Theoretical Analysis and Experimental Investigations of an Expander-Cycle Centrifugal Direct Injection Rocket Engine

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Abstract

A background context is given for the invention of the centrifugal direct injection rocket engine (CDIE), and a conceptual description is provided. Early parametric analysis is described, and detailed numerical simulations are used to estimate suborbital sounding rocket performance of a flight vehicle and the CDIE startup transient. A brief description is then given of the first three engines, E1, E2, and E3, with testing procedures and experimental data from engine E2 that illustrate a revised startup sequence. Finally, an overview of the current E4 development effort is given with justification for the radical changes in design.

Thesis Advisor: Manuel Martinez-Sanchez
Title: Professor of Aeronautics and Astronautics
Acknowledgments

This project started as a dream. The dream was that a group of volunteer students and engineers could develop a small rocket engine that could help enable cheap access to space. Over the past 3 years more than 100 people have worked to turn that dream into a reality. I would like to take this opportunity to recognize a few of the people who have contributed significantly to this work:

Andrew Heafitz, Byron Stanley, Sam Schweighart, Tom Nugent, Rainuka Gupta, Kris Juggenheimer, Col. John Keesee, Carrie Keesee, Alyssa Keesee, Kiyash Monsef, Col. Peter Young, Professor Jack Kerrebrock, Professor Manuel Martinez-Sanchez, Professor Ed Crawley, Professor Nam Suh, Professor Kim Vandiver, Amy Smith, the Edgerton Center, All of the MIT Rocket Team, and of course, my parents for giving me the educational resources that would allow me the opportunity to come to MIT.

You will never know how much your support and encouragement has meant to me. Thank you all. I hope we can continue to learn together.

-Carl
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Chapter 1: Background

1.1 Motivation

On November 17, 1997 two not-for-profit groups announced a $250,000 prize for the first non-governmentally developed vehicle to launch a 2kg payload to an altitude of 200km by November 8, 2000 [16]. I first heard of this “CATS prize” (Cheap Access To Space) the following summer when a friend showed me an announcement for the prize in *Scientific American* magazine. I thought that it sounded neat, but I had no intention of building a rocket at that time.

That fall I was taking a class on Rocket Propulsion taught by Professor Manuel Martinez-Sanchez. We were discussing turbo-pump technology in class when I started to think about ways to reduce the cost of a rocket engine. I thought that if the parts count could be reduced, and simple materials substituted for expensive super-alloys, there could be significant cost reduction. In addition, I thought it might be possible to dramatically simplify the geometry of many of the parts, allowing them to be built on standard, computer controlled (CNC) machine tools. Rapid, low-cost prototyping might be achievable in a turbo-pump-pressurized rocket engine. At the end of class I sketched my idea on the chalkboard for the professor. He said that it “might work”-- which was good enough for me! I became intoxicated by the idea of developing a low-cost rocket engine.

My excitement soon spilled over onto a few friends of mine, and we decided to start a student group with the goal of developing a new type of rocket engine technology and becoming the first amateur, student-run group to launch a rocket into space. The MIT
Rocket Team was born[14]. We were out to win the CATS Prize and to change the world with our low-cost engine technology.

\section*{1.2 The MIT Rocket Team and Cheap Access To Space (CATS)}

The goals of the MIT Rocket Team:

1) To take tangible steps toward decreasing the cost of space access
2) To educate the team members through fun, hands-on activities
3) To inspire non-team members to learn about rocketry and space

The overall goal of minimizing the “cost” of space access does not have a clear meaning unless some qualifiers are attached to what space access is and some figures of merit are assigned to the word “cost.” For example, if the goal is to minimize the capital cost of launching non-living payloads into orbital trajectories, various studies have shown that existing Inter-Continental Ballistic Missiles (ICBMs) could be modified to carry payloads capable of withstanding high-G loads to various orbital altitudes for very low capital cost. This scheme entirely eliminates development cost. The vehicles have already been built and in some cases fueled; all that is required is that the desired payload be adapted to the vehicle (and the trajectory must be appropriately reprogrammed).

If, on the other hand, the goal is to develop a launch vehicle that will launch any payload into orbit for the minimum sustained cost per pound of payload, other studies have shown that the solution to that cost minimization is to build a gigantic pressure-fed booster rocket[7]. These mammoth vehicles would have very high cost per launch, but
the payload would be so large that the cost per pound would be very low[7].

Unfortunately, cost-per-launch is a significant figure of merit that has prevented the development of these gigantic vehicles.

NASA has established the Space Launch Initiative (SLI) in order to develop the next-generation launch vehicle that will promise dramatically reduced cost of launching payloads into space. Since a well characterized propulsion system is critical to the design of a launch vehicle and therefore a space launch system, and since propulsion system development and characterization often takes many years to complete, it is desirable and often necessary to choose a propulsion system early in the process of choosing a system architecture.

Currently, there are many proposed system architectures for SLI, but as of this writing, the most favored propulsion system architectures employ a two stage vehicle using kerosene as a fuel and liquid oxygen as the oxidizer[11]. According to a number of system architecture studies carried out by the various SLI subcontractors, a LOX-kerosene propulsion system allows for minimum overall launch system cost[11].

The MIT Rocket Team has taken a grass-roots approach to minimizing the cost of space access. Since the project is entirely run by volunteer participants in an educational context, it has eliminated the prime cost factor of any project (monetary compensation for human work). It is therefore the epitome of low-cost space system development.

The initial concept of the Rocket Team’s engine project was to develop a small, reliable rocket engine that could be designed, tested, manufactured, and operated with a budget so small that the entire product development could be funded on the level of a student project. Initial budget estimates were approximately $285K for development of
the engine and launching a prototype sounding rocket powered by the engine as a proof-of-concept demonstration. Although this vehicle would not be capable of achieving an orbital trajectory, the engine might enable the development of small, low-capital-cost launch vehicles with multiple engines for improved system reliability and decreased development cost.

For the past three years the team has been working on the development of a small kerosene-oxygen rocket engine with the goal of dramatically reducing the cost of space access through volunteer engineering and low-cost hardware. Since that initial budget estimate was made three years ago, the team has completed fabrication and testing of three prototype engines for a total cost of approximately $95K. The explanation of that process and the preliminary design of the fourth engine is the subject of this thesis.

1.2.1 Overview of the Engine Concept

The Centrifugal Direct Injection Engine (CDIE) is a rocket engine with a spinning propellant injection manifold located inside the combustion chamber of the rocket engine. This concept combines the advantages of the turbo-pump fed liquid propellant rocket (low-pressure fuel and oxidizer tanks) in an efficient design that can be geometrically simple and cheap to construct.

The essence of the CDIE concept is a spinning fuel/oxidizer injection manifold located inside the combustion chamber of a rocket engine. This spinning manifold acts as a centrifugal pump for both the fuel and the oxidizer. The manifold contains easily machined channels that direct the flow of fuel and oxidizer in a radial direction (away from the center). The centripetal acceleration on the liquid in the channels of the
spinning manifold pressurizes the liquid as it is flung to the outer edges of the manifold—as in the rotor stage of a conventional pump. Instead of flowing from the rotor section into a diffuser section, however, the fuel and oxidizer are injected directly into the combustion chamber through small injectors located near the edge of the rotating manifold[6].

The rotation of the manifold is powered by one of three methodologies:

- Catalytic decomposition of either the fuel or the oxidizer after it has reached high pressure near the edge of the manifold – the byproducts of which are injected in a tangential direction from the edge of the disk so as to provide a torque to spin the disk and pump the fuel/oxidizer

- Vaporization of either the fuel or the oxidizer in a heat exchanger which is built into the manifold so as to take heat energy away from the combustion chamber -- the byproducts of which are injected in a tangential direction from the edge of the disk so as to provide a torque to spin the disk and pump the fuel/oxidizer

- Premixing of some or all of the fuel/oxidizer in micro-combustion chambers located in the spinning manifold -- the byproducts of which are injected in a tangential direction from the edge of the disk so as to provide a torque to spin the disk and pump the fuel/oxidizer.

The original CDIE engine concept was a modified expander cycle with a simplified geometry. Propellants from low-pressure fuel and oxidizer tanks are fed through main engine valves down through a shaft and into a rapidly rotating fuel injection manifold located inside the combustion chamber of the rocket engine. As the propellants are guided from the central axis to the outer edge of the manifold the static pressure of
the propellant rises (as in the rotor stage of a typical centrifugal pump). Instead of exhausting the propellant through a stationary diffuser to obtain a larger static pressure rise, however, one (or potentially both) of the propellants is routed through heat exchange passages on the inner surface of the rotating manifold. This geometry regeneratively cools the manifold while increasing the enthalpy of the (now vaporized) propellant. The propellant gas is then vented into the combustion chamber through small, tangentially-directed nozzles so as to impart a torque to the spinning manifold. Under the proper conditions this torque can be of sufficient magnitude to maintain a constant rotational speed, thereby pumping the propellants and achieving steady state operation[12].

The combustion chamber and nozzle of such an engine must be made of either a very high temperature alloy or an ablative material since the propellants only exist at pressures above the chamber pressure when they are inside the rotating injection manifold.

The most recent incarnation of the CDIE engine (version designator E4) is currently in the preliminary design stage. The design of this latest version incorporates a regeneratively cooled chamber and nozzle and a smaller rotating propellant injection manifold. The advantages of this new design will be expounded in chapter 5.

1.2.2 Advantages of the CDIE

The CDIE has a number of advantages over existing turbo-pump pressurized rocket engine designs. By tangentially injecting a vaporized propellant stream to provide the torque to power pumping, it is possible to eliminate the need for turbine blades. Turbine blades require particular care and attention during the design of a turbo-pump
because they are typically subjected to very high thermal and mechanical stress. In addition the fabrication of a turbine section can be very costly. The complicated geometry of turbine blades usually requires prototyping on special 4 or 5 axis CNC machine tools. Often, in production, blades are cast out of high-temperature, nickel-based super-alloys in special thermally controlled molds to ensure that the blades solidify in a single crystal formation (thereby maximizing their strength). Since the CDIE cools the entire rotating manifold with the full flow of propellants, average material temperature in the manifold can be maintained well below 600K thereby allowing the use of ultra-high strength-to-weight ratio aluminum alloys. These alloys are also very cheap and easy to machine.

In addition, because the efficiency of the force producing jets does not rely on very close, high-surface-speed tip clearances, it should be possible to make a more efficient small-scale shaft driver in a jetted CDIE configuration than could be done with a small turbine. This advantage arises because of the practical effects of realistic machining tolerances.

Since a regenerative cooling cycle is used to power the pumps in an efficient expander cycle, and that vaporized fuel is then injected into the combustion chamber at high pressure, the possibility exists for a higher fuel efficiency (specific impulse) than can be achieved in gas generator cycle.

In addition, because of the simplified geometry, a large reduction in parts count is achievable. All of these points add up to an engine that can provide good performance at a fraction of the cost of a conventional, turbo-pump-pressurized, rocket engine.
1.2.3 Disadvantages of the CDIE

Although the CDIE concept has promise, a number of disadvantages have appeared since the work began. Early on, the disadvantage of interconnectivity became apparent. Because the CDIE is an expander cycle engine, it is not possible to test the turbo-pump without actually running the engine. Although it might be argued that all expander cycle turbo-pumps suffer from this drawback, the implementation of a "fake heat source" is much easier in a typical turbo-pump because the pump is separable from the turbine. Flow can be removed from the pump at high pressure and taken somewhere else with normal tubing. In the original CDIE concept the "turbine" and the pump are much more difficult to separate because they are interconnected in the same rotating injection manifold apparatus and it is impossible to "tap" the high pressure flow from the pump because it is inside a rotating assembly. Version E4 addresses this deficiency by separating the pump from the turbine and allowing the injection of a high-pressure substitute gas to be used to drive the pump during in-lab testing.

Another disadvantage arises directly because of the elimination of the turbine blades. Since, the driving torque of the CDIE comes from jets of expanding gas, there is no direct rotational velocity feedback. In a typical turbine section if the rotor were to overspin for some reason, the turbine blades would develop a negative angle of attack which could (in extreme overspeed cases) actually cause a retarding torque. Since the CDIE is powered by jets (essentially little rocket engines) the force they produce is not limited at all by spinning faster than they were designed to go. The CDIE is therefore inherently less stable than a traditional turbo-pump. This marginal stability is compounded by the fact that the faster the disk spins, the higher the pre-injection pressure and therefore, the more
force will be produced by the jets. And, spinning the disk faster also increases the heat transfer from the combustion chamber. Since a higher pre-injection pressure is produced, more propellant is injected into the combustion chamber which also increases the chamber pressure and therefore the heat transfer. There was significant concern as to whether or not the entire system was stable. Detailed analysis and simulation using the best heat transfer models available suggests that the system is stable, but only marginally. It was decided to experimentally investigate the stability since an insufficient database exists to accurately predict the actual heat transfer with a cold, centrifugating boundary layer. The system also appears to be very sensitive to bearing and windage losses, although these effects could be simply practical problems with our current embodiment. A more quantitative stability analysis is presented later.

E4 will address this marginal stability issue by relying on heat transfer through the chamber and nozzle to vaporize the propellant. Since the speed of the gas flowing over the heat transfer surface will not increase linearly with disk speed, the heat flux will not be as directly increased by an increase in rotor speed. This should allow for a more predictable stable operating point. Even E4, however, will not have as clear a stable operating point as a turbine driven system.

Overall, the CDIE appears to be a tradeoff between cost and complexity. Although it is incredibly cheap to manufacture, the inherent complexity associated with the analysis and implementation of a highly interconnected system makes for a very difficult engineering problem. Fortunately, as student volunteers, our time is free, and we can learn a lot from working on a difficult problem like the CDIE. At the time of this writing
it is still unclear whether all of the difficulties associated with the CDIE can be simultaneously overcome in a flight-weight engine.

### 1.3 Early Work

#### 1.3.1 Initial Investigations

The first thing our newly formed team did was to investigate background information on this technology concept. At that time, we were not sure if we actually were working on something new. Our literature searches were not able to find any work that had been done on our engine concept so I named the concept Centrifugal Direct Injection Engine technology\(^1\).

The next step was to develop some simple analysis and design concepts that we could use to raise the funds necessary to start experimental investigation. I developed a spreadsheet to calculate basic engine parameters and I wrote a 3DOF launch vehicle simulator to parametrically analyze the vehicle and engine sizing that would be necessary to accomplish the CATS prize goal of launching a 2kg payload to a 200km altitude on a sub-orbital trajectory.

The results of this analysis were optimistic. It appeared that if we could develop a small, turbo-pump pressurized rocket engine, the structural weight savings afforded by the low pressure fuel and oxidizer tanks would allow the overall vehicle to be small enough (~40 kg) that we could afford to build the vehicle with a budget comparable to

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\(^1\) Since that time a more detailed patent search has revealed that Dr. Robbert Goddard worked on a similar technology [13], but our work is improved on his because we entirely avoid the fabrication of turbine blades, and we incorporate the use of regenerative cooling inside the rotating injector head.
other student-group projects (< $100,000). In addition, the small vehicle size would allow the development of a low-thrust main engine (< 2000 N) which would reduce engine development costs because of the smaller infrastructure requirements.

Based on those preliminary analyses, we were given initial funding by Professor Ed Crawley, Aeronautics and Astronautics Department Head and Professor Nam Suh, head of the Department of Mechanical Engineering. 2

1.3.2 The First Prototype Hardware

Given our newfound funding, it was initially decided to attempt a more rigorous numerical approach to the engine development. We wanted to investigate the possibility of performing detailed computational fluid dynamic modeling (CFD) of the engine. Discussions with professors in the department 3 and an early attempt at a simplified axisymmetric CFD model 4 lead to the conclusion that many orders of magnitude more computing power would be required in order to develop a CFD model which we would actually trust.

