1+ULLAGE)*(1/LH2DEN))	
20	=RETURN(A19)	
21		
22	OXTANK	CALCULATES OXYGEN VOLUME
23	=ARGUMENT("PROPMASS")	Input propellant mass
24	=ARGUMENT("MIXRAT")	Input mixture ratio
25	=ARGUMENT("ULLAGE")	Input ullage
26	=ARGUMENT("LO2DEN")	Input Liquid Oxygen density
27	=((PROPMASS)*(1/(1+(1/MIXRAT)))*(1+ULLAGE)*(1/LO2DEN))	
28	=RETURN(A27)	
29		
30	BOILO	CALCULATES OXYGEN BOILOFF
31	=ARGUMENT("DAYS")	Input number of storage days
32	=ARGUMENT("Q")	Input heat transfer
33	=ARGUMENT("SURFA")	Input tank surface area
34	=DAYS*(Q/213108.12*86400*SURFA*2.823)	
35	=RETURN(A34)	
36		
37	BOILH	CALCULATES HYDROGEN BOILOFF
38	=ARGUMENT("DAYS")	Input number of storage days
39	=ARGUMENT("Q")	Input heat transfer
40	=ARGUMENT("SURFA")	Input tank surface area
41	=DAYS*(Q/453337.4*86400*SURFA*2.823)	
42	=RETURN(A41)	
43		
44	Q	CALCULATES HEAT TRANSFER
45	=ARGUMENT("TEXT")	Input external temperature (K)
46	=ARGUMENT("T")	Input saturation temperature (K)
47	=ARGUMENT("THK")	Input Insulation Thickness
48	=1.4*(0.000000046791*(TEXT+T)*(TEXT-T)/THK+0.00000000000	
49	=RETURN(A48)	

