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INTERIM ACCESS TO THE INTERNATIONAL SPACE STATION

by

Tyson Karl Smith

A thesis submitted in partial fulfillment
of the requirements for the degree

of

MASTER OF SCIENCE

in

Mechanical Engineering

Approved:

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Logan, Utah

2010

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Abstract

Interim Access to the International Space Station

by

Tyson Karl Smith, Master of Science

Utah State University, 2010

Major Professor: Dr. Stephen A. Whitmore
Department: Mechanical and Aerospace Engineering

This thesis evaluates mission scenarios using the existing Evolved Expendable Launch Vehicles for delivering the Crew Exploration Vehicle to the International Space Station. The Space Shuttle is scheduled to retire in the year 2011 and the Ares I is being developed to replace it. With its current schedule, the earliest that the Ares I will become fully operational is 2016. The configurations in this thesis are presented to narrow the gap in which the USA does not have direct access to the International Space Station. They also present "buy down" options for the USA human space operations, if the current development issues of the Ares I cause it to not become operational at all. The three Launch options presented are the Atlas V HLV, the Delta IV Heavy, and the Delta IV with three common core boosters as the first stage and the Orion service module to be used as the second stage. The first configuration, the Atlas V HLV requires significant impulse from the Orion service module in order to reach the final International Space Station orbit. The second option, the Delta IV Heavy, launches the Orion as a passive payload and requires no impulsive maneuvering from the service module in order to reach the International Space Station orbit. The third configuration, the Delta IV Heavy with three common core boosters as the first stage, and the Orion spacecraft acting as the second stage, requires significant impulse from Orion's service module engine to achieve the International Space Station orbit. After final orbit

insertion all three configurations still have sufficient propellant for de-orbit and re-entry. The third configuration has a certain appeal, by eliminating the second stage only the common core booster on the Delta IV Heavy system need be human-rated. Finally, reliability and development cost assessments are presented and compared to the Ares I.

(108 pages)

To all those who didn't believe....

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Acronyms

ACS	Attitude Control System
AFRL	Air Force Research Laboratory
CBC	Common Booster Core
CCB	Common Core Booster
Cd	Aerodynamic Drag Coefficient
CEA	Chemical Equilibrium with Applications
CEV	Crew Exploration Vehicle
Cl	Aerodynamic Lift Coefficient
CONOPS	Concept of Operations
DOD	Department of Defense
EELV	Evolved Expendable Launch Vehicle
ELV	Expendable Launch Vehicle
GEO	Geostationary Orbit
g_0	Acceleration of Gravity at Sea Level, 9.8067 m/sec^2
GUI	Graphical User Interface
h	Orbit Altitude, km
i	Orbit Inclination, deg
Isp	Specific Impulse, sec
ISS	International Space Station
lat	Earth Latitude
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LO2	Liquid Oxygen
POST	Program for the Optimization of Simulation Trajectories
Re	Earth Radius, km
RMS	Root Mean Squared
RSRM	Reusable Solid Rocket Motor

SM	Service Module
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
TVC	Thrust Vector Control
μ	Planetary Gravitational Parameter for Earth, $3.986004418 * 10^5 km^3/sec^2$
USU	Utah State University
Vboost	Equivalent Launch Velocity Along Track Due to Earth's Rotation, km/sec
Vdrag	Velocity Loss Due to Aerodynamic Drag, km/sec.
Veast	Launch Velocity Due to Change in Orbital Plane, km/sec
Vgravity	Launch Velocity Required to Overcome Gravity, km/sec
Vnorth	Launch Velocity Launch Latitude, km/sec
Vorbit	Orbital Velocity, km/sec
Ω	Angular Velocity of Earth, km/sec
wi	Initial Weight of Launch Vehicle, kg
wf	Final Weight of Launch Vehicle, kg

Chapter 1

Introduction

A new human-rated space transportation system is being developed under NASA's Constellation Program. This new space travel system is being designed with the goal of establishing a sustainable human base on the Moon, which in turn will be a stepping stone in carrying humans to Mars. Key components of this system are the Orion spacecraft, Altair Lunar Lander, and the Ares I and Ares V launch vehicles. When development is complete this system will become the United States of America's primary space access system, replacing the Space Shuttle. For missions to the Moon the heavy-lift Ares V cargo launch vehicle will deliver the Altair lunar module along with an additional rocket motor, that will be used as an Earth departure stage, to Low-Earth orbit (LEO). The crew will then be launched. The Crew Exploration Vehicle (CEV) or Orion spacecraft will house the crew. The Orion will be mounted upon the Ares I launch vehicle. The Ares I will deliver the Orion to LEO where the Orion will rendezvous with the Altair. The Earth departure stage will boost both vehicles into a trans-lunar trajectory. The Ares I/Orion combination will also be used to deliver crew to the International Space Station (ISS). The Orion spacecraft is patterned after the Apollo capsule; however the Orion is two-and-a-half times larger than the Apollo. The Orion is 5.5 meters in diameter and weighs approximately 22.7 metric tons (25 tons). A launch abort system (LAS) has been added to the Orion, which will pull the crew and spacecraft to safety in the event of an emergency on the launch pad or at any time during the ascent [1].

The Ares I launch vehicle [2] is made up of an in-line, two-stage configuration topped by the Orion Crew Module (CM), the Service Module (SM) propulsion stage, and the Launch Abort System (LAS). The Ares I first stage is a five-segment reusable solid rocket booster that is based on the solid rocket motors used on the Space Shuttle. The first stage will be

mated with the upper stage via a newly designed forward adapter that will be equipped with booster separation motors that will separate the two stages during ascent. The upper stage is derived from the J-2 and the J-2S. The J-2 engine propelled the Saturn IB and Saturn V rockets, and the J-2S was designed in the early 1970's but never flew. The Ares I stands 309 feet tall and is referred to as the "big stick." In comparison the Space Shuttle stands at only 184 feet, The Ares launch vehicles are shown in Figure 1.1 [1] [3].

Although the Ares V [4] is in the early stages of development, it is currently proposed that the Ares V will be derived from shuttle hardware already in service. The Ares V first stage relies on two five-and-a-half segment reusable solid rocket boosters that are strapped onto a core booster tank; these boosters are derived from the Space Shuttle's reusable solid rocket motors (RSRM). The core tank feeds liquid oxygen and liquid hydrogen propellants to a group of six RS-68B engines. These engines are upgrades from the Delta IV, a launch vehicle developed by the U.S. Air Force as part of its Evolved Expendable Launch Vehicle program. This core stage is connected to the Ares V Earth departure stage with an interstage cylinder that contains booster separation motors. The Earth departure stage has a J-2X as an engine and is also powered by liquid oxygen and liquid hydrogen, which is also an evolved variation of the J-2 and the J-2S. During launch the reusable solid strap-on boosters and the core propulsion stage lift the Ares V to LEO. After separation from the core stage the Earth departure stage's J-2X is lit, placing the vehicle in a circular orbit. The shroud will then separate in preparation to rendezvous with the Orion capsule. [3].

The Ares I/Orion combination will also be used to deliver crew to the ISS. The Ares I will carry between three and six crew members to the ISS for a six-month stay and will be capable of returning them safely to Earth at any time during the mission. A profile of this mission is shown in Figure 1.2. The elements necessary to fulfill this mission are the Crew Exploration Vehicle (CEV) and the Ares I. The CEV consists of two parts: the Crew Module (CM) and the Service Module (SM). Fig. 1.3 shows the components of the CEV. The Ares I, also known as the Crew Launch Vehicle (CLV), will launch the CEV into an insertion orbit of 56 x 296 km at an inclination of 51.6 degrees, with a crew of either

Fig. 1.1: Ares Launch Vehicles.