It was decided that a "proof of concept" device should be constructed in an attempt to both demonstrate the feasibility of the concept and to begin development of a practical knowledge base. The device (hereafter "the prototype") was a 6 inch diameter, 3 layer aluminum disk with concentric liquid nitrogen and water feed lines running axially through a bearing shaft. An apparatus was constructed so that water and liquid

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2 In 2000 the Rocket Team also received funding from United Technologies Corporation and the MIT Edgerton Center.
3 Pers Comm, Professor Manuel Martinez-Sanchez and Professor Jack Kerrebrock, 1999
4 An early axisymmetric model was attempted by Sumita Pennathur in January 1999 using the commercial code FLUENT, but work was abandoned when it became clear that the model was oversimplified to the point of not being useful.
nitrogen would be gravity-fed into the spinning disk and blow-torches could be raised so as to impinge on the bottom of the spinning disk, thereby simulating the heat transfer from the combustion chamber.

Steady state operation of this prototype device was never achieved, but we were able to measure a statistically significant change in the spin-down time when blow-torches were on and LN2 was flowing. Although, from a scientific perspective, this device was nearly a complete failure, I believe it was a critical step in our development of the final engine and our development as engineers. It allowed us to gain a practical understanding of many of the issues that we would later encounter in the design and development of the engine such as liquid feed systems, cryogenic feed systems, and high-tip-speed device construction.

Despite the poor performance of the prototype (which is directly attributed to our poor engineering analysis of the device), we decided to proceed with the development of a test engine (E1) during the fall of 1999 based on the lessons we had learned during the prototype trials.

1.4 CDIE-E1

Most of the details of the design and testing of our first rocket engine (termed E1 for engine number one) is contained in the Master’s Thesis of Andrew Heafitz[4]. In this section I will attempt to summarize some of the design choices and lessons learned with the E1 engine without duplicating the information presented in Andrew’s Thesis.
1.4.1 Propellant Selection

It was decided early on that Engine El should be as similar to the final, desired engine as possible. It was therefore necessary to make a number of decisions for the vehicle development program early on. Early analyses compared the effects of different fuel choices on the size of the vehicle necessary to deliver the specified 2kg payload to an altitude of 200km.

The selection of fuel and oxidizer was primarily determined by a subjective compromise between practicality and performance. Although liquid hydrogen is arguably the best chemical fuel from the perspective of performance, its use for this project would have proven impractical. As we discovered during our prototype tests, the design of small cryogenic flow systems is always a challenge, but it would be an even greater challenge for a liquid hydrogen system for the following reasons:

- Liquid hydrogen will burn in air, and the flame has no color to it, making it very hard to see and inherently dangerous.

- Hydrogen is liquid at temperatures below 20K at atmospheric pressure. These temperatures are so low that a liquid hydrogen line that is not perfectly sealed and insulated will liquefy and then solidify the air around it. High concentrations of frozen oxygen from the atmosphere could build up inside the insulation around the lines – thereby creating a potential fire or explosion hazard.

In addition the low density of liquid hydrogen would also prove to be difficult to cope with in the design of a single stage fuel pump – necessitating much higher tip speeds than
necessary with other, more conventional (higher density), fuels. For this reason, and the reasons cited above, hydrogen was eliminated from consideration.

From the perspective of performance, the chemical fuel that will provide the “next best” specific impulse is methane – the lightest hydrocarbon. Liquid methane is the most abundant hydrocarbon in what is commercially known as “Liquefied Natural Gas” or LNG. Cryogenic LNG was never seriously considered as a fuel for E1, however, because kerosene can provide nearly the same performance as LNG and kerosene is a room-temperature storable fuel. Due to the simplified handling and availability, kerosene was chosen as the fuel for engine E1 and for the flight vehicle. Hydrazine and its derivatives were not seriously considered because of storage and handling issues associated with their use.

Oxidizer selection was not as easy. Again from a specific impulse perspective, liquid fluorine is the best possible oxidizer. Due to its intense reactivity and toxicity, however, it was never seriously considered for E1. Liquid oxygen is the “next best” oxidizer. Despite its low liquefaction temperature of 90K at atmospheric pressure, liquid oxygen has an extensive history as a rocket fuel oxidizer. It is not as dangerous a cryogen as liquid hydrogen because it will not react with air, and since it poses no environmental threat, tanks can be safely vented to the atmosphere. Other potential oxidizers that are appealing because they are room temperature storable liquids include nitric acid, nitrogen tetroxide, and hydrogen peroxide. Of these only hydrogen peroxide was seriously considered because nitric acid and nitrogen tetroxide both have special OSHA recommended exposure limits and environmental concerns that we did not want to need to worry about. In the end, concerns about the availability and stability of high
concentrations of hydrogen peroxide (it can spontaneously, exothermically decompose) and the higher performance of pure liquid oxygen resulted in the selection of liquid oxygen as the oxidizer.

1.4.2 Mission Definition

The CATS Prize specified that a 2 kg payload must be lofted to an altitude of at least 200km. A first order analysis was conducted early on to attempt to bound the problem from a performance perspective. Neglecting drag and gravity losses, the impulsive delta V required to loft something to an altitude of 200km is approximately 2 km/s. Based on parametric launch vehicle analyses I conducted during the summer of 1998 I knew that drag losses for a launch vehicle small enough to be built at MIT could be anywhere from 1 to 2 km/s. And a first order estimation of the gravity loss is just 9.81 m/s/s times the burn time of the engine. Based on these first-order estimates, the delta V of the mission was bounded between 2600 m/s and 5000 m/s.

Using the conservative 5000 m/s as a baseline, and assuming an exhaust velocity of 2500 m/s, the a mass-ratio (Mo/Mf) of 7.4 is achieved. This ratio would be a challenge, but it did not seem unreasonable for a vehicle with unpressurized tanks. Even if a mass ratio of only 5 could be achieved, we would still have over 4000 m/s delta V. Based on these numbers (assuming a mass ratio of 5) a simplified 1-D simulation incorporating the standard atmosphere was developed to more precisely quantify drag and gravity losses while varying initial thrust-to-weight (burn time). This simulation suggested that a longer burn (which would arise from having a low initial thrust-to-

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5 A numerical analysis of potential engine performance was conducted using NASA’s Chemical Equilibrium Analysis (CEA) code comparing the performance of 70%, 80%, 90%, and 100% concentrations of H2O2 with the performance of LOX. These analyses quantified the potential performance difference.
weight ratio) would minimize drag loss in the atmosphere and allow a higher apogee.

The following page contains the results of that 1-D analysis.
Figure 1: Apogee v. T/W for mass ratio of 5, 255s Isp, and 90 degree launch angle

Figure 2: Total lost delta V for mass ratio of 5, Isp of 255s, and 90 degree launch angle
At the time that basic analysis was conducted (January 1999), it was assumed that we could find a place to launch the rocket vertically. Unfortunately, it is unlikely that we will be able to launch vertically and even small angle deviations dramatically shift the optimum T/W. But at that time, it seemed like a reasonable assumption, so the decision was made to proceed with the design of a vehicle that would contain an initial thrust-to-weight ratio of 2. This low thrust was nice because it meant that significantly reduced testing infrastructure would need to be developed. Our baseline design called for a small vehicle that had an empty mass of 8kg (2kg payload, a 2kg engine and 4kg of structure) and a fully fueled mass of 40kg. The design thrust of the engine was therefore determined to be approximately 800N. That baseline called for an engine with a thrust-to-weight ratio of 40 --which seemed reasonable given the cube-square scaling of turbo-pump driven rocket engines[8].

That was the design that was initially presented, and that was the design that was funded. It was not until after quite a ways into the design cycle, when I started to develop more elaborate 3 DOF simulation routines, that I discovered that launching off-vertical would cause severe problems due to rapid pitch-over at low speed and low altitude. If the CATS Prize were still around, and we were starting again from scratch, knowing that the maximum launch angle for an unproven rocket is 80 degrees at NASA’s Wallops Flight Facility (for practical reasons, the most likely launch range), we would want to develop a vehicle with a thrust-to-weight ratio around 5, but even then, with a mass ratio of 5 and a specific impulse estimate of 255 seconds, a maximum altitude of around 45 kilometers is the most we could hope to achieve with the small-scale vehicle we were hoping to build.
A solution to this problem that we came up with out of desperation was to attach a solid-fuel booster section to our vehicle. High-powered amateur rocketry companies such as Aerotech Rocketry[17] build and sell large, solid propellant rockets. With the addition of three of these large motors in a booster section, our vehicle could, once again, achieve an altitude greater than 200km and we could continue building an engine and vehicle of a reasonable scale on the budget of a student group.

Figure 3: Apogee v. T/W for mass ratio of 5, Isp of 255s, and launch angle of 80 degrees
Figure 4: Three DOF simulation of 40kg GTOW sustainer rocket with 800N main engine and a booster section of three AeroTech M1315W solid propellant motors.\(^6\)

This vehicle was the design we intended to launch for the CATS Prize. More information about the preliminary design of this vehicle is contained in a document presented on February 28, 2000 at NASA’s Wallops Flight Facility.\(^7\)

\(^6\) A 6 DOF model of the vehicle was also prepared for NASA by Hayden Huang [10]. This model indicated that due to excessive angles of attack and violent pitching moments during reentry, breakup of the vehicle would be likely.

\(^7\) “MIT Rocket Team: Proposal for Launch at NASA’s Wallops Flight Facility” [10] is available by request from the author. The document is the product of the effort of many Rocket Team members.
Although mistakes were made along the way, the conclusion of this mission analysis was that it is possible to develop a small sounding rocket that will take a 2kg payload into space on something close to our original project budget of $285K. It was determined that the paramount obstacle in the path of accomplishing this project goal was the development of an 800N thrust CDIE engine.

1.4.3 General Design Characteristics of the Early Engines

During the early phases of the hardware design of CDIE1 (E1) a number of critical parameters needed to be isolated. We had a concept of how the engine should work, but since this would be the first build of this new type of engine, we did not know where to begin. At that time, I started the development of a complicated dynamic computer simulation of the engine. The simulation would take geometric inputs (injector hole sizes and locations, assumed discharge coefficients, throat diameter, expansion ratio etc.) and combine them with a 1-D dynamic model of the rotating propellant injection manifold to create a complete model of the engine function. I was quickly overwhelmed by the infinite variety of possible inputs. Based on the advice of Professor Jack Kerrebrock and Professor Manual Martinez-Sanchez, it was decided that a better tactic would be to reverse the design process. The design started by assuming that we want an engine that will produce approximately 200 lbf thrust using an optimum mixture ratio of kerosene and LOX. We then assumed a reasonable design chamber pressure (high enough to give us an advantage over pressure fed engines, but low enough so that we are not pushing the limits of the heat flux that can be absorbed by the silica-phenolic chamber-nozzle material i.e. below the fastrac chamber pressure).
When those limitations were imposed it was decided that a chamber pressure of approximately 35 atm (515 psi) would be the starting point of the design. An analysis conducted with NASA’s CEA (Chemical Equilibrium Analysis) program determined the optimum mixture ratio at that chamber pressure to be 2.5 (O/F). In theory that combination could result in a vacuum specific impulse of over 300s. Consultation with the engineers at NASA’s Marshal Space Flight Center who had been working on the development of their silica-phenolic chamber nozzle resulted in a slight decrease in the design mixture ratio to a value of around 2.3. The reason for this shift off of optimum is that a 10% fuel-film cooling layer is required to decrease the heatflux to the chamber-nozzle wall. That film should remain unburned until outside the nozzle. This film cooling imposes a performance penalty on engines of this design, but according to the NASA engineers, it is necessary to maintain an adequate thermal/structural margin of safety. The fuel film dramatically reduces the oxidation rate of the chamber-nozzle, and since we were not interested in pushing the state-of-the-art in chamber-nozzle design, we decided to follow the suggestions of the NASA engineers.

At that point we were able to solve for all of the relevant design flow parameters. The massflow was supposed to be .33kg/s (.23kg/s oxygen and .10 kg/s kerosene), and the throat diameter would be just over ½ inch. We were now left with the problem of designing the rotating manifold so that it would supply those quantities of propellants.

The first issue that we came across in this process was that we needed to have sufficient surface area on the rotating manifold so that we would transfer enough heat to drive the engine. A literature search revealed an absence of research into heat transfer from combusting gasses to cold rotating disks, but there has been significant research into
heat transfer from hot rotating disks to cool gas (as can be found in the turbine disks of
gas turbine engines.) I spent some time mapping the Bartz model for heat transfer in
rocket engine combustion chambers onto a rotating disk and then comparing those results
to the results that I got from the gas-turbine models. The results were close enough that
in the end, it was decided that we would use these models since we did not wish to
theorize on how a cold, centrifugating boundary layer would affect the Stanton number.

Based on the heat transfer models that we had found it was possible to develop a
relationship between the total heat flux through the disk and the tip speed. A higher tip
speed could result in a proportionally higher heat flux and therefore a smaller disk for our
given quantity of propellant flow, but the stresses inside the disk go with the square of the
tip speed. Since we knew that we did not have the engineering manpower to devise an
optimum design solution to the heat transfer/structural trade off, it was decided that we
would keep our tip speed low – even below the speed at which a thin ring of material
would burst. This design trade concession resulted in an abnormally large disk-to-throat
area ratio which is one of the fundamental disadvantages of the Centrifugal Direct
Injection Engine.

The next step in the design cycle was materials selection. The desirable
engineering characteristics for the rotating propellant injection manifold are high yield
strength to density ratio, high pressure oxygen compatibility, and high thermal
conductivity. The best material given only these constraints is diamond, but since
diamond was obviously out of the question for reasons of cost and fabrication, the 7075
alloy of aluminum was chosen for most of the rotating structural components. In areas
where heat transfer was not desirable 316 stainless steel was used in conjunction with
insulating vacuum or air gaps. Because of the short 5 second burn duration required for testing, it was decided that a large copper block would be an adequate heat-sink, so the combustion chamber was machined from copper. Low stress parts would be made from 6061 aluminum, bronze, stainless steel or teflon depending on the desired thermal conductivity and expected local wear.
Chapter 2: Parametric Modeling of the CDIE

For purposes of analysis it is helpful to define a series of locations in the engine at which the physical properties of the propellants are to be determined. These locations will be referred to with the following numerical subscripts:

Figure 4: Schematic Modeling of CDIE
**Definitions:**

\( \rho = \) Density

\( \omega = \) Angular Velocity of Propellant Injection Manifold

\( P_x = \) Pressure at point \( X \)

\( T_x = \) Temperature at point \( X \)

\( P_c = \) Chamber pressure

\( T_c = \) Chamber temperature

\( O/F = \) Mixture ratio (by mass)

\( M = \) Mach number

\( C_p = \) Constant pressure specific heat

\( R = \) Gas constant

\( r_x = \) Radius at point \( X \)

\( \gamma = \frac{C_p}{C_v} = \) Ratio of Specific Heats

\( Q = \) Heating Power

\( q = \) Heating Power per unit area

\( \Delta H_{f\to s} = \) Heat of Vaporization of Oxygen

\( T_{net} = \) Net Torque on Rotating Manifold

\( m_{o} = \) Oxygen Massflow

\( m_{f} = \) Kerosene Massflow

**Analysis:**

It will be assumed that both liquid oxygen and kerosene are incompressible fluids so the pressure at point 2 in both the LOX and kerosene circuits can be described by

\[
P_2 = \frac{1}{2} \rho \omega^2 r_x^2 + P_1
\]

(1)

And it will be assumed that the channels from point 1 to point 2 are adiabatic,

\[
T_2 = T_1
\]

(2)
The flow velocity of the kerosene into the combustion chamber is therefore determined by the angular velocity of the disk and the chamber pressure

\[ u_{f_{\text{inj}}} = \sqrt{\frac{2 (P_{2f} - P_c)}{\rho_f}} \]  

(3)

The effective area of the fuel injector is then

\[ A_{\text{eff}} = C_d A_f \]  

(4)

Where \( C_d \) is a non-dimensional discharge coefficient of the kerosene injector nozzle and \( A_f \) is the cross-sectional area of one kerosene injection nozzle. A discharge coefficient of 0.7 was assumed. Discharge coefficient data was taken from Sutton p.304 [1]. The kerosene massflow is then given by

\[ \dot{m}_f = n_{f_{\text{inj}}} \rho_f A_{\text{eff}} u_{f_{\text{inj}}} \]  

(5)

where \( n_{f_{\text{inj}}} \) is the number of kerosene injection nozzles. Equations (1) through (5) completely describe the kerosene injection as a function of \( \omega \), the angular velocity of the disk, and \( P_c \), the chamber pressure (given a series of assumed constants that describe the injector area, the number of injectors, and the discharge coefficient of the injectors).