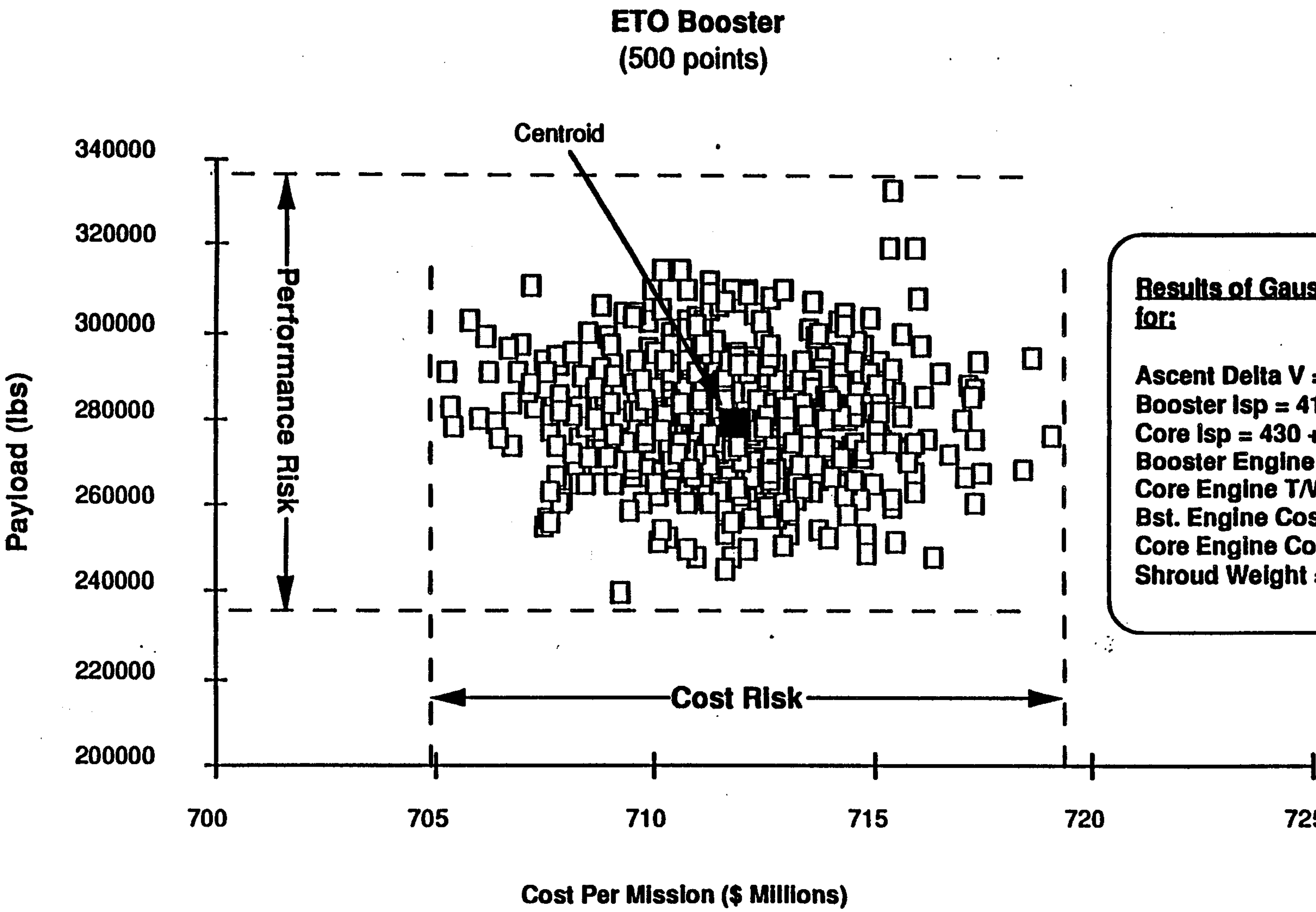
TANKT.XLM

	A	B
1	THICKNESS	CALCULATES TANK THICKNESS
2	=ARGUMENT("PRESS",1)	Input tank max. expected pressure
3	=ARGUMENT("DIA",1)	Input tank diameter
4	=ARGUMENT("STRESS",1)	Input tank material stess factor
5	=IF((1.4*PRESS*DIA)/(2*STRESS)<0.0013,0.0013,(1.4*PRESS*DIA)/(2*STRESS))	Calculate Thickness and define as > 1.3 millimeter
6	=RETURN(A5)	(Added safety factor of 1.4)
7		
8	MASS	CALCULATES TANK MASS
9	=ARGUMENT("DENS")	Input tank material density
10	=ARGUMENT("PI")	Input pi
11	=ARGUMENT("THICK")	Input tank thickness
12	=ARGUMENT("LENGTH")	Input tank length
13	=ARGUMENT("DIA")	Input tank diameter
14	=(DENS*PI*DIA*THICK*(LENGTH+DIA))*1.2	Calculate Membrane Mass * 1.2
15	=RETURN(A14)	
16		
17	INSUL	CALCULATES TANK INSULATION MASS
18	=ARGUMENT("INDENS")	Input insulation area density
19	=ARGUMENT("PI")	Input pi
20	=ARGUMENT("DIA")	Input tank diameter
21	=ARGUMENT("LENGTH")	Input tank length
22	=INDENS*PI*DIA*(LENGTH+DIA)	
23	=RETURN(A22)	
24		
25	SURF	CALCULATES TANK SURFACE AREA
26	=ARGUMENT("PI")	Input pi
27	=ARGUMENT("DIA")	Input tank diameter
28	=ARGUMENT("LENGTH")	Input tank length
29	=PI*DIA*(LENGTH+DIA)	
30	=RETURN(A29)	