Fig. 1.2: Ares I ISS Mission Profile.

three or six members, intended for a six-month ISS mission. Pre-mission defined phasing burns will be used to close in on the ISS once the CEV has reached the ISS orbit. These burns consist of a combination of ground and onboard targeted burns, along with burns performed once rendezvous navigation sensors acquire the ISS. Once the CEV has been properly docked with one of the two available CEV-compatible docking ports, the CEV is configured to a quiescent state and becomes a possible rescue vehicle for the duration of the mission. After the six-month ISS mission is complete the crew performs a health and safety check, closes the hatch, and checks for leaks. Once the CEV undocks from the ISS, the CEV exits the vicinity of the ISS and conducts an onboard targeted de-orbit burn. After this burn is complete, the CEV SM is jettisoned, and the CEV CM is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV executes the supposed landing at the primary land-based landing site. The current proposed mass for a fully loaded Orion spacecraft designed for an ISS mission carrying six crew members is 27,217 kg (60,003 lbs). A mass budget for the ISS configuration is shown in Table 1.1.

The Ares I will also be capable of carrying up to four crew members to explore the Moon for up to seven days. These missions are similar to the Apollo Moon surface missions that were performed in the late 60's early 70's. The Ares I will be capable of landing humans on the Moon, operate for a limited period of time on the surface of the Moon, and then return the crew safely to Earth. These missions are referred to as lunar sorties. Sortie missions allow for exploration and scouting for possible future lunar outposts. The current

Fig. 1.3: CEV Subsystem.

proposed mass for a fully loaded Orion spacecraft designed for a lunar mission carrying four crew members is 29,952 kg (66,033 lbs). A mass budget for the lunar configuration is shown in Table 1.2 [5] [6].

Table 1.1: Mass Budget for the CEV ISS Mission Configuration

Item	LAS	CM	SM	Launch Adapter	Total
Dry Mass, kg	2,596	8,779	4,938	1,640	17,953
Propellant Mass, kg	4,469	136	3,666	–	8,270
Consumable Mass, kg	–	35	207	–	242
4 Man Crew Mass, kg	–	760	–	–	760
Gross Launch Mass, kg	7,065	9,710	8,810	1,640	27,224

Table 1.2: Mass Budget for the CEV Lunar Mission Configuration

Item	LAS	CM	SM	Launch Adapter	Total
Dry Mass, kg	2,596	8,779	4,116	1,640	17,131
Propellant Mass, kg	4,469	136	7,910	–	12,515
Consumable Mass, kg	–	35	315	–	350
4 Man Crew Mass, kg	–	860	–	–	860
Gross Launch Mass, kg	7,065	9,810	12,340	1,640	30,855

Chapter 2

Ares Development Issues

With the Space Shuttle set to retire in early 2011, it is important that the Shuttle's replacement become operational as soon as possible. During this interim period U.S. human space operations must rely on the Russian Soyuz or possible commercial launchers (COTS) for crew access the International Space Station. [7] Even if the currently proposed time line is 100 percent successful, there will exist a five year operations gap where the USA will not have self-determined access to the ISS. Any further delay in the Ares I schedule will significantly impact, and could terminate the ISS science and mission operations.

The first Ares flight test, dubbed the Ares I-X, took place on October 28, 2009. Fig. 2.1 is a picture of the Ares I-X on the launch pad at NASA's Kennedy Space Center in Florida, before the launch on October 28, 2009. During its launch the Ares I-X reached an acceleration of nearly 3 g's and a Mach of 4.76, more complete results of the flight test are scheduled to be released in December of 2009. This demonstration was the first launch of the reusable solid rocket motor (RSRM). From this flight test NASA is hoping to gain valuable vibration performance data during the ascent of the stage. This test launched the integrated stack consisting of the four segment RSRM, a simulated upper stage, the Orion crew module, and the launch abort system. The Ares I-X is similar in size and mass to the actual Orion capsule and the Ares I launcher, but it includes a combination of simulated and space flight proven hardware. The test vehicle is powered by flight hardware that is currently used in the Space Shuttle, a four-segment reusable solid rocket booster. This configuration will be modified to include a fifth segment to simulate the Ares I five segment booster. Mock-ups of the upper stage, Orion crew module, and launch abort system are used to simulate the integrated spacecraft system. Fig. 2.2 shows the Ares I-X configuration [8].

Fig. 2.1: Ares I-X on the Launch Pad at NASA's Kennedy Space Center in Florida Just Before Launch (Image courtesy of www.nasa.gov).

Fig. 2.2: Ares I-X Configuration.

2.1 Thrust Oscillation Issues

Preliminary structural analysis on the Ares I indicated that a potential resonant interaction results in high dynamic longitudinal force levels in the upper stage and in the Orion capsule, these oscillations could potentially harm the crew. Conservative calculations of the frequency and amplitude of a thrust oscillation that could occur in the first stage as it burns out, and the way that vibration propagates through out the rest of the vehicle could cause a resonance that would damage critical components and harm the astronauts.

Thrust oscillation is a common problem in solid rocket motors with concerns arising when pressure oscillations drive resonant modes in the vehicle's structure. The elements of a solid rocket motor consist of: the solid propellant or grain, a case, a nozzle, thermal protections, and the igniter. For solid rocket motors, which are characterized by a segmented grain geometry, vortices in the combustion chamber flow can be produced by, obstacle vortex shedding behind the tip of the frontal thermal protection devices, angle vortex shedding off angles in the propellant grain geometry, or by parietal vortex shedding from the propellant surface. Figure 2.3 shows these three scenarios of vortex shedding. The vortex shedding within the motor can create a feedback type loop of oscillation with the oscillation starting small and growing exponentially over time. The interaction between aerodynamics and acoustics is the center of the pressure oscillation in the motors combustion chamber. When the vortex shedding is the same or near the natural frequency of the chamber, the oscillations become extremely large [9] [10].

Fig. 2.3: Three Forms of Vortex Shedding.

These oscillations are also sometimes present in liquid fueled rockets, as was the case with the Titan II. In a liquid rocket the oscillations are caused as the launch vehicle begins to vibrate during launch, which in turn causes the flow of propellant in the feed lines and

engine to vibrate causing an uneven flow of fuel. This uneven flow of fuel causes the vibration throughout the entire rocket to be magnified which increases the propellant vibration and so on in a feedback loop manner. In the case of the Titan II a team from the Aerospace cooperation led by Sheldon Rubin discovered that the key to overcoming thrust oscillation in a liquid rocket is in cavitation, or the forming of bubbles in the fluid at the inlets of the fuel pumps and the oxidizer. [11] The bubbles from the cavitation lowered the vibration frequency of the fluid inside the feed lines. Along with this cavitation on the Titan II the oxidizer feed lines were each equipped with an accumulator, or a container of gas that is used as a spring to lower the fluid frequency well below the structural frequency, weakening the feedback. A diagram of the accumulators installed on the Titan II are shown in Figure 2.4 [12] [13] [14].

Fig. 2.4: Accumulators Installed on the Titan II to Prevent Against Thrust Oscillation.

From the static ground test data collected from the four segment booster through out the history of the shuttle program, estimates show that the resonant burning amplitudes are 4.3 g's RMS at the Orion capsule. The resonant frequency is approximately 12 Hz [15]. NASA manned spaceflight safety specifications set a crew health vibration limit of less than

0.6 g's RMS in any axis over a 1 minute duration during ascent. Vibrations that reach above 0.6 g's RMS for 1 minute are considered intolerable to humans [16]. At such high vibration levels, as the Ares I produces, organs and the respiratory system can be damage. Subjects exposed to such levels for 1 and 3 minutes reported that they had to exert great force to finish the test, also pain was reported in the thorax, abdomen, and skeletal musculature. Varying effects on blood pressure and respiratory rate were also observed.