The oxygen flow cannot be modeled as simply. For the purpose of this analysis we will assume that the specific heat and the heat of vaporization of oxygen are constant with temperature and pressure. These assumptions allow the application of the following equation (assuming all vaporization happens between points 2 and 3).

\[ \dot{Q} = m_{\text{ox}} (\Delta H_{fg,ox} + C_p (T_4 - T_3)) \]  

(6)

It will also be assumed that \( T_3 = T_2 = T_1 = \) temperature of boiling liquid oxygen at atmospheric pressure (90 K). Equation (6) can then be rewritten to solve for \( T_4 \).
\[ T_4 = 90 + \frac{\dot{Q} m_{ox} - \Delta H_{fg,ox}}{C_p} \]  \hspace{1cm} (7)

If, however,
\[ \dot{Q} m_{ox} - \Delta H_{fg,ox} \leq 0 \]  \hspace{1cm} (8)

the flow will be mixed phase or a pure liquid. Since such a mixed phase flow is very difficult to analyze, it will be assumed that if there is any mixed phase flow at point 4, the oxygen flow will be treated as a purely incompressible liquid flow and equations similar to equations (3) and (4) will be applied to the oxygen flow with an assumed discharge coefficient. Although this is certainly inaccurate, it will provide bounds for the mixed phase oxygen massflow. Discontinuities in the analysis will be clearly apparent where the transition occurs.

The pressure of the oxygen in this model is given by the following equations:
\[ P_2 = P_i + \frac{1}{2} \rho_{LOX} \omega^2 r_2^2 \]  \hspace{1cm} (9)

In order to reduce computation time the following assumption is imposed on the oxygen flow model. In the CDIEsim code which will be described later, a more accurate model of the pressure drop is employed.
\[ P_4 \approx \frac{2}{3} P_2 \]  \hspace{1cm} (10)

This assumption should be a reasonable approximation for the pressure drop through the cooling passages for the design massflow rate, and by eliminating the necessity to solve for that pressure drop exactly, computation time is greatly reduced.

The condition for choked oxygen flow into the combustion chamber is checked
\[ \frac{P_4}{P_c} \geq \left[ \frac{(\gamma+1)}{2} \right]^{\gamma/(\gamma-1)} \]  \hspace{1cm} (11)
Where $\gamma$ is the ratio of specific heats for gaseous oxygen (1.4). If the condition given in equation 11 is true, the oxygen flow is choked and the oxygen massflow is given by

$$m_{ox}^* = \frac{P_4 A_{oxinj}}{c_{ox}^*}$$  \hspace{1cm} (12)

Where $A_{oxinj}$ cross sectional area of the throat of the oxygen injector nozzle and $c_{ox}^*$ is given by

$$c_{ox}^* = \frac{\sqrt{\gamma R_{ox} T_4}}{\gamma \sqrt{\left( \frac{2}{\gamma+1} \right)}} \left( \frac{\gamma+1}{\gamma-1} \right)^{(\gamma+1)}$$  \hspace{1cm} (13)

and where $R_{ox}$ is the gas constant for oxygen.

If the condition given in equation 9 is false, the oxygen massflow is found using the following equations:

$$m_{ox}^* = \rho_{ox} A_{oxinj} u_{ox}$$  \hspace{1cm} (14)

$$\rho_{ox} = \frac{P_c}{R_{ox} T_4}$$  \hspace{1cm} (15)

$$u_{ox} = \sqrt{\frac{2(P_4 - P_c)}{\rho_{ox}}}$$  \hspace{1cm} (16)

Inserting the values from equations (15) and (16) into equation (14) will give an approximate massflow of oxygen neglecting the effects of viscosity and compressibility.

The previous equations for massflow of kerosene and oxygen have assumed a constant chamber pressure, but the chamber pressure is a function of the massflow. It is therefore necessary to solve for the chamber pressure iteratively by using the following constraint.
\[ m'_{\text{tot}} = m'_{\text{ox}} + m_f = \frac{P_c A_t}{c^*_{\text{chamber}}} \]  

\( A_t \) in the above equation is the cross-sectional throat area of the rocket engine and \( c^*_{\text{chamber}} \) is the characteristic velocity of the chamber gas at combustion chamber temperature, \( T_c \).

The combustion chamber temperature is assumed to be purely a function of the mixture ratio \((O/F)\) which is equal to the oxygen massflow divided by the fuel massflow.

It is therefore possible to iteratively solve for the massflows of each of the propellants (and therefore the mixture ratio), the chamber pressure, and the chamber temperature simultaneously.

The thrust and specific impulse of the engine are then estimated using 1-D supersonic flow equations and a constant ratio of specific heats. The exit Mach number is iteratively solved from a given expansion ratio (a function of the geometry of the nozzle) using

\[
\frac{A_e}{A_t} = \frac{1}{M_e} \left[ 1 + \frac{(\gamma - 1)}{2} M_e^2 \right]^{\frac{(\gamma + 1)}{2(\gamma - 1)}}
\]  

A ratio of specific heats is interpolated from a plot of specific heat ratio versus mixture ratio. From that exit Mach number it is possible to calculate the temperature and pressure of the exit gas from

\[
T_e = \frac{T_c}{1 + \frac{(\gamma - 1)}{2} M_e^2}
\]
\[ P_e = \frac{P_e}{\gamma^2} \left( \frac{T_c}{T_e} \right) \]

(20)

It is then possible to get the exit velocity from

\[ u_e = M_e \sqrt{\gamma RT_e} \]

(21)

The thrust is then given by

\[ F = m_{i\infty} u_e + A_e \left( P_e - P_a \right) \]

(22)

Where \( P_a \) is the ambient pressure (assumed to be 1 atmosphere). The specific impulse (Isp) of the engine therefore can be estimated by

\[ I_{sp} = \frac{F}{m_{i\infty} g} \]

(23)

Where \( g \) is the acceleration of gravity at sea level.

It should be noted that the thrust and specific impulse that are calculated using these techniques are not conservative estimates. They assume complete combustion of the fuel and oxidizer inside the combustion chamber. The actual engines EI-3 were designed to employ a fuel rich film cooling layer along the inside wall of the combustion chamber in order to attempt to control the ablation and erosion which are expected to occur at the throat entrance. The presence of unburned fuel in the exhaust gas will significantly reduce the specific impulse of the engine.
Figure 5: Isp (s) v. Heat Flux (W/m²) and Rotational Speed (rad/s).

With the massflows and chamber pressure already solved, it is possible to calculate the net torque on the disk. For this preliminary analysis the effects of bearing loss was estimated from data collected during initial proof of concept tests. Aerodynamic drag on the disk was ignored. It is assumed that the power lost to drag is much less significant than the power of the jets and the pumping power.

The torque balance can then be written

\[ T_{\text{net}} = F_{\text{oxygen}} r_{\text{oxygen}} - m_{\text{o}} \omega_{\text{o}}^2 - m_{\text{f}} \omega_{\text{f}}^2 - B\omega \]  

(24)

where B is the bearing loss term and the force provided by the oxygen jets is
\[ F_{\text{ox},\text{rot}} = \dot{m}_{\text{ox}} u_{\text{ox}} \]  

(25)

for the subsonic case and

\[ F_{\text{ox},\text{rot}} = \dot{m}_{\text{ox}} u_{\text{ox}} + A_{\text{ox,\text{inj}}} (P_e - P_c) \]  

(26)

for the supersonic case. \( u_{\text{ox}} \) in equation (25) is the \( u_{\text{ox}} \) given in equation (16) and \( u_{\text{ox}} \) in equation (26) is found by assuming that the flow is choked with no expansion, therefore the flow is sonic:

\[ u_{\text{ox}} = \sqrt{\gamma RT_4} \]  

(27)

The jet exit pressure, \( P_e \), in equation (24) is then given by

\[ P_e = P_4 \frac{1}{\left(1 + \frac{(\gamma - 1)}{2}\right)^{(\gamma - 1)/\gamma}} \]  

(28)

Using equations (1) through (28), all of the torques and forces inside the engine can be calculated with an iterative solving routine that solves simultaneously for the chamber pressure and the massflows given any state of heat flux and angular velocity.

A routine was developed in Matlab which solves the system as described above. The results were plotted on a 3-D grid using the angular velocity, \( \omega \) and the heat flux as the two independent variables.
Figure 6: Thrust (N) v. Heat Flux (W/m$^2$) and Rotational Speed (Rad/s)

The figure above shows the variation in thrust with disk speed and heat flux for a specified geometry. The following figure shows the variation in net torque on the disk in the same regime. The intersection of this plot with the zero net torque plane indicates the (theoretical) stable operating points of the engine given the geometry specified.
Figure 7: Net Torque (Nm) v. Heat Flux (W/m^2) and Rotational Speed (Rad/s)

The purpose of this section was to illustrate the preliminary method of analysis used on the Centrifugal Direct Injection Engine. As displayed by the results of the analysis in figure 7, there are definite regimes of rotational speed and heat flux where the engine could function at a steady state (with zero net torque on the disk). This regime of steady state operation is characterized by very cold (near mixed phase) oxygen injection flow. In the actual engine, there will be bearing and windage losses which will decrease the net torque as the rotational speed increases. These effects should cause the stable operating oxygen flow to be gaseous instead of mixed phase. It is important to note,
however, that the engine does not rely on the effects of the losses in order to achieve stable operation. That said, it is still necessary to show that the engine will operate with an acceptable margin of stability at the operating points suggested by the previous figure. We will now proceed with the analysis of the stability of the system.

A first cut stability analysis was performed during January 1999. The method that was employed made three large assumptions that are not true of the actual system in order to make the mathematics manageable. The goal of the method was to illustrate graphically the variation in the net torque on the disk with the angular velocity. If the assumptions are made that the heat flux does entering the disk does not vary significantly with omega and that the chamber pressure is constant and that the oxygen injector jets are choked, it can be shown that the net torque on the disk is of the form

\[ T_{\text{net}} = a\omega^3 + b\omega^2 + c\omega + d \]  

(29)

where the coefficient “a” is a negative real number. This case is very similar to the more realistic model presented in the previous section as can be seen by taking a slice of the graph in figure 7 with a constant heat flux. The limit as the rotational speed goes to infinity is an infinitely negative net torque is placed on the disk. This negative net torque would cause the disk to spin down until an area of positive net torque is reached. The largest zero of equation (29) (at the appropriate heat flux and chamber pressure) is the stable operating point of the system.

There are conditions of heat flux and chamber pressure which do not yield any region of positive net torque (as can be seen by looking at the low heat flux values in figure 7), but the graphical analysis suggests that there is just one minimum value of heat flux at a given speed below which the net torque is always negative and above which the
net torque appears to be always positive. This model was verified with a dynamic simulation of engine operation that had a floating chamber pressure but assumed a constant heat flux.

A more complete stability analysis attempt was also conducted. One component of this stability analysis was the generation of the figures presented earlier in the section. These figures which treat the heat flux as an independent variable allow the visualization of the slope of the zero torque stable operating point line discovered in the previous analysis. It is postulated that if the actual disk heat transfer has a variation with disk speed that is less than the slope of the zero net torque line over the entire potential operating range of the engine, the engine will have one stable operating point.

Research was conducted into the field of heat transfer into rotating disk systems. Most of the literature revolves around heat transfer from a hot turbine disk to cooler air. Since the exact model of heat transfer into a cooled rotating disk in a rocket combustion chamber appears to be unexplored at this time, a generic model for heat transfer from a rotating disk has been employed as described in “Flow and Heat Transfer in Rotating Disk Systems” by J.M.Owen and R.H.Rogers. In this model the local turbulent heat flux is given by

\[
\dot{q} = 0.0238 \Pr^{3/5} \Re_{\phi}^{4/5} x^{8/5} \frac{k\Delta T}{r}
\]

where \(x\) is the non-dimensional radius \((r/R)\), \(r\) is the local radius, \(R\) is the radius of the disk, and \(\Re_{\phi}\) is the rotational Reynolds number

\[
\Re_{\phi} = \frac{\rho \omega R^2}{\mu}
\]
Since the chamber density, \( \rho \), is also a function of the rotational speed, \( \omega \), because of its dependence on \( P_c \), and \( P_c \)'s dependence on the square of \( \omega \), the derivative becomes
\[
\frac{\partial \dot{q}}{\partial \omega} = C_1 \omega^{7/5}
\] (32)

where \( C_1 \) is a constant that must be solved for because it depends on the chamber pressure and the massflows. The value of that derivative is compared to the slope of the zero net torque line to complete the stability analysis.

A similar approach to examining the stability of the CDIE is to compare the rate of variation of driving power to the rate of the variation of pumping power. For this analysis, we use the simplistic assumption that the driving power is directly proportional to the heat flux.

\[
P_{\text{drive}} \propto \dot{q} \propto \omega^{12/5}
\] (33)

Then since we know that for constant massflow the pumping power goes as
\[
P_{\text{pump}} \propto \omega^2
\] (34)

it is clear that by this analysis the engine would be unstable to small variations in omega because the driving power would increase faster than the load power. However, when the effects of variations in massflow, mixture ratio, bearing loss, disk windage, and non-ideal efficiencies in power transfer from heat to the driving jets are added, the engine is stable, as shown by the complete dynamic analysis, but the stability is only marginal. This marginal stability could result in large uncertainty in the actual operating speed of the engine. Because of this uncertainty, it was decided that all hot-fire tests would be conducted with an 8kW electric motor attached to the shaft that had the ability to
inductively brake the shaft and dump excess power into the batteries in case the actual operating point was not where it was expected to be.

A more complete dynamic simulation of the engine with full variation of heat flux as described in equation (30) is described in the following section. This model was used to help isolate the geometry of the first 3 builds of the engine and to help determine the timing of the startup sequence.
Chapter 3: Numerical Tools for Design and Analysis

In the process of designing this new type of engine it became necessary to develop predictive numerical models and dynamic simulations of the engine. These routines varied significantly in complexity, but were based on the simple equations of fluid motion, heat transfer, and thermodynamics. The code for three such routines is included in the appendices. This section will present many of the fundamental calculations and assumptions that were made in the generation of these models.

The first model that is presented is the 3 degree of freedom vehicle trajectory simulation some of the results of which were presented earlier. This routine was used in place of the more robust 6DOF routine developed by Hayden Huang [8] to rapidly evaluated launch vehicle trajectory profiles for a given launch angle. The relatively short run time of the code enabled multiple iterations per day. It also served as a good baseline cross-check to the more detailed 6DOF code.

The second model is the most detailed dynamic simulation that has been developed for predicting the startup and steady-state operation of the CDIE engine. It incorporates the effects of variable mixture ratio during startup, the Owen and Rogers heat flux model of equation (30), and the measured effects of bearing loss and injector discharge coefficients. The development of this piece of code inspired confidence that the engine would have a stable operating point, even if the actual steady-state operating point would be difficult to predict due to the marginal stability of the design.
3.1 “QuickSim” 3 Degree of Freedom Trajectory Simulator

During the early phases of the project it became clear that it would be necessary to develop a code that would give a reasonably accurate estimation of the maximum achievable altitude with a given vehicle and engine launching from a given angle. This code would also be useful in cross-checking the predictions of the 6DOF simulation code. Consequently, I undertook the development of a 3DOF flight vehicle trajectory simulator dubbed, “QuickSim.”

QuickSim was intended to be a very simple simulation routine employing an Euler integration scheme with a fixed time step. The time step would be adjusted and the results compared to check for convergence. The code was designed to be very easy to understand and debug so that other volunteer team members could review it and satisfy themselves that it would give theoretically accurate results.

Since QuickSim was only a 3DOF simulator, the torques and moments of inertia of the launch vehicle are not taken into account. A simplified assumption is used to predict the pitch angle of the vehicle for thrust vector alignment. This assumption is that the fins of the vehicle keep the rocket pointed into the relative wind so that the thrust vector is always in line with the velocity vector. This is, of course, a large assumption, but I could not think of a more accurate solution without going to a full 6DOF simulation.