Appendix C

Scattergrams - ETO Booster
- LOR
- Direct 2-Stage

Payload vs. Mission Cost

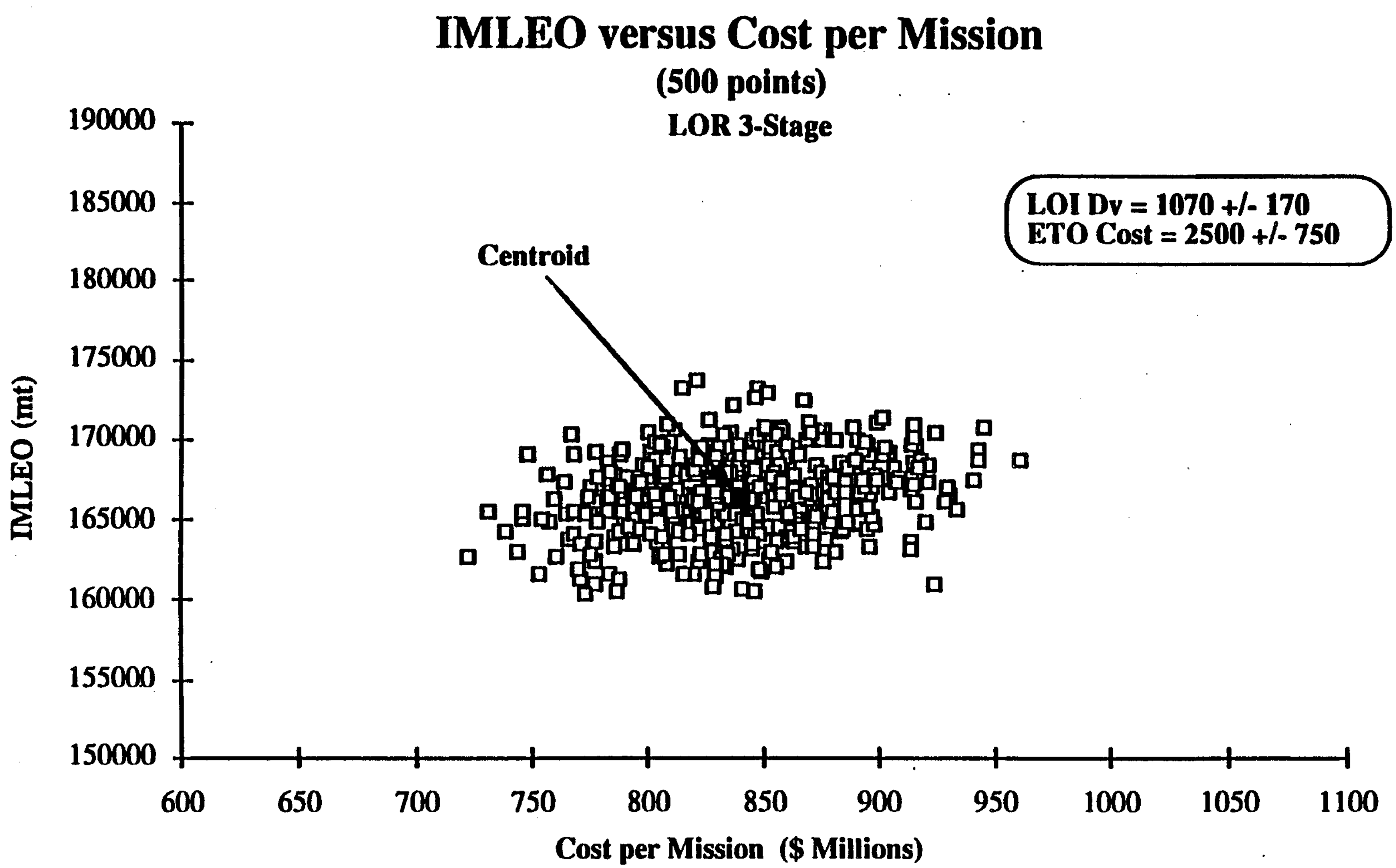


Results of Gaussian Distribution for:

Ascent Delta V = 9600 +/- 3%
Booster Isp = 414 +/- 3%
Core Isp = 430 +/- 3%
Booster Engine T/W = 72 +/- 3%
Core Engine T/W = 72 +/- 3%
Bst. Engine Cost = 20M +/- 3%
Core Engine Cost = 20M +/- 3%
Shroud Weight = 28400 +/- 3%

Lunar Orbit Insertion Delta V Sensitivity

(As related to fluid settling)



ADVANCED CIVIL SPACE SYSTEMS

Lunar Landing Delta V Sensitivity

(As related to fluid settling)

BOEING