The Ares I-X test is part of a flight test program that will include five flight tests of the Orion launch abort system that are scheduled to take place between 2009 and 2011. The integrated testing of the launch vehicle and capsule, dubbed the Orion 1 and Orion 2, is scheduled to take place in 2014. The first live-crew testing is scheduled to take place in 2015, with operational flights to the International Space Station in 2016. Thus we can see that even if this time line is met there will be a four-to five-year gap in which the U.S. will not have access to the International Space Station. During this interim period the U.S. must rely on the Russian Soyuz and Progress spacecraft in order to access the station. Any further delay in the Ares I schedule can dramatically impact the Space Station's science and mission operations.

The main problem lies in the lack of flight data. Currently there is only a four segment booster database consisting from ground testing along with some data from Development Flight Instrumentation, that only flew with Columbia during STS-1 through STS-4, along with some additional flights with the Solid Rocket Booster (SRB) instrumentation on STS-9. Because of this lack in flight data and the immaturity of the Ares design, it is not possible to accurately determine how bad the thrust oscillations will be, and how bad the systems and crew of the Ares and Orion will be impacted. Data gathering via actual flight hardware is scheduled to begin with STS-125, with sensors believed to have been added to Atlantis' boosters, although this is still unconfirmed. During the the Ares I-X flight tests extensive instrumentation is planned to confirm the magnitude of these thrust oscillations.

If the flight data collected from the Ares I-X flights along with the shuttle RSRM flight data verifies the predicted thrust oscillation resonance amplitudes, then mitigations

will be necessary. Possible options that have been proposed to correct the oscillations include de-tuning the stack by modifying the propellant grain, active reaction control system dampening, stiffening the upper stage structure, and adding isolation pads that would sit between the Orion and the J2-X stages. All of these options will undoubtedly take development and testing time, which will inevitably eat into the Orion mass margins and delay the program. Delaying the Constellation program which will also impact the U.S. access to the International Space Station. These delays will only lengthen the gap where the U.S. space program will be depended on the Russian launch vehicle for access to the International Space Station [17].

During a design review that transpired in October of 2008 the review graded the Ares I against ten criteria from NASA's program management handbook. During this review seven of marks were given a "C" or a "D" grade, giving the project an over all of a 2.1, or a low "C". If these issues are not address and resolved soon it is inevitable that the scheduling for the Ares I will be pushed back and the U.S. gap to the International Space Station will only continue to lengthen [18].

Chapter 3

Interim Access to the International Space Station

The Russian rocket Soyuz and the Space Shuttle are the only launch vehicles currently on the market that are capable of delivering a crew to the ISS. Once the Space Shuttle is retired in 2011, the Soyuz will be the only available access to the ISS. Thus until the Ares I is fully operational the United States will be dependent on Russia for manned missions to the Space Station, or in the case that Russia is not able to support the U.S. space program, the United States would not have human access to space at all.

This thesis explores other possibilities for testing and deploying the Orion spacecraft. These options are offered as an interim solution until the Ares launch systems become operational. The proposed options will provide a risk mitigation path. If fruitful, these alternate launch options would allow the operational gap for the ISS missions to be reduced by several years, especially in the case that the current problems extend the current Ares I operational schedule and cause significant time delays in the initial launch date. These options will also give the U.S. an alternative route if the Ares I was not to become operational at all.

3.1 Historical Presidents

There is precedence for this risk mitigation approach. In 1997 the Naval Research Laboratory (NRL) was awarded a contract to construct a space worthy Interim Control Module (ICM) for the International Space Station. This module was intended to be a risk buy down in the case that the Russian ISS-Zvezda service module was destroyed or not launched for an extended period of time. It was derived from a formerly classified Titan Launch Dispenser used to distribute reconnaissance satellites to different orbits. Although the ICM was never flown, it provided schedule insurance and political leverage for the

USA. The ICM is currently in a caretaker status at NRL's Payload Processing Facility in Washington, D.C. If the proposed launch options are fruit full they to may be used as political leverage and schedule insurance for the USA [19].

3.2 International Space Station Overview

The ISS is an internationally developed research facility that is being assembled in Low Earth Orbit (LEO) [19]. The construction of the space station began in 1998 and is scheduled to be completed by 2011. The first resident crew boarded the ISS on November 2, 2000. The ISS has been continuously staffed since then. Sixteen different countries have visited the ISS. The Space Shuttle, Proton, and Soyuz rockets have launched both unpressurized and pressurized modules that make up the International Space Station. The ISS is serviced by several un-crewed vehicles such as the Japanese Space Agency's H-II Transfer Vehicle, the Russian Progress Vehicle, and the European Space Agency's Automated Transfer Vehicle. The ISS lies at a high inclination and at an altitude that lies near the upper limits of what is normally considered to be LEO. The high inclination orbit was mandated when the USA teamed with the Russian Space agency, and the 51.7 degree inclination was the lowest orbit that the Soyuz could reach from the Baikonur Launch complex [20]. To allow Soyuz rendezvous mission, the ISS orbit is maintained at altitudes varying from a minimum of 278 km (173 mi) to a maximum of 425 km (264 mi). At this orbit the ISS constantly loses altitude because of atmospheric drag, and needs to be re-boosted to a higher altitude several times each year. Orbit decay rates vary from day to day, and are heavily influenced by thermoatmospheric density changes due solar activity. Fig. 3.1 shows the changes in mean altitude Z , experienced for a 12-month period beginning August 1, 2008. The station altitude "re-boosts" are clearly visible, as is the significant decay that occurred during the long delay between shuttle missions during early 2009. As the sun enters a period of greater solar activity and atmospheric drag becomes greater, the mean altitude of the ISS will tend towards the lower ends of the nominal range. For the purposes of this study, the mean ISS altitude is defined to be 350 km above mean sea level [21].

Fig. 3.1: International Space Station Mean Orbit Altitude.

Chapter 4

Preliminary Analysis of Available Launch Options

This study limited the list of possible launch vehicle candidates to current commercially available U.S. launchers and upper stages. Due to the size of the Orion spacecraft many of the current smaller launch vehicles on the market would not have the lift capability to deliver the Orion to the desired LEO orbit, thus the initial trade study looked solely at the Evolved Expendable Launch Vehicles (EELV) that are currently available, to assess their capability for delivering the Orion to LEO.

Key in this analysis phase was the selection of launch systems with adequate performance to deliver the required payload mass to the required orbit for the mission. During this trade, manufacturer mission payload charts were used in order to take a “first cut” in reducing the number of candidates to a “short list.” A few of these mission payload charts are shown below; Fig. 4.1 [22] shows the manufacturers payload performance charts for the Atlas V 552 Medium-lift and HLV configurations. Fig. 4.1 plots the Atlas V configurations available payload as a function of the mean orbital altitude for three inclinations available from the Eastern Launch Range, complex SLC-41, at CCAFS. Clearly, the 552 configuration has insufficient payload capacity 17,600 kg (38,800 lbs) to lift the required ISS Orion mass of 20,159.7 kg. Currently, payload charts for the V-HLV configuration are not available for public release; however, Fig. 4.1 shows the proposed baseline capability for the Atlas V configurations, and the stated low inclination lift capacity exceeds 27,000 kg for a 185 km altitude orbit. As with the Delta IV preliminary analysis, these charts do not account for the launch tower. The mass of the extended 5-meter shroud, 5,088 kg (11,214 lbs) budgeted in these graphs, will not be carried during launch. This mass savings partially offsets the LAS mass and will be explored in detail in the Modeling and Simulation section. All things considered, the data presented in Fig. 4.1 are sufficiently positive to allow the Atlas V HLV

to be carried forward as a viable launch option for the CEV [22].