Verification of QuickSim was performed by comparing its results to those of various freely available trajectory simulation programs such as Ascent4.0 (freeware) and a temporary license of Ascent Professional. While the trajectories were not identical, the altitude results were generally within a factor of 2 for similar inputs. Unfortunately, the source code of the Ascent programs was not available, so I was unable to quantify
differences in atmospheric models which could have significant impact on the expected trajectory.

Early analysis conducted with QuickSim suggested that with a liftoff thrust-to-weight ratio of 2 (which was earlier determined to be optimal for minimizing overall lost velocity due to gravity and drag at a vertical 90 degree launch elevation angle) launching at an elevation angle of 80 degrees would result in the crashing of the vehicle only a few hundred meters downrange. This simulated phenomenon can be explained by low speed off of the launch rail resulting in excessive “pitch-over” of the launch vehicle at very low altitude. Since our budget is very limited, we could not afford the cost of developing an engine with substantially higher thrust, and since the 200 lbf thrust engine was already underway, it was decided that we would use commercial off-the-shelf (COTS), high-power model rocket motors such as those manufactured by Aerotech Inc[17].

QuickSim was then modified to incorporate a routine which would model the vehicle with a solid booster stage. The boosters would be ignited as soon as the main CDIE engine had been confirmed to be operating normally. After the boosters lit, the rocket would be released. QuickSim showed that because the boosters allowed the rocket to achieve a much higher velocity as it left the launch rail, there would be much less pitch-over and the vehicle could be made to achieve both of the CATS prize goals. By conducting the first launch at an angle of 80 degrees – the maximum angle allowed for an unproven vehicle at NASA’s Wallops Flight Facility, the predicted trajectory with 3 M1315W boosters would apogee over 120km. This would be sufficient to claim the $50K second prize. If that vehicle performed well, the identical flight backup vehicle
Figure 8: Flight vehicle trajectory at 80 degree launch angle

could then be launched at a higher angle. QuickSim shows that if the same vehicle was launched at an angle of 82 degrees, the apogee would soar well above 200km — sufficient to capture the $250K first prize. Obviously, these trajectories are very sensitive to many variables, especially engine performance. So further simulation would be necessary based on actual engine data before a final vehicle configuration could be decided upon.
But the QuickSim routine had given us hope that accomplishing the CATS Prize goals might be possible for our team on our budget.

Figure 9: Flight vehicle trajectory at 82 degree launch angle.

As the project progressed, the design of the vehicle employed a thin carbonfiber tube as the primary skin and support structure. I was concerned that as the vehicle accelerated through the upper atmosphere, the heat flux due to air friction would cause the epoxy matrix to melt and the rocket to disintegrate. I decided that a numerical
evaluation of the effects of this high speed flight on the structure was necessary, so I modified the QuickSim code to employ a routine for calculating the heat flux to the wall and the wall temperature as the rocket ascends into space. The conclusion of this analysis was that some type of shielding would be necessary due to the desired high burnout velocity of the rocket (approximately Mach 6). Without some type of thermal protection, the thin composite wall would rapidly disintegrate at the desired speeds.

The program was then modified to evaluate a cork skin, ablative thermal protection scheme. It was concluded based on this analysis that a .25” cork skin bonded to a thin composite tube structure that was formed with a very high temperature PEEK epoxy could theoretically endure the high speed flight based on the cork ablation analysis used for the Minuteman IV missile as long as the char temperature of the type of cork used was below the weakening temperature of the epoxy.

Since it is unlikely we will have the ability to test such a thermal protection scheme before the flight of the vehicle, I have recommended against the usage of this type of structure in favor of using the metal wall of the fuel tanks as the outer skin and structure. The results of the QuickSim code were an integral part of this recommendation.
3.2 "CDIE_sim" A Dynamic Model of the CDIE

CDIE_sim was developed to numerically investigate the stability of the centrifugal direct injection engine while quantitatively taking into account the non-idealities of the system that were not present in the crude analytical stability solution presented in the previous chapter.

The code is a dynamic simulation of the first five seconds of engine operation that recalculates all state parameters (massflows, pressures, temperatures, heat fluxes, bearing loss, windage etc.) every 100 microseconds. Because of the large chamber volume, this timestep tends to result in a practical maximum change in chamber pressure per time step that is of order 1 atmosphere. This maximum change is assumed to be a small enough percentage of the design operating pressure (~ 35 atm) that only real transients will be taken into account. Since all average molecular residence time values are larger than the timestep, this is another good indicator that the code can be trusted to converge to a real, dynamic, simulated solution.

The CDIEsim code employs the heat transfer model into rotating disk systems described by Owen and Rogers and reproduced in chapter 2 of this document. It is not known if this model will provide sufficient accuracy to predict the operating point of the engine because the Owen and Rogers model was developed for heat transfer into turbine disks in gas turbine engines. These disks are generally heated by conduction from the turbine blades and are kept cool by the bypass air blowing over their face. In this situation the boundary layer is hot and consequently less dense than the bulk air. The less dense air in the boundary layer would experience a mix of centrifugal forces and buoyancy forces which act in opposite directions. These opposing forces could cause local
instabilities in the flow similar to Rayleigh-Taylor type instabilities. These local instabilities could enhance turbulent transport and therefore increase the local heat flux. Since the Owen and Rogers model is semi-empirical, it would already account for these effects.

In the CDIE engine the boundary layer is cold because the combustion chamber gas is hot relative to the cold, rotating propellant injection manifold. Consequently, the buoyant force and the centrifugal force on the boundary layer are operating in the same direction. This scenario could decrease the turbulent transport relative to the above scenario. It is possible, therefore, that the Owen and Rogers model will overestimate the local heat transfer. Such a decrease in the heat flux could result in a lower stable operating point for a given design, or in the worst case, a complete failure of the engine. Conversely, Professor Martinez-Sanchez has suggested that the cold boundary layer could increase the heat flux because the extra radial force would create a faster radial flow near the surface, shearing the fluid more and thinning the boundary layer. This would increase the heat flux relative to the predictions. The large uncertainty even in the theory of how the heat flux would be affected resulted in the need for a conservative experimental approach. Since this type of cold centrifugating boundary layer has not yet been experimentally investigated, it did not seem logical to conclude that the engine would not work based on the above hypothetical, qualitative argument.

Practically speaking, it seemed far more likely that a failure of the engine would be due to radical local variations in the mixture ratio from poor mixing of the kerosene and oxygen. This variation would cause local pockets of hot and cold gas that would decrease the overall heat transferred into the disk. Since all of these heat transfer effects
were beyond the scope of the CDIEsim engine model, it was decided that the Owens and Rogers model would be used and the effective area of the disk would be adjusted to match experimental data.

As previously mentioned, CDIEsim contains improvements on the theoretical model presented in Chapter 3. These improvements include estimations of windage and bearing losses based on experimental data taken during atmospheric spin tests of engines E1 and E2, more accurate calculations of pressure drops in the cooling passages inside the disk based on the pressure drop model described in Sutton [1]. Accurate moment of inertia data based on the calculated solid model moments in ProEngineer was also included. The calculations were cross-checked with order-of-magnitude measurements. CDIEsim also incorporated an off-axis oxygen jet angle which was measured in experiment to be approximately 15 degrees.

Initially, CDIEsim was created just to evaluate whether a stable operating point existed for the CDIE engine. As explained in chapter 3, the engine is marginally unstable in an ideal implementation (when windage and bearing loss effects are not accounted for), so it was not certain if the engine would work at all even when the real world effects were taken into account.

After a stable operating point seemed possible, detonation tests with simulated explosives were modeled to see if the stable operating point was only a local phenomenon. The results of that series of simulated tests were interesting because they showed that the engine had so much relative inertia in the rotating manifold that short term radical fluctuations in chamber pressure such as those that would be consistent with the detonation of a buildup of propellant in the chamber did not cause any type of
noticeable change in the rate of speed of the disk. The large inertia of the disk relative to the small massflow and power levels cause rapid chamber pressure fluctuations to be effectively damped by the low-pass-filter-like interaction of the rotating manifold with the propellants. This seemed to be very good news in that the engine could be very practically stable (even though it was not theoretically stable) – if we could get it to work.

Due to the large number of uncertainties that would contribute to the driving heat flux, the exact operating point of the engine could not be predicted with any confidence, so the uncertainty was experimentally compensated by the use of a high power electric motor/generator as described in Andrew's thesis. The following plots contain some sample outputs of CDIEsim. They plot the thrust, specific impulse, oxygen and kerosene massflows, the oxygen jet force, the chamber pressure, the net torque on the rotating manifold, the heat flux into the disk, the angular velocity, and the chamber temperature during the first five seconds of operation from two different start speeds. These figures display the tendency of the engine to have a single stable operating point. The two runs were conducted from simulated starting speeds of 3400 rad/s and 4800 rad/s respectively. Both scenarios result in a steady operating speed of about 3700 rad/s with a thrust of roughly 800N (about 180 lbf.). Due to the uncertainty in the heat flux because of the postulated phenomena described above, this theoretical design was the one used for engines E2 and E3. This model was not yet fully operational when E1 was constructed. It was hoped that even with a somewhat decreased heat flux, the engine would still find a lower stable operating speed with a corresponding lower chamber pressure and thrust.
Figure 10: CDIE transient startup simulation from start speed of 3400 rad/s

Figure 11: CDIE transient startup simulation from start speed of 4800 rad/s
As can be seen in the previous figures, the transient time constant of the engine was expected to be approximately 2 seconds. This time constant was convenient because it was long enough to allow the potential for a software controlled shutdown of the engine if various programmable safeties were triggered. The time constant was also short enough to allow for a small quantity of fuel at the test site. It was decided that a 5 second burn would be sufficient test duration based on transient, 1-D, order-of-magnitude analyses that were confirmed with CDIEsim.

In addition to predicting performance metrics of the engine, the CDIEsim routine was also used to monitor the expected structural safety margin in the highest stress region of the engine, the outer wall of the rotating propellant injection manifold. Because of the high pressure inside the rotating manifold (2000 to 3000 psi for the expected operating speed) and the relatively low chamber pressure (500 to 600 psi), the manifold had to be designed as a pressure vessel. Consequently, a Von Mises yield analysis was done on the thinnest part of the outer rotating wall that took into account the hoop stress and the axial stress due to the internal pressure and the material stress caused by the high speed rotation of the disk. This analysis was done during the design cycle on paper. CDIEsim was modified to incorporate an on-the-fly Von Mises analysis routine that would calculate the yield margin of safety for that point in the disk. The following figure displays the yield margin of safety corresponding to a start from 3400 rad/s.

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8 Thermal stresses were not incorporated into this model because they were determined to be beyond the scope of this simple analysis.
9 Unfortunately, this sidewall was not the highest point of stress in the manifold as we discovered during the analysis of the destruction of E2. Finite element analysis by Tom Nugent later determined that there was a stress concentration because of the sharp interior corner. This design flaw was a serious oversight on my part that was present in the first two versions of the engine.
The CDIEsim analysis routine was completed during the design phase of engine E2. It was used to analyze the gas and liquid film coefficients, investigate structural safety margins, verify and evaluate a stable operating point, and simulate the test firing of engines E2 and E3. Although we were not able to obtain real data to verify the routine, all first-order cross-checks of the program suggest that it is outputting reasonable numbers.

Figure 12: Transient Von Mises yield analysis of the outer wall of the manifold.
Chapter 4: The E1, E2, and E3 Test Engines

Since the founding of the MIT Rocket Team in December 1998 three liquid bipropellant rocket engines have been constructed and tested to varying degrees. The engines are referred to as E1, E2, and E3. While each successive engine included a number of design changes, the overall configuration, size, and design thrust of each remained constant. All engines were designed to run on kerosene and liquid oxygen (LOX) and all were designed to produce roughly 200 pounds of thrust at a chamber pressure of 35 atmospheres.

Ironically, the first engine (E1) produced the most visible signs of success in a hot-fire test before a later test caused its catastrophic destruction. Engine E2 failed in the final LOX pumping test before its first hot-fire test, and E3 has suffered from high-speed turbo-machinery problems that delayed the schedule to the point where testing had to be cancelled. Although from this description it may sound like negative progress is being made, such a conclusion would not be accurate since this project has been an incredible hands-on-engineering learning experience for all those involved. This experience has inspired us to dedicate ourselves to the design of a fourth engine (E4) which will be a substantially different from the first three while still embodying the lessons learned during the development of E1, E2, and E3.

The following sections describe the development and testing of engines E1, E2, and E3. Descriptions of the design changes in each engine are included along with some of the actual experimental data. Finally, a summary of some of the important lessons that we have learned from these engines concludes this section.
4.1 The Development of E1

The design of engine E1 began in the fall of 1999. The marginal success of the proof-of-concept test rig had led us to the conclusion that the theory of the engine operation was sound thermodynamics that could potentially be turned into a real rocket engine. The general design features of this engine were outlined in section 1.4. The detailed design of the engine including parts drawings was completed in late December of 1999.

Figure 13: Cross-section of E1 with parts description

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Figure detailed by Andrew Heafitz, reproduced here with permission of the artist.
The previous figure contains only the turbopump assembly which constitutes the upper half of the engine. The engine is assembled by placing the turbopump assembly into the top of the combustion chamber/nozzle. The chamber/nozzle is made from a large block of copper for the test rig, but the internal contours are similar to those of the silica-phenolic chamber/nozzle design that was proposed for the flight weight vehicle. The full engine cross-section is displayed in figure 12, but since all of the most of the design time goes into the turbopump assembly, generally the names E1, E2, and E3 refer to the turbopump assembly since the chamber/nozzle design stayed constant throughout all three iterations.

Figure 14: Proposed flight vehicle engine cross-section with E1 assembly and silica-phenolic chamber/nozzle with carbon fiber outer wrap

The copper chamber/nozzle employs a straight taper expansion section which is different from the bell nozzle depicted in figure 12. The convergent section is identical until very near the throat region.
The fabrication of the parts of E1 was outsourced to Mackenzie Machine and Design for rapid fabrication. By late March of 2000 the engine was built and ready to be assembled.

![Image of E1 parts before testing]

**Figure 15: E1 parts before testing**

At that time we only felt comfortable doing high speed spin tests at the Rocket Team engine test facility in Thornton, NH. Because the test facility is located two hours north of Boston and we had only been able to conduct low speed tests in our labs at MIT, it took 5 months of spin tests and 3 unsuccessful trips to NH to work out the bugs of the high speed turbo-machinery in E1.

Our fourth attempt to test fire the engine was in September of 2000. At that point we had just about accepted the reality that there was no way that we were going to be able to launch before the CATS Prize deadline of November 8, 2000. It is safe to say that the entire team was sick of making fruitless weekend trips to NH only to have to go home without firing the engine. Fortunately, the engine seemed to be spinning well that day, and all of the necessary pre-hot-fire tests had been completed without doing excessive damage to the engine. It was time to fire.
At that time, the startup sequence was a major point of debate with too many people arguing with hand-wavy explanations based on little actual analysis. But the entire project up until that point had been run as a giant last minute effort because we thought we had a hard deadline of November 8, so arguing at the last minute over important technical details that should have been decided long before the test was not at all unusual. Fortunately, despite our tremendously worn out efforts, we were lucky that day. The engine actually turned on and safely turned off a half second later in a semi-controlled manner. During which time only minor damage was caused to the engine. The team was ecstatic.

Figure 16: Ignition of E1 during test 1041 in September 2000
Although the test was cut short by a programming error, the engine had successfully ignited and 120 lbf of thrust was measured. This thrust level was only 60% of the design thrust, but the burn was so short (less than ½ second) that the engine did not have sufficient time to achieve a steady state operation, so heat flux data could not be retrieved from the test. A detailed breakdown of the 1041 test firing is contained in the Master’s thesis of Andrew Heafitz.

![Figure 17: Chamber Pressure and Tachometer plots, test 1041](image)

E1 was tested two more times at the New Hampshire test facility. Neither test was as successful as 1041. During test 1098 in November, a crack located at the base of the LOX feed tube, which was initiated by operator error in July of 2000, grew to the point of failure. The LOX mixed with kerosene while still inside the propellant injection manifold. This internal mixing and ignition caused the rapid destruction of E1.
While the development of E1 was certainly a very rushed process, many valuable lessons were learned along the way. It became clear that future development could be greatly accelerated by enabling the high-speed spin testing at our facilities in MIT. In addition, the crack that was initiated at the base of the LOX tube had to be prevented in future design iterations by strengthening the joint and removing welded joints wherever possible. Finally, after the E1 hardware had been completed and test 1041 was conducted, review of the heat transfer analysis revealed a mistake in the calculation of the liquid film coefficient. Evidence of the error could be seen after the $\frac{1}{2}$ second test firing in test 1041: the outer edge of the disk showed signs of local melting. This design mistake slipped through the design review because of the rapid pace of the E1 analysis and design. We did not have the manpower or time to thoroughly review most of the E1 analysis. The next engine, E2, would incorporate design process changes in addition to design changes which came about as a result of our experiences with E1.
4.2 E2

By the time E1 destroyed itself, the CATS Prize competition had expired. This allowed time for a detailed review of the analysis of E1. Aside from making the obvious change to the design of the base of the LOX tube to prevent a similar failure in the future, it was clear that insufficient analysis had gone into the design of the E1 heat exchange passages because of the time pressure to get something made for a CATS Prize launch. Since that time pressure no longer existed, I conducted a more detailed model of the heat transfer through the rotating manifold (often referred to as the disk). This analysis confirmed that the cooling passages in E1 were substantially oversized and that if we had

Figure 19: The remnants of E1 after the destruction during test 1098
fired for a more extended period of time than the 1041 burn, the disk would likely have melted. In a way, therefore, it was fortunate that the typo in the computer sequence caused the premature shutdown of E1 in test 1041.

The new cooling passage design called for channels that were approximately 0.14 inches in height and width compared to the 0.3 inch dimensions of the E1 passages. This change of configuration lead to a much more complicated bottom disk design. The smaller channel size of E2 necessitated multiple channels in the inside “side” wall of the bottom disk piece. The E1 design had no such channels. These smaller channels made the bottom disk substantially harder to machine and consequently increased the cost of that piece significantly over the E1 counterpart since the interior channels had to be cut with a saw cutter on a CNC mill and it was much more likely that a tool would break during the machining of the bottom channels.

Figure 20: E1 bottom disk (left) and E2 bottom disk (right)

Aside from the cooling passage changes, E2 remained substantially similar to E1 in size and construction.
Figure 21: E2 cross-section; Smaller cooling channels in the bottom disk
4.2.1 Summary of Design Changes

Despite the superficial similarity of E1 and E2, the design, development, manufacturing, and testing of E2 was much more methodical than the E1 effort. Since we had resolved many of the issues associated with the operation of high speed turbomachinery during the testing of E1, we took those lessons and applied them to E2. Since we wanted the same lessons to apply, we only changed what we thought that we absolutely had to change. The mantra "don't make it better unless it broke last time (or will theoretically fail this time)" was what guided the design of E2. The following changes were made to the design:

- Cooling passage geometry was modified to incorporate smaller channels and side-wall channels which would provide the proper liquid film coefficient.

- The welded, stainless steel vacuum insulated assembly that made up the LOX tube was replaced with an aluminum inner LOX tube piece and a stainless steel outer LOX tube (with a thicker, machined flange) so as to avoid a destruction similar to the 1098 test.

- Air gaps with O-rings were used instead of the e-beam welded vacuum for insulating the kerosene from the LOX tube and pump.

- The outside of the bottom disk was rounded to better cool the outer corner which had melted during test 1041.
• The upper manifold piece which contains the gas separator seal was constructed out of bronze instead of aluminum because of bronze's superior wear qualities (the aluminum sealing surfaces in E1 gauled significantly during spin testing).

• The small gear that was attached to the outside of the kerosene disk shaft between the two bearings which was used to drive the engine with the electric motor was fixed in place mechanically with spacer rings and a steel key instead of the LockTite glue which failed in the previous engine because of thermal cycling.

• A high-speed catastrophic failure containment device consisting of a .25" thick, kevlar reinforced steel shroud was used so that high-speed spin testing could be done in lab.

• Backwards-facing oxygen injector nozzles were fabricated so the jet power, the pumping effectiveness transient, and pumping power could be isolated from the power required to spin the disk during in-lab liquid nitrogen pumping tests\textsuperscript{12}.

\textsuperscript{12}This method of isolating the jet power and massflow was first suggested by Professor Martinez-Sanchez.
Figure 22: E2 assembly with gear interface and bronze top

The extra time spent on in-lab tests and the design of those few changes in E2 lead to an engine that required very few adjustments when the parts returned from the machine shop. Unfortunately, there was a significant design flaw that made it through our reviews despite the extra time that was spent on the changes which will be discussed later in this section.

4.2.2 E2 Testing

The E2 testing program was designed to systematically attack the problems we had had with the E1 engine and to test a theory that our previous startup sequence was
not correct. Since confirmation of this theory would have a significant impact on the startup sequence of E2, the test plan was designed around the necessity to have both backwards and forwards LOX nozzle tests.

The first step was just to spin the engine at high speed. In order to accomplish that goal, we did not require a high-pressure seal on the shaft. We had encountered significant difficulty with the chamber separator seal in E1 and since we knew that we would have to disassemble the engine again later to put in the forwards LOX nozzles, we decided to insert the chamber separator seal at that later time. Therefore, when the parts of E2 were received from the machine shop, the engine was first assembled with the backwards LOX nozzles and no chamber separator seal. The order of operations was supposed to be as follows:

1. Low speed spin test: the purpose of this test is to listen for rubbing and to see if it is safe to proceed to higher speed tests.

2. High speed spin test with coastdown: the goal here is twofold – to work out high speed turbomachinery problems (basically to see if it will spin fast without rubbing or excessive wear on any surfaces), and if the engine does not make any bad noises, we allow it to coast down to a stop. The data from this spindown curve is used to compute the power required to spin the engine at steady-state at any given speed. This computation is done by looking at the instantaneous change in speed and computing a power required to perform that change. These computed curves are then compared to low speed tests (which are assumed to have only bearing losses). Since the bearing loss torques are assumed to be proportional to the speed of rotation and the windage losses are assumed to be proportional to the square of the speed, we can extrapolate
these losses to estimate the losses with more highly loaded bearings and more windage loss due to increased combustion chamber gas density.

3. Liquid Nitrogen pumping test with backwards nozzles: by running liquid nitrogen through the LOX system with the LOX nozzles pointed “backwards” so as to provide a retarding torque on the disk, it is possible to clearly measure the effect of the jet force by looking at the power required to spin the engine. When the nozzles point forwards, the jet power compensates for the increased power that is required to pump the LN2 or LOX and effectively cancels any sign of increased power required (until the disk floods); it is therefore impossible to see the first signs of “good pumping” by looking at the power data. It is only by pointing the LOX injector nozzles backwards that we can clearly see the effectiveness of the pump and the jets in the data.

4. Water pumping test through the kerosene system: the purpose of this test is to measure discharge coefficients of the kerosene injector holes and to look at the effectiveness of the rotating seals at different feed pressures. The flow rates can also be cross-checked against the power data to confirm that both measurements are reasonable. Quantities of water that flow through the dump lines are collected and measured so that an accurate prediction of kerosene leakage can be made.

5. Water pumping test through the LOX system: the purpose of this test is to cross-check power and flow measurements and to see the effectiveness of the gas separator seal at containing liquids. During this test as during the tests with water in the kerosene system, a small scrap of paper with a water soluble ink line was placed in the Helium manifold of the gas separator. If water leaked through and made the ink run, it would
be assumed that the gas separator was not working at the pressures that it had been set to.

6. Disassemble engine, clean and dry all parts, then reassemble engine with forwards LOX nozzles and graphite chamber separator seal. Inspect and remove the ink stained paper from the water flow tests.

7. Break-in graphite seal: because the chamber separator seal is a very high pressure gas seal, there tends to be a large gas leak flow through this seal. In an effort to minimize gas usage, the chamber separator was left out of the earlier tests so that it would have the minimum number of cycles on it before the hot fire test. In order to make the seal as tight as possible, it was machined as a tight slip fit around the grooves of the kerosene disk. Then, when it is assembled, it requires a break-in period in order to make sure that the shaft can spin freely at high speed.

8. Chamber Separator test: after the chamber separator seal has been broken in, the leak flow must be quantified and the pressure in the separator manifold must be measured to ensure that hot chamber gas will not escape through the seal an impinge on the lower ball bearing.

9. Bearing Loss test: the bearing losses must be quantified with the bearings loaded to the full pressures that they will experience during operation. In order to accomplish this goal, the combustion chamber is plugged at the throat and the entire chamber is pressurized to the operating chamber pressure of 515psi. The power required to spin the engine is then measured to quantify the losses\textsuperscript{13}.

\textsuperscript{13}A bearing deflection test was added to this test battery when it was found that the elastic deflection of the lower bearing was sufficient to cause the top of the propellant injection manifold to rub against the chamber ceiling. This test consisted of loading the bearing on an arbor press and measuring the deflection with calipers.
10. LN2 Pumping test with forwards nozzles: this test was conducted in order to have an appropriate comparison for the backwards nozzle test on the same set of hardware.

11. LOX pumping test (at NH test facility): this test is conducted prior to a hot-fire test in order to make sure that the entire cryogenic flow system is operating as designed and to compare to the previous LN2 flow tests to make sure that the time until flooding is about the same so the startup sequence can be finalized.

12. Hot Fire test (at NH test facility): the goal of this test is to successfully start the engine, fire it for 5 seconds while collecting data on the heat transfer into the disk via the motor power measurement, then perform a safe shutdown.

   This theoretical order of operations was supposed to take place during the spring of 2001, culminating in a hot-fire test at the beginning of the summer. Although the order of operations was roughly followed, the timeline extended into the middle of the summer with an attempted hot-fire test trip on the weekend of July 20-22, 2001. We considered this relatively short overrun a success from a planning perspective.

4.2.3 E2 Results

   Analysis of the E2 in lab test data was used to periodically cross-check theoretical predictions of engine performance and to help develop a startup sequence for the engine. Specifically, measurements of liquid nitrogen flow through the LOX system allowed us to modify estimated nozzle/passage discharge coefficients which resulted in the enlargement of the LOX nozzles from 1.9 to 2.2mm diameter. In addition, the backwards nozzle testing confirmed the theoretical prediction that “good pumping” is achieved on a timescale of approximately 0.2-0.5 seconds after the LOX engine valve opens. What we had thought was a sign of good pumping in E1 is now believed to be the flooding of the
entire bottom disk due to total cooling to cryogenic temperature and the injection of liquid cryogen into the combustion chamber\textsuperscript{14}. This phenomenon typically occurs on timescales of 3-5 seconds after the LOX engine valve is opened depending on cryogen and flow rate. This theoretical prediction was confirmed by comparing the power curves for liquid nitrogen pumping with backwards nozzles to those of liquid nitrogen pumping with forwards nozzles.

It is important to determine when "good pumping" occurs in the engine because if combustion is initiated before sufficient pressure is developed inside the rotating propellant injection manifold, the chamber pressure could reverse the flow of the LOX causing the engine to "burp" back into the propellant storage tanks. At best it would be a failed startup, at worst, the entire test stand could be destroyed. On the other extreme, if you wait too long to ignite the engine, the entire chamber could be filled with liquid oxygen which would inhibit mixing and could cause a "hard start" scenario which could also result in the destruction of the test stand. It would also be non-optimal to wait until the bottom disk is flooded before starting the engine because the increased oxidizer massflow which results from liquid injection would cause a very lean mixture in the chamber which may have a significantly lower combustion temperature which would decrease the heat flux to the disk and thereby perpetuate the cycle of liquid oxygen injection. In addition, when liquid oxygen is injected directly into the combustion chamber, it tends to flash-vaporize on the walls of the chamber (since the chamber for the test stand was designed as a big copper heat sink, it can also be a big heat source). This vaporization causes the chamber pressure to spike to pressures of over 100psi\textsuperscript{15}. If the

\textsuperscript{14}Ref. [4] p.45
\textsuperscript{15}Test 1091, [4] p. 46
Kerosene flow has not been initiated prior to this disk flooding, the kerosene tank would have to be pressurized to more than 100psi in order to initiate the flow of kerosene. Clearly, this is not practical, so it is very important to find the period of time after “good pumping” has been initiated and before the disk floods. The kerosene flow must be initiated and ignited in this window for a successful start that would lead to steady-state operation.

The following plots are of a useful calculated variable termed “Anomalous Power.” This power measurement is computed in the following way:

- Current and Voltage from the battery pack that drives the electric motor are measured directly, this electrical power is assumed to be equivalent to the shaft power because of the high efficiency of the motor system near our operating speed. This is the “Motor Shaft Power.”
- The change in speed at any given time is calculated and the corresponding change in rotational energy of the engine system is computed. These small changes in energy are divided by the discrete chunk of time the computer looks at between data points and an “Inertial Power” is determined. This power is stored in the angular momentum of the engine system.
- During the high speed spindown tests, the power that caused the spindown (i.e. Bearing losses and windage losses) was calculated by looking again at the instantaneous change in speed of the engine system. A steady-state power loss can then be associated with every speed in the spindown range. This power is known as the “Steady State Power.”
• When the actual motor shaft power is subtracted from the imperically derived inertial power and steady state power, what is left is termed “anomalous Power.” During normal spin tests, this computed power is always zero (except for the dip during startup which is believed to be caused by motor inefficiencies at low speed). If there are other external events that absorb power from the engine system (or provide power to the engine system), they show up clearly in the anomalous power plots. These events can be fluid pumping, rubbing seal (negative anomalous power), or the oxygen jets (positive anomalous power) or all three. Events that absorb energy from the rotating system (such as a rubbing seal) would show up as negative spikes on the anomalous power plot.

As can be seen in the anomalous power data in test 2051, there is a sharp transition almost immediately after the LOX engine valve is opened (at t = 79 seconds). This 2 kW power drop is accounted for by pumping power and jet power for the measured massflow at that rotational speed. The latter sharp drop at t = 83 seconds is explained by the complete cooling and flooding of the rotating propellant injection manifold. After this time, the steady 3.3 kW power level is explained by the increased massflow of LN2 associated with direct liquid cryogen injection into the combustion chamber.
These previous data can be contrasted to test 2104 which is a liquid nitrogen pumping test with forwards nozzles. As can be seen in the following figure, there is no initial power drop corresponding to the time at which the LOX engine valve is opened, instead there is only a single drop in the anomalous power roughly 6 seconds after the valve is opened corresponding to the flooding and liquid cryogen injection.

Figure 23: Test 2051, Liquid Nitrogen pumping with backwards nozzles
Figure 24: Test 2104, Liquid Nitrogen with forwards nozzles

Notice that there is little change in speed of the engine after the LOX engine valve is opened (in fact, the engine continues to increase in speed), this is explained by the heat capacity of the disk vaporizing the initial LOX flow and acting as jets – thereby compensating for the pumping power. Again the 3.3kW of power is required when the disk has fully flooded.

The initial dips in anomalous power that are seen during spin up are believed to be caused by the inefficiencies of the electric motor system at low speed as they show up
on all plots of anomalous power and the motor efficiency is assumed to be close to 100%\textsuperscript{16}.

During test 2053 there was a loud "screech" that corresponded to the strange dip in anomalous power between 81 and 83 seconds, this anomalous rubbing seemed to clear itself out by 84 seconds and we were able to use the data from the rest of the test despite the odd noise at the beginning. If we had not seen the anomalous power go completely back to zero before the LN2 flow began we would have thrown out the data, but since we could verify that the burr had completely cleared itself, we were able to use the data from the test with confidence. Again, the immediate drop in anomalous power is clearly visible

\textsuperscript{16}See motor discussion on p.40 of “Implementation and Startup of a Centrifugal Direct Injection Rocket Engine” for justification of this assumption.
at t = 87 seconds after the LOX engine valve is opened, then 5 seconds later (at t = 92s), the disk is flooded\(^ {17} \). It is believed that the 6kW spike in anomalous power that is associated with the disk flooding can be explained by the effective instantaneous increase in the moment of inertia of the disk due to the mass of the liquid that fills all of the channels and the settling chamber. This power spike would be classified as an inertial power spike on top of the increased pumping power due to the higher flow rate and the larger pumping radius. That inertial power spike then vanishes as all of the rotating cavities are filled with liquid and all that is left is the 3.3 kW associated with the pumping.