Fig. 4.2 [23] shows the manufacturers payload performance charts for the Delta IV Heavy configuration. These charts plot the available payload as a function of the mean orbital altitude for three common inclinations available from the Eastern Launch Range, complex SLC-37, at Cape Canaveral Air Force Station (CCAFS). Clearly the Delta IV Heavy has excess payload capacity 22,500 kg (49,590 lbs), compared to the estimated ISS Orion mass of 20,159.7 kg. The mass of the Orion escape tower, which will be jettisoned during launch, is not factored into these graphs. Also for the proposed Orion launch, the Delta IV 5-meter payload shroud mass of 3,520 kg (7,758 lbs) budgeted in these graphs will not be carried. This mass savings partially offsets the LAS mass. These graphs present conservative estimates and are clearly not optimized for a specific launch trajectory. It is likely that the real launch vehicle will offer slightly higher payload capacities. Based on the data presented in Fig.4.2, the Delta IV Heavy configuration will be carried forward as a viable launch option for the CEV [23].

Fig. 4.1: Atlas V 500 Medium Series Lift Capability, 350 km ISS Orbit Highlighted.

Fig. 4.2: Delta IV Heavy Lift Capability, 350 km ISS Orbit Highlighted.

The data shown in Fig. 4.1 and Fig. 4.2 was reason enough to add both of these launch vehicles to the “short list.” Remembering that the Orion spacecraft fully loaded with six crew members has a mass of roughly 27,000 kg at this point it is assumed that some off loading of propellant in the service module might be needed in order to reach the International Space Station. Other key requirements that were used in making this “short list” included; sufficient reliability to allow the spacecraft to be human-rated, acceptable lift-off G-loads for human-rating, both static and dynamic, sufficient lift and ΔV for ISS access and return with acceptable propellant margins, brought on-line before 2014 for un-crewed operations and 2015 for crewed launch.

4.1 Evolved Expendable Launch Vehicles

In May 1994 the Department of Defence (DOD) produced a study, known as the Moorman Study, that tackled the matter of increasing costs of hardware associated with the medium and heavy lift launch vehicles [24]. This discussion included the Titan IV and the fact that the rate of production for this launch vehicle seemed to not be very proficient.

The study also included the amount of manpower that was required in manufacturing and operating these systems, and the multiple launch complexes at Cape Canaveral and Vandenberg. Thus the DOD proposed a plan that would alleviate these conditions. This plan was to evolve existing systems into a new family of medium and heavy lift launch vehicles and to accommodate schedule opportunities where several satellite systems were to undergo design changes. This method would put a cost saving method to the problem. In September of that same year Congress provided funds to develop these expendable launch vehicles evolved from existing systems. And it was just a few months later in November when the DOD developed an EELV implementation plan, this plan would reduce the total cost for a medium and heavy lift vehicle space launch. It included a program strategy that would incorporate industrial competition, with the result of a production contract with the goal of maximizing common systems and components that would reduce manufacturing costs and make a quicker production rate, and decrease the number of launch complexes, launch crews and support requirements in an effort to reduce overall operation costs. In August of 1995 Alliant Techsystems (ATK), Boeing Defense and Space Group, Lockheed Martin Technologies, and McDonnell Douglas Aerospace were all awarded 15 month contracts to work on the EELV concept. Ultimately it would be McDonnell Douglas and Lockheed Martin that would receive \$60 million contracts in December of 1996 to perform pre-engineering and manufacturing development studies for the EELV. McDonnell Douglas would later be bought out by Boeing. Lockheed Martin and Boeing would finally begin manufacturing these rockets thus giving us the current EELVs that are used today, the Boeing Delta IV and the Lockheed Martin Atlas V [25].

4.1.1 Atlas V

Lockheed Martin developed the Atlas V series to help meet the demands of government and commercial launch services. The first stage in the Atlas V uses the RD-180 engine along with what is called the common core booster (CCB). Two additional CCB's are added as strap-on boosters in the Atlas V HLV configuration. The CCB contains aluminum isogrid tanks with a diameter of 3.8 m (12.5 ft) These tanks no longer share a common bulkhead

but are now all independent. The upper stage of the Atlas V is the Centaur. The Centaur was the world's first oxygen-hydrogen upper stage when it was developed in the 1960's, and since then the avionics and propulsion systems have been upgraded. The Centaur tanks are pressure-stabilized steel tanks that share a common bulkhead. The Pratt and Whitney RL10 engines are mounted on the bottom of the tanks.

Since its maiden flight in 2001 the Atlas V has had 11 successful flights and only one partial failure flight. The Centaur III stage is 3.1 meters (10 ft) in diameter, 11.68 meters (38.5 ft) long and uses 20,830 kg (45,920 lbs) of propellants. The nominal payload shroud is 4.1 meters (13.45 ft) in diameter; however, an optional extended 5.4-meter (17.7 ft) by 26.4 meter (86.6 ft) payload fairing is available for the 500-series and HLV. The extended shroud weighs 5,088 kg (11,214 lbs). [22] The first four Atlas V launches (21 Aug 2002, 13 May 2003, 17 July 2003, and 17 Dec 2004), were deemed success. On April 14, 2008, an Atlas V Medium+ lifted its heaviest payload to date into orbit, a 14,625-pound (6,634 kg) telecommunications satellite built by Space Systems/Loral. The family of Atlas Launchers is shown in Fig. 4.3 [26] [27].

4.1.2 Delta IV

The Delta series was developed in 1986, following the failures of the Challenger and several Expendable Launch Vehicles (ELV), in order to provide launches for the United States Air Force. The Delta IV heavy is the largest of the Delta series. The Delta IV Heavy was designed for lifting large spacecrafts to LEO, and is also capable of delivering spacecrafts directly to geostationary orbit (GEO), and can be used for earth escape missions. The first stage of the Delta IV Heavy is named the Common Booster Core (CBC). The Rocketdyne RS-68 powers the CBC. The RS-68 engine burns liquid oxygen (LO_2) and liquid hydrogen (LH_2), and is capable of delivering a thrust of 2,891 kN (650,000 lb) with a specific impulse of 410 sec. The RS-68 is capable of throttling back to 60% of its full throttle level. The thrust chamber is a hot isostatic press bonded upgrade of the SSME design. The engine has a gas generator, two turbo pumps, and a regeneratively cooled thrust chamber. Pitch and yaw control are provided through a hydraulically gimballed nozzle and thrust chamber. Roll

Fig. 4.3: Atlas Family of Launch Vehicles.

control for the Delta IV Heavy is provided by the gimbaling of the RS-68 engines on the two strap-on boosters. The second stage of the Delta IV Heavy is powered by the cryogenic RL10B-2 engine from Pratt and Whitney. The RL10B-2 is derived from the flight-proven RL10 family. This engine uses an extendable nozzle that produces a thrust of 110 kN (24,750 lb) with a specific impulse of 462 sec. The engine is gimballed by electromechanical actuators that provide high reliability and also lower both the weight and overall cost of the engine. The RL10B-2 propulsion system and the attitude control system (ACS) use off the shelf flight proven components. During the coast stage of the flight the propellant is managed by directing hydrogen boil-off through aft-facing thrusters to provide settling thrust, and also by the use of the ACS as deemed necessary. During burns the propellant tank remains pressurized by using hydrogen bleed from the engine for the LH_2 tank and helium from the LO_2 tank. An extra helium bottle is added to missions that require more than one restart. The nominal mission duration is 2.3 hrs, but may be increased to over 7 hrs by adding hydrazine bottles and batteries to this stage. Fig. 4.4 [26] shows the family

of Delta IV launchers [23].

Fig. 4.4: Delta IV Family of Launch Vehicles.

4.2 Man-rating the EELV's

The Atlas V and Delta IV were primarily designed for large military payloads or for sizable geostationary communications satellites, but as seen in Fig. 4.1 and Fig. 4.2 both are capable of lifting large spacecrafts to LEO. United Launch Alliance (ULA) currently operates both systems. In 2006, Holguin, performed a study addressing the issues associated with human-rating the Atlas V launch system. Recently, Pulliam performed a similar study for human rating the Delta IV-Heavy configuration. Both reports concluded that the tasks were feasible and affordable [28] [29].