Figure 26: Test 2104, LN2 pumping with forward nozzles into combustion chamber

In addition to the backwards nozzle data, the chamber pressure and thermocouple data support the theory that the disk is completely flooded after 4-6 seconds. This

\(^{17}\)The variance of the time until flooding of the disk is caused by changes in the flow rate due to different LOX tank pressures upstream of the engine. Lower tank pressures cause increased cavitation near the pump entrance and decrease the flow rate which causes an increase in the time until flooding. Many tank pressures were tested before the final pressure of 90 psi was decided.
evidence can be seen in figure 26 by observing that the combustion chamber thermocouples do not drop in temperature until after $t = 22$ seconds; they then begin to drop very quickly indicating that there was a sudden change in heat flux from the copper block combustion chamber.

This sudden change in heat flux is likely due to liquid nitrogen directly impinging the walls of the combustion chamber. The liquid nitrogen would have a much larger film coefficient, and therefore a much more pronounced cooling effect than cold gaseous nitrogen.

![Graph](image)

**Figure 27: Test 2104, LN2 pumping chamber pressure**

While the thermocouple data suggests that liquid is impinging the walls by $t = 22$ seconds, the chamber pressure data in figure 27 corresponds more closely with the anomalous power data and suggests that there is liquid injection occurring at $t = 19$ seconds. This can be seen by the sharp increase in chamber pressure which could be associated with flash vaporization.
While much of this evidence is circumstantial, when put together with the backwards nozzle data, the jet power theory, and the first order heat capacity calculations, the argument is fairly convincing that good pumping is achieved almost instantaneously after the LOX engine valve is opened. There then appears to be a 4-6 second window during which the engine could theoretically be started before the rotating propellant injection manifold chills to the liquid cryogen temperature and floods.
4.2.3 E2 Failure

During test 2108, a design flaw in E2 caused the catastrophic failure of the engine. Test 2108 was supposed to be the last test before the first E2 hot fire test. The goal of the test was to pump liquid oxygen (LOX) for the first time in order to verify that LOX pumping could be achieved and to compare the time until flooding of the bottom disk with the time until flooding during the liquid nitrogen pumping tests, so as to confirm the estimated 4 second delay.

Since LOX has a higher density than LN2, the pressure inside the rotating manifold during a LOX pumping test is higher than the manifold pressure during a LN2 pumping test at the same rotational speed. This increased fluid density and pressure should have resulted in a higher massflow and consequently, somewhat faster cooling of the manifold. It was necessary to confirm this prediction and measure the ignition window prior to the hot-fire test. It was also necessary to prove that LOX could be safely pumped through the engine without a combustion event occurring inside the pump.

Unfortunately, the inside of the bottom disk had not been designed as a pressure vessel. While much analysis had gone into the design of the heat exchange passages, I forgot that the bottom disk piece needed to have rounded interior corners in order to prevent stress concentrations at sharp interfaces. This fundamental oversight on my part led to the destruction of E2 during test 2108.

The high pressures inside the rotating manifold that were caused by pumping LOX during test 2108 initiated a fast fracture in the corner of the bottom disk near the stress concentration. The bottom of the manifold blew off and seemed to partially obstruct the throat of the combustion chamber. This partial obstruction caused a rapid
spike in the chamber pressure since the LOX was not able to escape as it was vaporized by impacting the chamber wall.

![Graph showing chamber pressure and tachometer data](image)

**Figure 28: Test 2108 E2 LOX pumping test failure**

Post mortem finite element analysis conducted by Tom Nugent confirmed that the local stress in the corner of the bottom disk exceeded the yield stress of 7075 T6 Aluminum[5]. The red-to-gray boarder in the following figure corresponds to a maximum principal stress of 500MPa. The yield strength of 7075-T6 Aluminum is 505MPa. The gray area in the outer, lower corner of the disk is figure 29 corresponds to stresses that exceed this limit.
Interestingly, the finite element analysis suggests that the stress concentration should have produced yielding even in LN2 pumping tests. It is possible that small cracks were initiated during the plastic deformation of that corner during previous LN2 pumping tests. Then the increase internal pressure associated with the LOX pumping test caused even higher stresses which initiated a fast fracture at the lower corner.
Although E2 was never hot-fire tested, the methodical nature of the testing program was successful in working out the bugs of the engine prior to a hot-fire test. E2 also served to confirm the theoretical startup predictions which were different from those concluded upon during the E1 test battery. To borrow a NASA phrase: E2 was a “successful failure.”

4.3 E3

Following the failure of E3 during its first LOX pumping test, it became immediately apparent that what I had thought to be the point of highest stress in the engine was not at all the point of highest stress. Consequently, the Von Mises analysis that I had integrated into my design code did not predict an accurate structural margin of safety. It was clear that more sophisticated tools were necessary to verify that a redesigned engine (E3) would survive the elevated internal manifold pressures associated with LOX pumping. Fortunately, Rocket Team member Tom Nugent had some experience in conducting finite element analyses. With the help of his friends in the Laboratory for Experimental and Computational Micromechanics, he was able to verify
that the design changes that I proposed making to the bottom disk for E3 would result in safe stress levels in the manifold\textsuperscript{18}.

Figure 31: E3 Finite Element Analysis

Although a stress concentration is still present in the lower inside corner, it moderately exceeds the yield stress of the aluminum only when the thermal gradient associated with the design heat transfer is imposed on the model, this high stress level does not exceed yield during LN2 pumping tests or during LOX pumping tests – only actual hotfire tests. Also, since the corner is now rounded there is a smaller chance for coplanar crack initiation. The E3 disk should therefore survive a near unlimited number of pump tests and a limited number of hotfire tests before failure.

\textsuperscript{18}Many thanks to Nuwong Chollacoop, Andrew Gouldstone, the Laboratory for Experimental and Computational Micromechanics (LEXCOM), and Tom Nugent for running the finite element analyses[5].
Because the bottom of E3 is much more rounded than the E2 bottom, it was necessary to have a multi-tiered “inverted wedding cake” looking piece to cap off the tops of the heat exchange passages in the bottom disk. This piece, known as the “bottom-middle-disk” became a much more massive component of the engine than it had been in E2. It is currently hypothesized that the increased mass of the rotating manifold coupled with its lowered, more cantilevered, center of gravity caused greater imbalance than was achieved in E2 and also a lower first mode natural frequency. What is known is that E3 was never able to achieve full speed rotation. There were many attempts to increase the upper gas separator clearances to prevent the rubbing events that occurred at speeds lower than the operating speed. These rubbing events prevented the engine from spinning up to its full design speed of 36,000 RPM. As a last ditch attempt to get the engine to spin at full speed, the bottom-middle-disk was completely removed from the E3 assembly in an attempt to mitigate the theorized problems associated with the high speed turbomachinery. The engine still experienced rubbing that prevented the achievement of full speed rotation.

Because of the unknown nature of the problem, the difficulty that we had had in its diagnosis, and the fact that critical rocket team members (including myself) had scheduling issues that would at the very least delay continued testing for 6 months, the E3 design was abandoned while we stepped back and reevaluated the overall goal of the team. The following section deals with the conclusion of that reevaluation and discusses the future of the MIT Rocket Team.
Chapter 5: Preliminary Design of E4

Testing of engines E1, E2 and E3 illuminated a number of problems with the original design concept. These problems can be generally characterized by simple mechanical design that lead to highly interconnected systems which in turn resulted in experimental difficulties because everything had to be working at the same time in order to get good data. The interconnected nature of the system also meant that it was often difficult to isolate the root cause of a problem. When the cause of a problem could be isolated, the interconnected nature of the design often meant that the solution to one problem would cause another problem. After the failure of E3 during spin testing, these problems became impossible to ignore. The frustration over the lack of progress lead to a serious reevaluation of the design of the engine. The conclusion of this reevaluation was that the next design iteration should be substantially different from the first three. This new engine would encompass the new goals of the MIT Rocket Team after the conclusion of the CATS Prize competition, and it would embody the lessons learned from the first three engines.

This latest engine is called E4, and at the time of this writing, it is still largely a conceptual design. An early conceptual sketch of E4 is presented in the following figure. Of note is the use of a regeneratively cooled chamber and nozzle. LOX is pumped in a more conventional manner in this conception. It is routed in to a typical centrifugal pump that is on a central shaft. The high pressure LOX is then piped down to the base of the nozzle, where it is injected into the cooling passageways. From the exit of the cooling passages, the vaporized LOX is run through at turbine of some type (it is as yet
undecided as to whether the turbine will be of conventional type or a tip-jet type). The
exhaust of this turbine is directly injected into the combustion chamber. The power from
this turbine is used to drive the LOX pump and the fuel pump which is still internal to the
rotating shaft/turbine assembly (as in the previous engines). The potential advantages of
this conceptual design will be examined in this section.

It should also be noted that in addition to the E4 configuration change, a
philosophical change was made in the development sequence for this engine. The goal of
this new philosophy is to bite off smaller, more manageable chunks of the development,
so that the final engine can be assembled from smaller pieces which were easier to test.
**Possible E4 Design:**

- Low pressure LOX
- Low pressure gas separator/seal
- LOX pump
- High-speed, LOX cooled, Vespel bushing
- Stationary, high pressure O-ring seal
- Turbine with internal (CDIE type) fuel pump
- High pressure E4 Improver

**LOX transfer line**

- No Chamber Separator
- Higher TPR Ratio
- No ablation of nozzle
- Good Heat Transfer Models
- Smaller Size (no large disk)
- Same Thrust as E3
- No ball bearings
- Higher pressure cooling jacket
- Similar parts count
- WE CAN MAKE ALL THE PARTS ON CAMPUS!
- LOWER COST (smaller)
- Higher Isp (no film cooling)
- Ethanol instead of Kerosene
- Incremental Testing Possible
- No electric motor startup

Hoping the construction of this engine will involve more of the team than the construction of past engines.

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Figure 32: Preliminary Design Concept of E4

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The Rocket Team accepted the challenge of moving on to E4. With the very crude conceptual design presented on the previous page in mind, baseline goals for the design of E4 were established.

E4 Design Goals

1) Provide sufficient thrust to loft a 1.5 stage sounding rocket carrying a virtual presence camera/transmitter package to an altitude at which a “user” on the ground would have the experience of a view from space including the curvature of the earth (100-200km).

2) The rocket thrust should not exceed the capacity of the available test facilities.

3) The rocket engine should be reusable (More than 2 start/stop cycles).

4) The development should take advantage of existing rocket team infrastructure.

5) The design should allow as much subsystem testing in lab at MIT as possible before a hot-fire test is required.

6) Performance should be maximized within the constraints outlined above and always with safety and reliability in mind.

The goals presented above for the design of the most recent version of the MIT Rocket Team’s engine reflect the change of course that the team has undertaken since the conclusion of the CATS Prize contest. These goals reflect more of the need to continue to inspire people about space. In order to accomplish that goal, the Rocket Team has joined forces with the Boston Museum of Science (MOS) to try to develop a continuously
evolving, interactive rocketry exhibit. The final product of this endeavor will hopefully be a permanent exhibit that tells the story of the MIT Rocket Team.

After the CATS Prize had expired, the team wanted to continue to develop the rocket with the goal of eventually launching a payload into space. Since we were no longer constrained to launch a dummy payload, we decided the most useful thing that we could do with our proposed vehicle would be to launch a camera package into space. This camera package would have a large number of small CCD cameras that point out in all directions from the rocket. The current payload design calls for the use of 6 cameras. The images from these cameras will be broadcast back down to computers on the ground where they will be recorded. The data will then be reconstructed into a full spherical view from inside the vehicle as it flies into space. This reconstructed view would then be viewable with a virtual reality headset so that a user could effectively see the view from inside the rocket as it launches into space. Such a presentation could be made into a very inspirational exhibit at the museum. The desire to develop and launch this type of payload has led to the imposition of engine design goal number 1. Because of payload bandwidth requirements, the on-board transmitter will likely be fairly high power (10-20W). The electronics and batteries required to perform accomplish the goals of the payload exist as COTS components, but the weight exceeds the planned 2kg payload limit. Therefore, the entire vehicle has been resized to allow for an increased payload mass. The largest scaling possible was determined to be 2:1. The limiting factor for this scaling was the capacity of the existing Rocket Team test facilities. A review of the test stand equipment suggested that we could safely test engines up to 400 lbf. thrust at the
Thornton, NH test facility. The maximum payload mass and E4 design engine thrust were set by this limitation.

Based on the design problems that we have encountered in previous engines and the analyses described in the previous chapters, I generated the preliminary design concept for E4 that will continue to be refined in the upcoming months. The detailed design of E4 is to be conducted by members of the rocket team during 2002 and 2003.

The design of E4 incorporates a single major configuration change that enables a large array of improvements over the previous engines (E1-3). One of the biggest problems with the designs of E1-3 is that the rotating propellant injection manifold needed to have a very large area relative to the throat of the engine. This large area was demanded by the need to vaporize the propellant inside the manifold. Instead of relying on heat transfer through the walls of the rotating propellant injection manifold, the E4 design uses a more conventional, regeneratively cooled nozzle and combustion chamber. This single change in configuration allows the following improvements to be easily implemented:

- Smaller size (higher thrust-to-weight ratio) enabled by the removal of the heat exchange passages from the rotating propellant injection manifold. Since a large disk area is not required, the disk and combustion chamber can be made much smaller for a given propellant massflow.
- Lower shaft moment of inertia (faster startup)
- Smaller seal gaps and fewer leak paths: this improvement is enabled by a combination of smaller size for a given massflow and the removal of all high pressure differential rotating seals.
- Elimination of chamber separator N2 gas seal: Since the LOX is now the fluid that is “sealing” the combustion chamber, we do not require a large supply of high pressure nitrogen gas to prevent chamber gasses from impinging our bearings; the bearings are designed to run in liquid oxygen, and any leakage into the chamber should not substantially change the specific impulse of the engine.

- No ablative throat erosion problems: One of the early concerns with E1 was the rate at which ablative material would be removed from the throat of the engine. The NASA Fastrac engineers that we consulted with early in the project said that they did not experience any erosion in the throat of their engine, but they had a 10% fuel film cooling layer that we could not guarantee, and they only measured their throat diameter to the nearest 0.01 inches. A 0.01 inch change in the diameter of our throat would result in a 5% change in our throat area. A regeneratively cooled copper throat should have less variation.

- No fuel-film cooling layer required (higher Isp possible): Because we are now relying on heat transfer through the chamber wall, we do not have to try to guarantee a film cooling layer on the inside chamber wall.

- More instrumentation of engine possible: this characteristic is due to the separable nature of the components of the design. Since the LOX pump is separate from the heat exchanger and the injectors, we can monitor temperature and pressure changes through individual components and thereby separate our problems.

- Better (existing) heat transfer models: This advantage is obvious; it allows us to use a much more extensive database of theory to predict the performance more accurately than we could have in E1-3.
• More easily predictable stable operating point: since the speed of rotating of the injection manifold should not directly effect the heat transfer into the LOX (except through changes in chamber pressure), the engine is theoretically stable since a factor of $\omega$ has been taken out of the heatflux equation.

• All parts could be machined in house: the smaller size should enable the fabrication of many of the parts in the shops on campus, thereby reducing the cost to the rocket team and allowing more team members to get “hands-on” rocket engine construction experience.

In addition to the above advantages of E4 which are based on the configuration change from the previous designs, there are a number of other changes that will be made to E4:

• E4 will use ethanol instead of kerosene as a “test” fuel. Although ethanol has a lower energy content than kerosene and will consequently produce 10-20 seconds lower specific impulse, it is a much cleaner fuel to work with (in fact we use it to clean the engine parts prior to assembly). In addition, ethanol can be easily acquired in 99.9% pure qualities that are very clean. Since we are making the engine much smaller, the fuel injectors need to be as small as practical to achieve good mixing in the combustion chamber. Any dirt or grime in the fuel could clog those injectors, so the purity of the ethanol is very attractive. Ethanol also has a much lower viscosity than the grades of kerosene that were available to us, this should result in decreased pressure losses in the small passageways. It also does not smell as bad as kerosene.
• A separate, spark activated ignitor is being developed that runs off of the LOX and ethanol tanks. It is hoped that this will result in increased cleanliness, reliability and flexibility over the use of small ESTES rocket motors as pyrotechnic ignitors.

• Finally, the dump gas mixtures from the gas separator seal will be piped down to the over-expanded portion of the nozzle of the engine. This should improve the performance of the gas separator by reducing the static pressure of the dump area, and it should also slightly increase the thrust of the engine.

One of the critical issues that was encountered during the testing of E1-3 was that because the design connected the turbine, pump, and injectors together in the same rotating unit, it was often impossible to separately test the subsystems. For example, in order to test the power required of the LOX pump, it was imperative that we take into account the force of the jets that would be produced during in-lab testing. Similarly, the propulsive efficiency of the tip jets was hard to measure because we had to flow gas through all of the heat exchanger passages to drive the “turbine.” It has become clear that designing an engine with separable components for subsystem testing will drastically simplify the overall engine development. As a consequence of this realization, E4 has been designed so that incremental testing of the engine subsystems is possible.

The following subsystems have been isolated for individual development, and a rough step-by-step development path has been outlined for each subsystem.

• Turbopump system:

  1. bearing and turbopump theoretical design (& necessary component tests)
  2. LN2 pump rig (driven electrically to pressurize LOX for chamber tests)
  3. Turbine test rig (inductively braked to characterize efficiency)
4. E4 turbopump (actual engine turbopump)

- Chamber/Nozzle system:
  1. Bartz heat transfer model (preliminary design)
  2. Pressure-fed chamber (stationary injectors to simulate rotating injection)
  3. E4 chamber (actual engine chamber/nozzle)

- Ignition system:
  1. Dynamic startup model (computational model similar to CDIEsim)
  2. Prototype ignitor assembly (tested on campus)
  3. Engine mockup ignitor (used on pressure fed chamber)
  4. E4 ignitor (actual engine ignitor)

The separation of the subsystems will also aid in the work breakdown for the team. In the past, we have had problems with keeping idle hands busy while Andrew and I work on the engine. Since we were the only two people on the team intimately familiar with all parts of the engine, we had to do all of the work on the engine systems. By taking a more distributed approach to the development of E4, we hope that many other people on the Rocket Team will become intimately familiar with the E4 engine systems. In this ideal context, Andrew and I act more as design advisors to the various people on the team who are responsible for the detailed design of a subsystem.

Based on experience with CDIE engines 1-3, it seems reasonable that we could be hot-fire testing a future version of E4 in its integrated package sometime in the summer of 2003. This engine design could be very close to a flight weight configuration, so detailed launch vehicle design and construction could begin as early as fall 2003 with a target launch date of summer 2004.
It continues to be my hope that the final engine design that is developed over the course of this project could have other applications that could help to decrease the overall cost of space access.
References


6. Dietrich, Carl C., United States Patent # 6,272,847


15. Space Frontier Foundation Web Site: http://www.space-frontier.org/

16. CATS Prize Web Site: http://www.space-frontier.org/Events/CATSPRIZE_1/

17. Aerotech Rocketry Web Site: http://www.aerotech-rocketry.com/
Appendix A: 3DOF Suborbital Trajectory Simulator “QuickSim”

% 3DOF MIT Rocket Team Vehicle Trajectory Simulator
% Simple Euler Integration Routine
% Carl Dietrich
% 2/9/00

clear;
clf;

pi = 3.1416;
g = 9.80665;

% temperature data
fid3=fopen('temperature.txt','r');
[tmp]=fscanf(fid3,'%lg %lg');
status=fclose(fid3);
sz3=size(tmp);
ind=1;
for i=1:2:sz3(1)
    tmper(ind,1)=tmp(i);
    tmper(ind,2)=tmp(i+1);
    ind=ind+1;
end

Launch_Rail_Length = 10;
Launch_Angle = 85*pi/180;

%Main Vehicle Specs:
Mo=40; %92.26;
Mf=8; %17.87;
Isp=255; %245
Thrust=800; %2500;
Diameter = 5.6225*0.0254; %8.02*0.0254; %5.6225*.0254;
S=pi*Diameter^2/4;
mdot=Thrust/(Isp*g);
Burn_Time = (Mo-Mf)/mdot
length = 3.5;%4.11;%m
surface_area = length*pi*Diameter;

%Booster Specs: AeroTech M1939
%Number_of_Boosters = 3;
%Mbo = Number_of_Boosters*(20.063/2.2)+2;
%Mbf = Mbo - Number_of_Boosters*(12.625/2.2);
%Ispb = 187;
%Boost_Thrust = Number_of_Boosters*(380*4.448);
%Diameterb = .098;
%Sb=Number_of_Boosters*pi*Diameterb^2/4;
%mdotb= Number_of_Boosters*(380*4.448)/(Ispb*g);
%Booster_Burn_Time = (Mbo-Mbf)/mdotb
%Booster Specs: AeroTech M1315W
Number_of_Boosters = 3;
Mbo = Number_of_Boosters*(12.6/2.2)+2;
Mbf = Mbo - Number_of_Boosters*(7.7/2.2);
Ispb = 195;
Boost_Thrust = Number_of_Boosters*(283*4.448);
Diameterb = .075;
Sb=Number_of_Boosters*pi*Diameterb^2/4;
mdotb= Number_of_Boosters*(283*4.448)/(Ispb*g);
Booster_Burn_Time = (Mbo-Mbf)/mdotb

%Initialize some variables...
wall_temp = [250 250 250 250 250 250 250 250 250 250];
wall_temp_prev = wall_temp;
nose_wall_temp = wall_temp;
nose_wall_temp_prev = wall_temp;
mach = [.2 .4 .6 .8 1.0 1.2 1.4 1.6 1.8 2.0 2.5 3.0 4.0 5.0 10.0 10.5 25.0];
cdm = [.37 .37 .37 .37 .41 .65 .70 .67 .60 .56 .50 .465 .425 .39 .37 .36 .35];
t=0;v=0;vy=0;v=0;vx=0;vy=0;ax=0;ay=0;rail=1;m=Mo+Mbo;
theta=Launch_Angle;Cf=0;Drag=0;
Drag_Loss=0;Gravity_Loss=0;gamma=1.4;recovery=1;St=.01;Cp=1004;k=.48;
c_cfrp=1004;
rho_cfrp=1700;
rho_cork_eng = 30;%lbm/ft^2
rho_cork = rho_cork_eng /2.2/144/.0254^2;%kg/m^3
kvc_eng = .044; %BTU/hr/ft/F
kvc = kvc_eng * 1077/3600/12/.0254*9/5; %W/m/K
Cp_cork_eng = .47; %BTU/lbm/F
Cp_cork = Cp_cork_eng*1077*2.2*9/5; %J/kg/K

f=1;e=.9;Boltz=5.67e-8;Qside_tot=0;
tcf = .030*.0254;
tvc = .25*.0254;
tch = 0;
wall_thickness= tcf+tvc+tch; [a b]=size(wall_temp);dL=wall_thickness/b;
wall_position=[dL:dL:wall_thickness];
therm_time=100;j=b;
dt=.1;
dt_therm = dt/therm_time;

Start_Time = clock;

while(y>=0)
if (m > Mf)
    if (t < Booster_Burn_Time)
        T = Thrust + Boost_THrust;
        m = m - m-dot*dt - m-dotb*dt;
    else
        if (t < Booster_Burn_Time + dt)
            m = m - Mbf;  % Separation...
        end
        T = Thrust;
        m = m - m-dot*dt;
    end

    Drag_Loss = Drag_Loss + Drag*dt/m;
    Gravity_Loss = Gravity_Loss + g*sin(theta)*dt;
else
    if (T == 0)
        Burnout_Time = t;
    end
    T = 0;
end

if (sqrt(x^2 + y^2) > Launch_Rail_Length)
    rail = 0;
end

g = 6.67E-11 * 5.98E24 / (6.38E6 + y)^2;

if y < 11000
    density = 1.225 * (1 - 0.0226 * y / 1000)^4.256;
else
    density = (0.3629) * exp(-g * (y - 11000) / 286.97 / 288.16);
end

% std density is 1.225 kg/m/m/m at sea level, but varies
% with altitude My model is based on Aerodynamics, Aeronautics
% and flight Mechanics by McCormick on pages 23-25.
ambient_temp = interp1(tmper(:,1),tmper(:,2),y);
c = sqrt(1.4*287*ambient_temp);
M = v/c;

if (t < Booster_Burn_Time)
    Drag = cd*.5*density*v^2*(S + Sb);
else
    Drag = cd*.5*density*v^2*S;
end

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Wall Temperature Changes with heat flux

Two = ambient_temp*(1 + recovery*(gamma-1)*M^2/2);

tt = 0;

while (tt < dt)
while(j>0)
    if(j==1)
        dT_out_side = wall_temp_prev(j) - wall_temp_prev(j+1);
        q_side = St*density*v*Cp*(Two - wall_temp_prev(j));
        wall_temp(j) = wall_temp_prev(j) + dt_therm*(q_side -
            kvc*dT_out_side - e*Boltz*wall_temp_prev(j)^4)/(rho_cork*Cp_cork*dL) ;
        %dT_out_nose = nose_wall_temp_prev(j) -
        %nose_wall_temp_prev(j+1);
        %q_nose =
            St*density*(1+((gamma-1)*M^2/2)^((1/(gamma-1)))*v*Cp*(Two -
            wall_temp_prev(j));
        %nose_wall_temp(j) =
            nose_wall_temp_prev(j) + dt_therm*(q_nose -
            kvc*dT_out_nose - e*Boltz*nose_wall_temp_prev(j)^4)/(rho_cfrp*C_cfrp*dL) ;
        else
            if(j<b)
                dT_in = wall_temp_prev(j-1) - wall_temp_prev(j);
                dT_out = wall_temp_prev(j) - wall_temp_prev(j+1);
                wall_temp(j) = dt_therm*(dT_in*dT_out)/(rho_cork*Cp_cork)*dL^2 +
                    wall_temp_prev(j);  
                %dT_in_nose = nose_wall_temp_prev(j-1) -
                    nose_wall_temp_prev(j);  
                %dT_out_nose = nose_wall_temp_prev(j)-
                    nose_wall_temp_prev(j+1);
                %nose_wall_temp(j) =
                    dt_therm*(dT_in_nose - dT_out_nose)/(rho_cfrp*C_cfrp)*dL^2 +
                    nose_wall_temp_prev(j);
            else
                dT_in = wall_temp_prev(j-1) - wall_temp_prev(j);
                wall_temp(j) =
                    k*(dT_in/dL)*dt_therm/(rho_cork*Cp_cork) +
                    wall_temp_prev(j);  
                %dT_in_nose = nose_wall_temp_prev(j-1) -
                    nose_wall_temp_prev(j);  
                %nose_wall_temp(j) =
                    k*(dT_in_nose/dL)*dt_therm/(rho_cfrp*C_cfrp) +
                    nose_wall_temp_prev(j);
            end
        end
    end
    j = j - 1;
end
wall_temp_prev = wall_temp;
nose_wall_temp_prev = nose_wall_temp;
j = b;
tt = tt + dt_therm;
end

%figure(3);
%plot(wall_position,wall_temp);
%drawnow;
%hold on;

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

if (rail == 1)
    theta = Launch_Angle;
    NormalF = m*g*cos(theta);
    if (T*sin(theta) > g*sin(theta))
        arail = (T*sin(theta) - m*g*sin(theta) - Cf*NormalF - Drag)/m;
    else
        arail = (T*sin(theta) - m*g*sin(theta) + Cf*NormalF + Drag)/m;
    end

    ax = arail*cos(theta);
    ay = arail*sin(theta);
else
    theta = atan(vy/vx);
    ax = (T-Drag)*cos(theta)/m;
    ay = (T-Drag)*sin(theta)/m - g;
end

x = x + vx*dt + .5*ax*dt^2;
y = y + vy*dt + .5*ay*dt^2;
vx = vx + ax*dt;
vy = vy + ay*dt;

v = sqrt(vx^2 + vy^2);
t = t + dt;

if (y > yprev)
    Qside_tot = Qside_tot + qside*dt;
end

State = [t m x y v M qside];% qnose];
Wall_Temperature_Profile(i,:) = wall_temp;
Nose_Wall_Temperature_Profile(i,:) = nose_wall_temp;
Flight(i,:) = [State];
i = i + 1;
y prev = y;
end

LostV = Drag_Loss + Gravity_Loss
Calculation_Time = etime(clock, Start_Time)
Heat_absorbed = Qside_tot * surface_area
Qab_eng = 2500; % BTU/lbm
Qab = Qab_eng * 1077 * 2.2; % J/kg
% If all heat goes to ablation and none goes to heating up carbon fiber or virgin cork
% ablative_mass = Heat_absorbed/Qab
min_ablative_thickness = ablative_mass / rho_cork / surface_area + .020*.0254; % extra is because of bad thin layer performance
min_ablative_thickness_eng = min_ablative_thickness/.0254

subplot(3,2,1)
plot(Flight(:,3),Flight(:,4));
xlabel('Distance (m)');
ylabel('Altitude (m)');

subplot(3,2,2)
plot(Flight(:,1),Flight(:,2));
ylabel('Mass (kg)');
xlabel('Time (s)');
axis([0 Burnout_Time 0 (Mo+Mbo)]);

subplot(3,2,3)
plot(Flight(:,4),Flight(:,6));
xlabel('Altitude (m)');
ylabel('Mach #');

subplot(3,2,4)
plot(Flight(:,1),Flight(:,5));
xlabel('Time (s)');
ylabel('Velocity (m/s)');

subplot(3,2,5)
plot(Flight(:,1),Flight(:,4));
xlabel('Time (s)');
ylabel('Altitude (m)');

subplot(3,2,6)
plot(Flight(:,1),Flight(:,7));
xlabel('Time (s)');
ylabel('Convective Heat Flux (W/m^2)');
hold;
plot(Flight(:,1),Flight(:,8),'r');
legend('side','nose');

figure(2);
mesh(Wall_Temperature_Profile);
Appendix B: Dynamic Engine Startup Simulator “CDIEsim.m”

with supporting code “kerosene_massflow.m,”
“ox_massflow.m,” “chamber_mix.m,” and “thrust.m,” with input
file “constants.m” and output file “simplots.m”

“constants.m”

%%% Input parameters and constants:

pi = 3.1416;
g = 9.80665;

Pa = 1.0135e5; % ambient pressure (sea level)
start_speed = 3300; % rad/s
wall_temp = 300;  % hot gas side wall temperature
B = .00012; %Bearing loss factor % the .00012 damping factor is from
the disk spin down times...
Cd = .08;%.0477; % Drag coefficient based on tip speed and disk area

Throat_diameter = .0128;
At = pi* Throat_diameter^2/4;
expansion_ratio = 20;
Ae = At*expansion_ratio;

rox = .042;  % radius of the highest pressure oxygen nozzle inside
the disk
rf = .0342;  % .0508 radius of the highest pressure kerosene nozzle
(at edge of disk)
rjet = .053;  % .048 radius of oxygen injector jet
rdisk = .051; % disk radius for heat transfer...