4.3 Preliminary EELV Lift Capability Analysis

In order to further verify the likelihood of the Delta IV and the Atlas V lifting the Crew Exploration Vehicle (CEV) and Service Module (SM) to the ISS, it was deemed necessary to perform a series of “back of the envelope” calculations. A simplified discussion of this

analysis is described in the ensuing paragraphs. These “back of the envelope” calculations were done by calculating the velocity that would be required to reach the ISS orbit, equation 4.1, where ΔV_{north} and ΔV_{east} are the velocities caused by the rotation of the earth and the change in orbital plane from the launch site and can be calculated from equation 4.2 and equation 4.3. ΔV_{drag} accounts for the velocity loss do to aerodynamic drag. During these “back of the envelope” calculations a drag loss of about 16% of the total velocity was assumed. $\Delta V_{gravity}$ is the velocity required to overcome the gravitational field of the earth and can be calculated from equation 4.4 [30].

$$\Delta V_{total} = \sqrt{(\Delta V_{north}^2 + \Delta V_{east}^2) * \left(1 + \frac{\Delta V_{drag}}{\sqrt{\Delta V_{north}^2 + \Delta V_{east}^2}}\right)^2 + \Delta V_{gravity}^2} \quad (4.1)$$

$$\Delta V_{north} = \Delta V_{orbit} * \cos \Psi_{launch} \quad (4.2)$$

$$\Delta V_{east} = \Delta V_{orbit} * \sin \Psi_{launch} - V_{boost} \quad (4.3)$$

$$\Delta V_{gravity} = \sqrt{2 * \frac{\mu * h}{R_e * (R_e + h)}} \quad (4.4)$$

In equation 4.4 μ is the standard gravitational parameter, h is the altitude of the orbit, and R_e is the radius of the earth. A fixed orbit approximation is used and can be calculated from Equation 4.5. lat is the latitude of the launch site and i is the inclination of the desired orbit.

$$\Psi_{launch} = \arcsin \frac{\cos(i)}{\cos(lat)} \quad (4.5)$$

The velocity added to the launch due to the rotation of the Earth is calculated by equation 4.6, where Ω is the mean angular rotation of the Earth’s rotation. The orbital velocity can be calculated from equation 4.7.

$$V_{boost} = R_e * \Omega * \cos(lat) \quad (4.6)$$

$$V_{orbit} = \sqrt{\frac{\mu}{R_e + h}} \quad (4.7)$$

In order for a launch vehicle to deliver a spacecraft to a certain orbit it must be capable of delivering the velocity calculated in Equation 4.1. The maximum velocity that a launch vehicle can achieve given a certain payload can be calculated using Equation 4.8. In Equation 4.8 g_o is the gravitational constant, I_{sp} is the specific impulse of the rocket motor and w_i and w_f are the initial and final weights of the launch vehicle.

$$\Delta V = g_o * I_{sp} * \ln\left(\frac{w_i}{w_f}\right) \quad (4.8)$$

Fig. 4.5 shows the results from the “back of the envelope” calculations for the Delta IV Heavy launch vehicle. This analysis was done for an orbit of 350 km, ISS orbit. According to the calculations done and as shown in Fig. 4.5, the Delta IV Heavy is capable of lifting a spacecraft of just over 24,000 kg. Fig. 4.6 shows the results from the “back of the envelope” calculations for the Atlas V launch vehicle. These results assume an ISS orbit of 350 km. As seen in Fig. 4.6 the Atlas V should be capable of lifting a spacecraft of about 22,000 kg to the ISS orbit. In Fig. 4.5 and Fig. 4.6 “Delta V budget” is the residual velocity from Equation 4.1 and Equation 4.8, the payload mass that causes this value to be zero is the maximum payload capability for the launch vehicle. Results from these “back of the envelope” calculations for the Delta IV and Atlas V agree with what was found in the published payload charts shown in Fig. 4.2 and Fig. 4.1.

Fig. 4.5: Back of the Envelope Calculation Results for the Delta IV Heavy Launch Vehicle.

Fig. 4.6: Back of the Envelope Calculation Results for the Atlas V Launch Vehicle.

Chapter 5

Modeling and Analysis

In order to verify that the launch vehicles from the “short list” are capable of lifting the Orion spacecraft to the ISS, direct simulation was necessary. The direct simulation offers the opportunity to optimize the mission-specific endo-atmospheric portion of the launch trajectory. The optimization process was done using an interactive simulation developed at Utah State University(USU). This simulation contains a graphical user interface (GUI) that gives the user direct in-the-loop control. The simulation allows for the pitch angle to be prescribed at each data frame by direct joy-stick input, a pre-defined set of way points, or by a feedback-control loop. This 4-degree-of-freedom (DOF) interactive simulation allows the user the opportunity to evaluate a wide array of possible maneuvers and trajectories. The GUI also includes real-time displays that allow the user to develop extensive intuition as to the specific mission parameters. This simulation technique was originated at NASA in the 1970’s during the lift body flight test program and played a large role in the facilitation of this analysis. This interactive simulation approach was used as a time-saving measure in lieu of more traditional trajectory optimization tools like the Program to Optimize Simulated Trajectories (POST). One of the major drawbacks of POST is the difficulty of setting up the program, and sensitivity of the final solution to the initial trajectory guess. The interactive simulation also allowed perturbed conditions about the optimal trajectory for Monte-Carlo analysis of expected orbit insertion accuracies [31] [32] [33].

5.1 Launch Simulation

The USU developed launch simulation uses the 3-DOF equations of motion written as a 6 state system of equations to describe the vehicle dynamics, which are described in the satellite coordinate system. These equations of motion are integrated with respect to time

Fig. 5.1: Forces Acting on Spacecraft During Atmospheric Interface.

as an initial value problem. In this system of equations the i_r component points in the radial direction away from the center of the earth and represents the local vertical direction of the spacecraft. The local horizontal direction is represented by the i_v component, which is perpendicular to the radial direction. The i_i component is orthogonal to the other two axes. This coordinate system stays fixed to the spacecraft at all times, and is always perpendicular to the instantaneous orbital plane. The angle of motion relative to the local horizontal is described by the angle γ which represents the flight path angle. The position of the spacecraft relative to the orbital perigee is described by the true anomaly angle ν . I is the inclination of the orbit, and the direction of motion is along the orbital track relative to the Earth's equator. The radial position of the spacecraft relative to the center of the Earth is designated by the symbol r . A derivation of the equations of motion used in the USU simulation can be found in, Schulz, Prado, and Morales [34] and are shown in Appendix B. The forces acting on the spacecraft during atmospheric interface are shown in Fig. 5.1 [35].

During the simulation the instantaneous orbital altitude, geodetic latitude, geocentric longitude, atmospheric properties, Mach number, dynamic pressure, angle-of-attack, and the gravity vector direction are calculated during a transformation of the state vectors from the satellite reference frame to the Earth Fixed WGS-84 coordinate system, this takes place at each data frame during simulation. The oblate earth model accounts for gravity out to the

J_2 gravitational harmonic. A constant earth rotation rate is assumed. The simulation also allows the user to select the atmospheric model, which the 1977 US standard atmosphere or the 1999 Global Reference Atmospheric Model (GRAM-99) can be used. The spacecraft altitude is approximated as the difference between the spacecraft radius vector and the Earth radius at the local geodetic latitude [36] [37] [38].

The 4-DOF simulation allows the vehicle pitch angle to be prescribed or controlled independently of the equations of state. In this simulation the pitch angle is defined as the angle between the vehicle longitudinal axis and the instantaneous horizontal plane. Pitch angle is designated by the symbol θ . The pitch angle is described in more detail in Section 5.4. The simulation gives the user in the loop control allowing the user to prescribe these way points. The pitch angle can also be controlled by a pitch profile optimizer, or external inputs prescribed from a user-controlled video game joystick. Increments to the prescribed nominal pitch angle can also be modified in real-time by a “pilot” using the joystick input with user-selectable input gains. The pitch profile optimizer will be described in more detail later in this section. As mentioned earlier, user adaptable displays are available to the “pilot” and serve as aids for flying the desired trajectories. Snap shots of the simulations front panel are shown in Fig. 5.2, Fig. 5.3, Fig. 5.4.

The Aerodynamic lift and drag forces act both in the direction of the vehicles motion and perpendicular to the vehicles motion. In the simulation the aerodynamic forces F_{lift} and F_{drag} , are calculated from a table lookup of the trim lift and drag coefficients expressed as a function of angle of attack and Mach number. A “hold last values” approach is used when the angles-of-attack in the lookup tables trimmed are exceeded. These aerodynamic coefficients are calculated using MissileDATCOM. A further discussion of MissileDATCOM and the calculation of the aerodynamic coefficients will be discussed in more detail in Section 5.6. The aerodynamic angle of attack is calculated by taking the difference between the the pitch angle and the local flight path angle. The calculation of the aerodynamic forces will be discussed further later in this chapter.

A key part for this analysis is the development of thrust models that are derived for

Fig. 5.2: A Snap Shot of the Launch Simulation Front Panel.

Fig. 5.3: A Snap Shot of the Launch Simulation Front Panel.

Fig. 5.4: A Snap Shot of the Launch Simulation Front Panel.

each of the stages, derived from the numerous stage components. The simulation allows for throttling of the engine, this feature was used to limit the g-loading during launch in order to meet NASA’s requirements for g-loading in a manned launch vehicle. The interstage times of the simulation and the zero-g coast phases for the trajectory were simulated by setting the throttling to zero. Further discussion of the engine models is discussed in the following section.

5.2 Evolved Expendable Launch Vehicle Propulsion Stage Models

The engine mass flow, nozzle exit velocity, and nozzle exit pressure were modeled using one dimensional De-Laval flow equations to enable a thrust calculation as a function of local altitude. A derivation of the thrust and rocket equation is presented in Appendix A. The data for each of the engines was collected from a variety of public domain sources. The derived propulsion models account for the effects of changing ambient pressure, exit cone angle momentum losses, and allow for mass flow-based engine throttling. The engine and stage properties as modeled in the simulation for the Delta IV Heavy are presented in Table 5.1. Table 5.2 presents the engine and stage properties for the Atlas V-HLV configuration [30] [39] [40].

The equilibrium gas-chemistry code Chemical Equilibrium with Applications (CEA) was used to model the combustion products based on mean properties for the specified propellants. The CEA code was developed at NASA Glenn Research Center, and has been successfully applied for the analysis of rocket combustion, detonation, and flow across non-adiabatic shockwaves. The code posits chemical reactions across the shockwave and then minimizes the Gibbs free energy in order to reach thermodynamic and transport properties at chemical equilibrium. The CEA code has extensive internal libraries for gas thermodynamic and transport properties including standard and non-standard temperature and pressure conditions. The CEA-derived, thermo-chemical data were adjusted as required to give consistency between the specified engine performance parameters and one dimensional thrust models.

Table 5.1: Delta IV Heavy Propulsion Stage Properties

Simulation Element	First Stage (Core + 2 Boosters)	Second Stage
Engine	RS-68	RL 10B-2
Number of Engines in Stage	3	1
Length, m	40.8	12.1
Diameter m	5.1	5.1
Dry Mass, kg	26,800 (<i>each</i>), 80,400 (<i>total</i>)	3,510
Wet Mass, kg	226,400 (<i>each</i>), 679,200 (<i>total</i>)	29,155
Propellant Mass, kg	199,660 (<i>each</i>), 598,800 (<i>total</i>)	25,645
Propellant Mass Fraction	0.8816	0.880
Propellant	LOX/LH_2	LOX/LH_2
Mixture Ratio	6	5.65
Thrust (vacuum), kN	3,315 (<i>each</i>), 9,945 (<i>total</i>)	110.1
Thrust (sea level), kN	2,890 (<i>each</i>), 8,670 (<i>total</i>)	–
Exit Area, m^2	4.194 (<i>each</i>), 12.583 (<i>total</i>)	4.5452
Expansion Ratio	21.5	285
Throat Area, m^2	0.1951 (<i>each</i>), 0.58251 (<i>total</i>)	0.0630
Combustor Pressure, MPa	9.72	3.21
Combustor Temperature, K	3600	3210
I_{sp} (vacuum), sec	420	462.4
Nozzle Exit Angle, deg	29.807	0
Burn Time, sec	248	1056.2
Ratio of Specific Heats	1.160	1.150
Molecular Weight	13.138	15.2955
Mass Flow, kg/sec	804.84 (<i>each</i>), 2,414.53 (<i>total</i>)	24.28
Nozzle Exit Mach Number	3.641	5.1273
Nozzle Exit Pressure, kPa	51.456	0.7589
Nozzle Exit Velocity, m/sec	3,850.65	4,389.43
Nozzle Exit Temperature, K	1,747.2	1172.5

Table 5.2: Atlas V HLV Heavy Propulsion Stage Properties

Simulation Element	First Stage (Core + 2 Boosters)	Second Stage
Engine	RD-180	RL 10A-4-2
Number of Engines in Stage	3	1
Length, m	32.5	11.68
Diameter m	12.5	3.05
Dry Mass, kg	18,800 (<i>each</i>), 56,400 (<i>total</i>)	2,130
Wet Mass, kg	276,500 (<i>each</i>), 829,500 (<i>total</i>)	22,960
Propellant Mass, kg	257,700 (<i>each</i>), 773,100 (<i>total</i>)	20,830
Propellant Mass Fraction	0.932	0.907
Propellant	$LOX/RP - 1$	LOX/LH_2
Mixture Ratio	2.27	5.85
Thrust (vacuum), kN	4,152 (<i>each</i>), 12,456 (<i>total</i>)	198.4
Thrust (sea level), kN	3,827 (<i>each</i>), 11,481 (<i>total</i>)	–
Exit Area, m^2	3.2085 (<i>each</i>), 9.6225 (<i>total</i>)	2.384
Expansion Ratio	36.87	85
Throat Area, m^2	0.0870 (<i>each</i>), 0.261 (<i>total</i>)	0.02805
Combustor Pressure, MPa	25.76	3.21
Combustor Temperature, K	3,020	3530
I_{sp} (vacuum), sec	337.8	450.5
Nozzle Exit Angle, deg	28.8	0
Burn Time, sec	205.6	463.8
Ratio of Specific Heats	1.1162	1.1449
Molecular Weight	24.24	16.32
Mass Flow, kg/sec	1,253.3 (<i>each</i>), 3,759.9 (<i>total</i>)	44.916
Nozzle Exit Mach Number	3.753	4.376
Nozzle Exit Pressure, kPa	82.557	3.33
Nozzle Exit Velocity, m/sec	3,101.7	4,062.6
Nozzle Exit Temperature, K	1,594.9	1,479.5

5.3 Orion Propulsion Model

The main propulsion system for the Orion spacecraft is derived from the Delta II second stage, the Aerojet AJ-10 rocket engine. This engine is powered by nitrogen tetra-oxide (N_2O_4) and mono-methyl hydrazine (MMH). These propellents only produce a mediocre performance but were selected as a safety feature. The Orion spacecraft has start and restart capability, and produces a vacuum thrust of 33 kN (7,500 lb). For LEO missions the service module propulsion system is primarily designed to perform maneuvers during the docking process with the ISS. During a lunar mission, the SM will be used to correct the trans-lunar trajectory, and to be able to leave the lunar orbit for the return trip to Earth. The ISS version of the SM provides 690 m/sec (2,270 ft/sec) of on-orbit maneuvering ΔV . The lunar version of the SM provides 1,400 m/sec (4,590 ft/sec) of ΔV . The engine and stage properties for the Orion spacecraft are shown in Table 5.3.