Theta = 15*pi/180;  %57.3 angle of oxygen jets from the tangent line
phi = 0*pi/180;  % angle of kerosene jets from the radius line
twall = .0033;  % disk wall thickness (minimum)

% Some cooling passage dimensions
f_cool = .05;
h = .136*.0254;
w = .140*.0254;
A_passage = h*w;
perim_passage = 2*h+2*w;
rhyd = A_passage/perim_passage;
D_equiv =4*rhyd;
L_passage = 3*(pi*rox)+3.2*(pi*rox); % very approximate;

rchamber = .063; % chamber radius
thickness = .017%.025; % the height of the disk -- for heat transfer
disk_mass = .5;
I = .5*disk_mass*rdisk^2; % moment of inertia of disk
L = .10; % length of the combustion chamber
V_chamber = 1.5*pi*rchamber^2*L; % combustion chamber volume

Doxinj = .0023; % Diameter of ox injector
Aoxinj = pi*Doxinj^2/4; % area of oxygen injector
Dfinj = .033*.0254; % Diameter of fuel injector
Afinj = pi*Dfinj^2/4; % area of fuel injector
Adisk = pi*rdisk^2; % area of disk for drag calc

n_oxinj = 2; % number of oxygen jets
n_finj = 4; % number of fuel injectors

Cdfinj = .6; % discharge coefficient of fuel injection nozzles
Cdoxinj = .7;

gam_ox = 1.4; % ratio of specific heats of oxygen
delta_Hfgox = 213000; % J/kg -- heat of vaporization of LOX
R_ox = 259; % J/kg/K -- oxygen gas constant
Cp_ox = 910; % J/kg/K -- constant pressure specific heat of oxygen
T_LOX = 90; % K -- temperature of LOX at highest pressure point in system
rho_LOX = 1140; % kg/m^3 -- density of LOX
rho_f = 810; % kg/m^3 -- density of kerosene

sigma_y_Al = 505e6; % kg/m^3 density Al
rho_Al = 2800; % kg/m^3 density Al
k_Al = 130; % W/m/K thermal conductivity of Al
E_Al = 72e9; % Pa elastic modulus
CTE_Al = 23.6e-6; % /K coef thermal expansion
nu_Al = .33; % Poisson's ratio

omega_burst = sqrt(sigma_y_Al/(pi*sin(pi/4)*rho_Al*rdisk^2));
```

CDIEsim.m

% simulation routine
clf;
clear all;
constants;
t = 0;
dt = 1e-4;
Pc = 101350;
cstar = 350;
omega = start_speed;
qdot = 1;
mdot_ox_prev = 0;
j = 1;

while (t < dt*5e4)
    mdot_out = Pc*At/cstar;
    % [mdot_ox, mdot_kero, Pc, F_oxjet,Tc,R,gamma] =
    chamber_steady_state(omega, qdot);
    [mdot_kero, Pf, uf] = kerosene_massflow(Pc, omega);
    [mdot_ox, T_four, F_oxjet,Pmax] = ox_massflow(Pc, omega, qdot, mdot_ox_prev);

    [Tc,R,gamma,cstar, flag] =
    chamber_mix((mdot_ox+mdot_ox_prev)/2/mdot_kero); % change 5/4/02 mdot_ox
    dPc = ((mdot_ox+mdot_kero)-mdot_out)*dt*R*Tc/V_chamber;
    rho_chamber = Pc/(Tc*R);
    residence_time = V_chamber*rho_chamber/mdot_out;

    % Stress in disk:
    sigma_rot = sin(pi/4)*pi*rhoAl*omega^2*rdisk^2;
    sigma_press = (Pmax-Pc)*rdisk/twall;
    sigma_theta = sigma_rot+sigma_press;
    sigma_z = (Pmax-Pc)*Adisk/(2*pi*rdisk*twall);
    % Mises Yield Criterion:
    Y = sqrt(0.5*{(sigma_theta-sigma_z)^2+sigma_z^2+sigma_theta^2});
    MOS = sigma_yAl/Y;

    % Thermal Stress
    q_A = qdot/(Adisk+2*pi*rdisk*thickness);
    dT_w = q_A*twall/k_A;
    sigma_thermal = 2*CTEAl*E_Al*dT_w/(1-nu_Al);
```
Tnet = F_oxjet*rjet*cos(Theta)+mdot_kero*uf*rf*sin(phi)-
(mdot_ox*omega*rox^2)-(mdot_kero*omega*rf^2)-B*omega-
Cd*.5*rho_chamber*omega^2*risk^3*Adisk;
Ox_massflow = mdot_ox;
Kero_massflow = mdot_kero;
Jet_Force = F_oxjet;
Chamber_Pressure = Pc;
Thrust= thrust(mdot_out,expansion_ratio,Ae,Pc,Pa,Tc,gamma,R);
gdot = heat_flux(omega,Pc,Tc,R);
omega_dot = Tnet/I;
if (mdot_ox+mdot_kero>0 & Thrust>0)
   Isp = Thrust/((mdot_ox+mdot_kero)*g);
else
   Isp = 0;
end
State(j,:) = [t Thrust Isp Ox_massflow Kero_massflow Jet_Force
Chamber_Pressure Tnet gdot omega Tc MOS sigma_thermal residence_time];
omega = omega + omega_dot*dt;
mdot_ox_prev = mdot_ox;
Pc = Pc + dPc;
j = j+1;
t = t + dt;
if(rem(j,500) == 0)
   t =t
end
end
simplots;
"simplots.m"
%simplots.m -- to plot the state of the engine

subplot(2,5,1);
plot(State(:,1),State(:,2));
title('Thrust');

subplot(2,5,2);
plot(State(:,1),State(:,3));
title('Isp');

subplot(2,5,3);
plot(State(:,1),State(:,4));
title('oxm');

subplot(2,5,4);
plot(State(:,1),State(:,5));
title('kerom');

subplot(2,5,5);
plot(State(:,1),State(:,6));
title('jetforce');

subplot(2,5,6);
plot(State(:,1),State(:,7));
title('Pc');

subplot(2,5,7);
plot(State(:,1),State(:,8));
title('Tnet');

subplot(2,5,8);
plot(State(:,1),State(:,9));
title('qdot');

subplot(2,5,9);
plot(State(:,1),State(:,10));
title('omega');

subplot(2,5,10);
plot(State(:,1),State(:,11));
title('Tc');

figure(2);
plot(State(:,1),State(:,12));
title('MOS');

figure(3);
plot(State(:,1),State(:,13));
title('Max Thermal Stress');

figure(4);
plot(State(:,1),State(:,7));
title('Chamber Pressure During Startup Transient');
axis([0 .1 0 3.5e6]);
xlabel('Time (s)');
ylabel('Chamber Pressure (Pa)');

figure(5);
plot(State(:,1),State(:,14));
title('Average Chamber Residence Time');

v=axis;
axis([0 .1 v(3) v(4)]);
xlabel('Time (s)');
ylabel('Chamber Residence Time (s)');
"kerosene_massflow.m"
%%% kerosene massflow solver

function [mdot_kero, Pf, uf] = kerosene_massflow (Pc, omega)

constants;

Pf = .5*rho_f*omega^2*rf^2;

if (Pf>Pc)
    mdot_kero = nfinj*Cdfinj*Afinj*sqrt(2*(Pf - Pc)*rho_f);
    uf = Cdfinj*sqrt(2*(Pf - Pc)/rho_f);
else
    if (Pc>Pf)
        mdot_kero = -nfinj*Cdfinj*Afinj*sqrt(2*rho_f*(Pc-Pf));
        uf = Cdfinj*sqrt(2*(Pc-Pf)/rho_f);
    else
        mdot_kero = 0;
        uf = 0;
        BLAHHHHH='bad';
    end
end

"ox_massflow.m"
%%% oxygen massflow solver

function [mdot_ox, T_four, F_oxjet, u_ox] = ox_massflow (Pc, omega, qdot, mdot_oxprev)

constants;

v_passage = mdot_oxprev/(rho_LOX*A_passage);

P_two = .5*rho_LOX*omega^2*rox^2;
P_four = P_two - f_cool*.5*rho_LOX*v_passage^2*L_passage/D_equiv; % estimation for pressure loss in heat exchange passages

if(P_four > Pc)
    if ((Pc/P_four) <= (2/(gam-ox+1))^(gam_ox/(gam_ox-1)) )
        a = (T_LOX - delta_Hfg_ox/Cp_ox);
        b = (qdot/Cp_ox);
        c = (-P_four^2*Aoxinj^2*gam_ox*(2/(gam_ox+1))^((gam_ox+1)/(gam_ox-1))/R_ox);
        if (4*a*c < b^2)
            mdot_ox = n_oxinj*(-b + sqrt(b^2-4*a*c))/(2*a);
            mdot_ox_two = n_oxinj*(-b - sqrt(b^2-4*a*c))/(2*a);
            T_four = T_LOX + qdot/(Cp_ox*mdot_ox) - delta_Hfg_ox/Cp_ox;
            if (T_four < T_LOX)
                T_four = T_LOX;
        end
end
\[ u_{\text{ox}} = \sqrt{g_{\text{ox}} R_{\text{ox}} T_{\text{four}}}; \]
else %Not enough heat -- mixed phase injection
\[ \text{mdot}_{\text{ox gas}} = \frac{q_{\text{dot}}}{\Delta H_{f g_{\text{ox}}}}; \]
\[ c_{\text{gas}} = \sqrt{g_{\text{ox}} R_{\text{ox}} T_{\text{LOX}}}; \]
\[ c_{\text{star}} = \frac{3}{2} c_{\text{gas}}; \]
\[ T_{\text{four}} = T_{\text{LOX}}; \]
\[ A_{\text{gas}} = \frac{\text{mdot}_{\text{ox gas}} c_{\text{star}}}{P_{\text{four}}}; \]
\[ A_{\text{Liq}} = A_{\text{OX inj}} - A_{\text{gas}}; \]
\[ \text{if} \ (A_{\text{Liq}} < 0) \]
\[ \quad \text{disp}('\text{Warning: negative liquid flow area -- unsteady mixed phase!}') ; \]
\[ \quad A_{\text{Liq}} = 0; \]
\[ \text{end} \]
\[ v_{\text{liquid}} = c_{\text{gas}}/2; \quad \% \text{Estimation of liquid droplet velocity in gas stream.} \]
\[ \text{mdot}_{\text{ox liq}} = \rho_{\text{LOX}} A_{\text{Liq}} v_{\text{liquid}}; \]
\[ \text{mdot}_{\text{ox}} = \text{mdot}_{\text{ox gas}} + \text{mdot}_{\text{ox liq}}; \]
\[ X = \frac{\text{mdot}_{\text{ox gas}}/\text{mdot}_{\text{ox}}}{\text{mdot}_{\text{ox}}}; \]
\[ u_{\text{ox}} = (X c_{\text{gas}} + (1-X) v_{\text{liquid}}); \]
\[ \text{end} \]
\[ F_{\text{OX jet}} = \text{mdot}_{\text{ox}} u_{\text{ox}} + A_{\text{OX inj}} (P_{\text{four}} - P_c); \]
else % oxygen is subsonic
\[ a = (T_{\text{LOX}} - \Delta H_{f g_{\text{ox}}}/C_{p_{\text{ox}}}); \]
\[ b = (q_{\text{dot}}/C_{p_{\text{ox}}}); \]
\[ c = (-2(P_{\text{four}} - P_c)P_c A_{\text{OX inj}}^2 / R_{\text{ox}}); \]
if (4*a*c < b^2)
\[ \text{mdot}_{\text{ox}} = n_{\text{OX inj}} ((-b + \sqrt{b^2 - 4*a*c})/(2*a)); \]
\[ \% \text{mdot}_{\text{ox two}} = n_{\text{OX inj}} ((-b - \sqrt{b^2 - 4*a*c})/(2*a)); \]
\[ T_{\text{four}} = T_{\text{LOX}} + q_{\text{dot}}/(C_{p_{\text{ox}}} \text{mdot}_{\text{ox}}) - \Delta H_{f g_{\text{ox}}}/C_{p_{\text{ox}}}; \]
\[ \text{if} \ (T_{\text{four}} < T_{\text{LOX}}) \]
\[ \quad T_{\text{four}} = T_{\text{LOX}}; \]
\[ \text{end} \]
\[ u_{\text{ox}} = \frac{\text{mdot}_{\text{ox}} R_{\text{ox}} T_{\text{four}}}{(P_c A_{\text{OX inj}})}; \]
else %Not enough heat -- LOX is directly injected
disp('\text{Warning: negative liquid flow area -- unsteady mixed phase!}') ;
\[ \text{mdot}_{\text{ox gas}} = q_{\text{dot}}/\Delta H_{f g_{\text{ox}}}; \]
\[ c_{\text{gas}} = \sqrt{g_{\text{ox}} R_{\text{ox}} T_{\text{LOX}}}; \]
\[ c_{\text{star}} = \frac{3}{2} c_{\text{gas}}; \]
\[ T_{\text{four}} = T_{\text{LOX}}; \]
\[ A_{\text{gas}} = \frac{\text{mdot}_{\text{ox gas}} c_{\text{star}}}{P_{\text{four}}}; \]
\[ A_{\text{Liq}} = A_{\text{OX inj}} - A_{\text{gas}}; \]
\[ \text{if} \ (A_{\text{Liq}} < 0) \]
\[ \quad \text{disp('Warning: negative liquid flow area -- unsteady mixed phase!');} \]
\[ \quad A_{\text{Liq}} = 0; \]
\[ \text{end} \]
\[ v_{\text{liquid}} = c_{\text{gas}}/2; \quad \% \text{Estimation of liquid droplet velocity in gas stream.} \]
\[ \text{mdot}_{\text{ox liq}} = \rho_{\text{LOX}} A_{\text{Liq}} v_{\text{liquid}}; \]
mdot_ox = mdot_ox_gas + mdot_ox_liq;
X = mdot_ox_gas/mdot_ox;
u_ox = (X*c_gas+(1-X)*v_liquid);
end
F_oxjet = mdot_ox*u_ox;
end

else % if Pc >= P_four
mdot_ox = 0;
T_four = T_LOX;
F_oxjet = 0;
u_ox = 0;
end

"chamber_mix.m"
% chamber temp as function of mixture ratio

function [Tc, R, gam_chamber, cstar, flag] = chamber_mix(mass_ratio)

% propellants are kerosene and LOX
% can get all relevant data from CEA...

% format = [O/F Tc cstar gamma molec_wt]
data = [0.60 500 800 1.35 15,
0.80 1000 1350 1.3 16,
1.60 2750 1690 1.2 18,
1.80 3100 1760 1.18 20,
2.00 3418 1838 1.16 21,
2.25 3553 1836 1.14 23,
2.50 3614 1817 1.14 25,
2.60 3680 1800 1.13 26,
2.80 3710 1780 1.13 27,
3.00 3700 1750 1.13 28,
18.0 1600 1000 1.36 31];

[a,b]=size(data);

if (mass_ratio > data(a,1))
flag = 1;
cstar = 350;
Tc = 300;
gam_chamber = 1.4;
R = 8314/32;
else
if (mass_ratio < data(1,1))
flag = -1;
cstar = 100;
Tc = 299;
gam_chamber = 1.4;

end

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R = 8314/14;
else
    Tc = interp1(data(:,1),data(:,2),mass_ratio);
cstar = interp1(data(:,1),data(:,3),mass_ratio);
gam_chamber = interp1(data(:,1),data(:,4),mass_ratio);
molec_wt = interp1(data(:,1),data(:,5),mass_ratio);
    R = 8314/molec_wt;
    flag = 0;
end
end
"thrust.m"

%% find Thrust given expansion ratio, Pc and Pa

function [Thrust] = thrust(mdot,A_R,Ae,Pc,Pa,Tc,gamma,R)

if (Pc/Pa > ((gamma-1)/2)^(gamma/(gamma-1)) )
     Me = 1.05;
     dMe = .05;
     A_R_err = 1;
     A_Rtest = (1/Me)*((1+(gamma-1)/2*Me^2)/(1+(gamma-1)/2))^(gamma+1)/(gamma-1);

     while (A_Rtest < A_R - A_R_err)
         Me = Me + dMe;
         A_Rtest = (1/Me)*((1+(gamma-1)/2*Me^2)/(1+(gamma-1)/2))^(gamma+1)/(2*(gamma-1));
     end
     Te = Tc/(1+(gamma-1)/2*Me^2);
     Pe = Pc/(Tc/Te)^(gamma/(gamma-1));
     ae = sqrt(gamma*R*Te);
     ue = Me*ae;

     if(Pe<.4*Pa)
         Pe = .4*Pa;
         Te = Tc*(Pe/Pc)^(gamma-1)/(gamma));
         Me = sqrt(2/(gamma-1)*(Tc/Te-1));
         A_R_eff = (1/Me)*((1+(gamma-1)/2*Me^2)/(1+(gamma-1)/2))^(gamma+1)/(2*(gamma-1));
         Ae = Ae/A_R*A_R_eff;
     end

     Thrust = mdot*ue + Ae*(Pe-Pa);
else
     Thrust = 0;
end