5.4 Throttled Propulsion Model Equations

From the stage data presented in Table 5.1,

$$F_{thrust}(t) = T(t) \times \vartheta \times (\dot{m} \times U_{exit}) + (P_{exit} - P_{\infty}) \times A_{exit} \quad (5.1)$$

In 5.1 $F_{thrust}(t)$ is the time variable thrust of the engine T_t is the throttle (variable over the prescribed range for the engine), P_{exit} is the exit pressure, P_{∞} is the ambient operating pressure, U_{exit} is the nozzle exit velocity, and A_{exit} is the nozzle exit area. ϑ is the corrector to take into account the nozzle exit flow angle. This correction angle is calculated from equation 5.2 where η is the nozzle exit angle.

$$\vartheta = \left(\frac{1 + \cos \eta}{2} \right) \quad (5.2)$$

The throttle mass depletion equation is shown in 5.3

$$M_{stage}(t) = M_o - \int (T(t) \times \dot{m}) dt \quad (5.3)$$

Table 5.3: Orion Propulsion Stage Properties

Simulation Element	Orion/SM Stage Data
Engine	AJ-10 (Aerojet)
Number of Engines in Stage	1
Length, m	4.78
Diameter m	5.0
Dry Mass, kg	5,144.7 (<i>ISS</i>), 4,430.6 (<i>Lunar</i>)
Wet Mass, kg	8,810.3 (<i>ISS</i>), 12,340.4 (<i>Lunar</i>)
Propellant Mass, kg	3665.6 (<i>ISS</i>), 7909.8 (<i>Lunar</i>)
Propellant Mass Fraction	0.426 (<i>ISS</i>) 0.641 (<i>Lunar</i>)
Propellant	N_2O_4/MMH
Mixture Ratio	1.7
Thrust (vacuum), kN	33.4
Exit Area, m^2	0.6137
Expansion Ratio	85
Throat Area, m^2	0.00722
Combustor Pressure, MPa	2.5
Combustor Temperature, K	3,300
I_{sp} (vacuum), sec	319.97
Nozzle Exit Angle, deg	0
Burn Time, sec	344.4
Ratio of Specific Heats	1.278
Molecular Weight	21.891
Mass Flow, kg/sec	10.644
Nozzle Exit Mach Number	5.424
Nozzle Exit Pressure, kPa	1.410
Nozzle Exit Velocity, m/sec	3,056.6
Nozzle Exit Temperature, K	654.3

In equation 5.1 and equation 5.3 $\dot{m}$ is the nominal full throttle propellant mass flow, M_o is the initial gross stage mass, and $M_{stage}(t)$ is the current stage mass depleted as a function of time. The throttle operates between 0 and 1.0, and as mentioned earlier, the ambient pressure is calculated as a function of altitude using the 1976 US Standard atmosphere or GRAMM-99 models. The throttling term is included in the model to insure that g-levels do not exceed the crew safety standard, as prescribed by NASA, during ascent.

5.5 Orion De-Orbit ΔV and Propellant Reserve Requirements

A necessary constraint for this mission was the requirement that the Orion spacecraft must have enough propellant remaining at the end of the mission to de-orbit. The de-orbit process assumes an initial circular orbit of 120 km (400,000 ft) atmospheric interface and an impulsive de-orbit burn. Figure. 5.5 shows the amount of ΔV required to de-orbit from a given orbital altitude and the entry interface flight path angle. From Fig. 5.5 we see that for the ISS orbit of 350 km (218 mi) and a re-entry flight path angle of -1.5 degrees a ΔV of about 105 m/sec (345 ft/sec) is required to de-orbit from the ISS. The algorithm used to calculate the required de-orbit ΔV is based on the classical analysis presented by Milstead [41].

From Table 1.1 we see that the dry mass for the Orion CM and SM is 14,854 kg (32,739 lbs). This mass includes all on-board consumable propellant except for the propellant available for the main engine. From the data in Table 1.1 and the rocket equation (5.4) we can calculate the amount of propellant that will need be reserved in order to perform the de-orbiting of the spacecraft. This amount will be deducted from the available propellant mass and added to the dry mass budget for all simulations in this analysis [42].

$$M_{prop} = M_{dry} \times (e^{\frac{\Delta V}{g_o \times I_{sp}}} - 1) \quad (5.4)$$

In equation 5.4 M_{dry} is the spacecraft's dry mass, g_o is the Earth's gravitational constant, and I_{sp} is the specific impulse of the rocket motor.

Fig. 5.5: ΔV Required for Orion De-Orbit From ISS.

5.6 Launch Vehicle Aerodynamics

The aerodynamics of most commercially operational launch platforms are usually highly guarded, usually not readily available to the general public. After an extensive literature search the aerodynamic coefficients were not available for the Delta IV and Atlas V. Since the aerodynamic data was not available it was necessary to generate the aerodynamic coefficients independently in order to perform the analysis for this study. The aerodynamic coefficients are calculated using MissileDATCOM. MissileDATCOM is a simulation developed by the Air Force Research Laboratory (AFRL) used to calculate the aerodynamic coefficients for launch vehicles and missiles. MissileDATCOM is a semi-empirical analysis. This code is the de-facto aerospace and defense industry standard for modeling rocket and missile aerodynamics and has been in continual development for over twenty years. MissileDATCOM predictions are valid for both subsonic and supersonic conditions, and extensive studies have demonstrated that the predictions are accurate for a variety of configurations at moderate angle of attack. MissileDATCOM has been traditionally supplied free of charge by the USAF to American defense contractors. The program is written in Fortran 90 and both compiled and source code versions are available. A detailed users guide is available [43] [44] [45].

MissileDATCOM calculates aerodynamic forces, moment coefficients, and non-dimensional vehicle stability derivatives, but for this analysis only the lift and drag coefficients were needed. The calculations performed by MissileDATCOM are done in the total angle of attack/aerodynamic roll coordinate system, commonly known as the “aeroballistic axis system.” MissileDATCOM uses a component build-up superposition approach for calculating the forces and moments. With this technique, estimates are made for isolated body and fin panels, semi-empirical interference effects are then applied, and the results are summed to give overall aerodynamic coefficient, moment, etc. The inputs for MissileDATCOM are Mach number, angle of attack, Reynolds number, altitude, and the launch vehicle’s dimensions in order to calculate the aerodynamics of the vehicle, Fig. 5.6 shows the possible body geometry inputs that MissileDATCOM will accept. Optionally, flat-plate models, adjusted

for compressibility, can be used to account for skin friction effects [46].

Fig. 5.6: Dimension Accepted by MissileDATCOM.

The lift sectional coefficient (C_L) and the drag sectional coefficient (C_D) for the Delta IV Heavy are graphed in Fig. 5.7 and Fig. 5.7. And the lift and drag coefficients for the Atlas V are shown in Fig. 5.9 and Fig. 5.10. These plots show the lift and drag coefficients as a function of Mach number and angle of attack. The reference area for the Delta IV is 52.04 m^2 (560 ft^2), and the reference area for the Atlas V is 34.05 m^2 (367 ft^2). Only the “engine-on” launch configuration aerodynamics were accounted for. The assumption here is that the vehicle has reached a sufficiently high altitude by the time that the first stage has burned out that rocket base-drag and stage two aerodynamics can be ignored. As expected for both configurations, when the angle of attack rises beyond 10 degrees a severe drag penalty is encountered.

5.7 Pitch Profile Optimization

The launch vehicle simulation allows the option of running the simulation with an endo-atmospheric pitch profile optimization routine. During this analysis initially the pitch profile was adjusted by user input, then finalized using the optimization routine. An “apogee

Fig. 5.7: Delta IV Heavy Lift Coefficient (C_L) Plotted as a Function of Mach Number and Angle of Attack.

Fig. 5.8: Delta IV Heavy Drag Coefficient (C_D) Plotted as a Function of Mach Number and Angle of Attack.

Fig. 5.9: Atlas V Lift Coefficient (C_L) Plotted as a Function of Mach Number and Angle of Attack.

Fig. 5.10: Atlas V Drag Coefficient (C_D) Plotted as a Function of Mach Number and Angle of Attack.

targeting” method is used to deliver the payload to the burnout of the second stage to an orbit with an apogee that is just below the desired final altitude and maximum energy level. This analysis assumes that after the second stage has burned out that the launch vehicle is at a high enough altitude the atmospheric effects are small enough that they can be disregarded during the simulation. The most effective way to circularize the final orbit the trim burn is done at the orbit apogee in the direction of the orbit velocity.

At first glance one might think that the most effective flight path would be to fly the launch vehicle using a ballistic profile because this offers the least amount of drag. But the ballistic profile produces a very lofted trajectory to the ISS orbit altitude with little forward velocity.

The ballistic profile does not offer enough centripetal acceleration to keep the vehicle in orbit. In order to gain the amount of centripetal acceleration required to keep the launch vehicle in orbit a gravity assist maneuver is performed soon after the initial launch. The optimization process is begun with the ballistic profile and then systematically ”turning the corner” during successive launch iterations. An exponential decay model is used with the ballistic pitch angle to model the optimal pitch profile. This is done using the two parameter model shown in equation 5.5, where $\theta(t)$ is the optimal pitch profile, $\theta(t)_{ballistic}$ is the ballistic profile, λ is a pitch decay slope parameter, n is a scaling exponent, and t is time from launch. The exponential decay model not only produces a smooth turn but also ensures a small angle of attack during the endo-atmospheric burn. The small angle of attack not only keeps the aerodynamic drag losses to a minimum but it also reduces axial loading on the launch vehicle. With a small angle of attack the ballistic pitch angle can be approximated by the inertial flight path angle. equation 5.6

$$\theta(t) = \theta(t)_{ballistic} \times e^{-\lambda \times t^n} \quad (5.5)$$

$$\gamma = \arctan \frac{V_r}{V_v} \times e^{-\lambda \times t^n} \quad (5.6)$$

In equation 5.6 V_r and V_v are the local vertical and local horizontal velocities respectively. Parameter iterations are performed as two loops with n number of iteration in the outer loop and λ being driven to a value that satisfies the apogee constraint in the inner loop. These parameters are fixed at the beginning of the simulation run. The pitch profile parameters are iterated between simulation runs to satisfy the apogee constraint and maximize orbital energy. This is done using a modified Newton-Raphson method shown in equation 5.7.

$$\lambda^{j+1} = \lambda^j - \frac{R_a^j - R_{target}}{R_a^j} \times \lambda^j \quad (5.7)$$

R_a is the orbits apogee at the burnout of the second stage, R_{target} is the target orbit radius, and j is the iteration index. Each of the inner loop runs represents a complete run of the endo-atmospheric portion of simulation [47].

In order to save computation time the simulation uses an *ad hoc* method in lieu of more traditional lagrange multiplier method. This alternate method is feasible because the second stage burnout apogee is constrained, maximizing the vehicles orbital energy is equivalent to maximizing the second stage burnout perigee radius. The parameter pair that gives the largest second stage burnout perigee radius, orbital energy, and satisfies the apogee constraint provides the optimal pitch profile [48].

Chapter 6

Results and Discussion

The preliminary analysis for this study included evaluations of several launch profiles and EELV configuration, including options where the CEV/Orion was delivered to orbit as an inert payload with the launch vehicle providing all of the required impulse, and options where a fraction of the available ΔV from the Orion SM was used to insert the spacecraft into its final orbit and perform the final circularization trim burn. The most promising options are included a continuation. These options include two configurations for the Delta IV Heavy and one configuration for the Atlas V HLV. Each configuration is capable of reaching the ISS orbit with sufficient propellant margin in order for the Orion spacecraft to be de-orbited as discussed earlier. This chapter will provide mission time lines and present detailed launch trajectories for these selected configurations. These configurations are shown in Fig. 6.1. The first configuration, the Atlas V HLV requires significant impulse from the Orion service module in order to reach the final International Space Station orbit, Because propellant from the CM/SM is used this configuration is referred to as “CM/SM Active.” The second option, the Delta IV Heavy, launches the Orion as a passive payload and requires no impulsive maneuvering from the service module in order to reach the International Space Station orbit. Since propellant from the CM/SM is required for this option it will be referred to as the “CMV/SM Passive” configuration. The third configuration, the Delta IV Heavy with three common core boosters as the first stage, and the Orion spacecraft acting as the second stage, requires significant impulse from Orion’s service module engine to achieve the International Space Station orbit, because this configuration uses propellant from the SM this option is deemed the Delta IV “CM/SM Active” configuration.

Fig. 6.1: Preferred EELV Configurations for the ISS Mission.

6.1 Atlas V HLV Configuration

In this section the launch configuration for the Atlas V HLV is presented. This configuration is shown as the first configuration in Fig. 6.1. In this configuration the payload is replaced with the CM/SM and the first and second stages of the Atlas V HLV are used to lift the Orion spacecraft to the ISS orbit. This configuration assumes that the interstage adapter for the Orion spacecraft has been adjusted in order to be capable of interfacing with the Centaur III upper stage. During this configuration the Atlas V first and second stage are fully loaded and are burned to depletion during the mission. Also during this launch option the CM/SM is loaded with 3,666 kg of propellant. This propellant is used for the final orbit insertion and trim burn of the ISS orbit, as stated earlier because propellant from the CM/SM is used this configuration is referred to as “CM/SM Active.” The concept of operations (CONOPS) for this configuration is shown in Fig. 6.2 The time line for this mission from launch to orbit insertion is also shown in Fig.6.2. This time line does not include the possible phasing orbit necessary to intercept the ISS.

Fig. 6.3 shows the pitch angle for the optimized trajectory for this configuration, while Fig. 6.4 plots the altitude versus time, Fig. 6.6 shows the velocity, and Fig. 6.5 shows the accelerations of the engines. The time scale on these plots is logarithmic to better show the features early in the launch. As discussed earlier the target orbit for this profile was to reach an apogee of 350 km at stage two burnout. Also plotted in Fig. 6.3 is the ballistic or minimum drag profile. As seen in the figure the ballistic profile and the optimized profile differ greatly. At the time of the first stage burn out the ballistic profile is at 35 degrees where as the optimal profile is at 15 degrees, by turning the corner earlier in the launch centripetal acceleration supports the weight of the vehicle, and thus all of the propulsive thrust can be used to accelerate the vehicle, getting a greater performance out of the vehicle.

In Fig. 6.5 and Fig. 6.6 the different stages are evident, in these plots each of the spikes or peaks is where one stage reaches burnout. The acceleration plot, Fig. 6.5, is labeled with the different launch events. The acceleration plot also overlays the throttling for the vehicle. At 145 seconds into the flight the launch vehicle begins to throttle back linearly

Fig. 6.2: Atlas V HLV Launch Concept of Operations and Timeline, CM/SM Active.

Fig. 6.3: Pitch Angle for the Atlas V Active Configuration.

Fig. 6.4: Altitude for the Atlas V Active Configuration.

Fig. 6.5: Acceleration for the Atlas V Active Configuration.

Fig. 6.6: Velocity for the Atlas V Active Configuration.

with time from 100 percent to 53 percent. This was done to meet the NASA crew safety requirements. The requirement is that a crew sensed acceleration can not exceed 7.0 g's. The vehicle begins to throttle back once the accelerations surpass 4.0 g's. This throttling slope was selected interactively to insure that the 7-g limit was not exceeded. Also shown in Fig. 6.5 is the long coast phase, also prescribed by the user. In this configuration the coast from the second stage burnout to the CM/SM ignition is the time that is required for the spacecraft to reach the orbit apogee at 350 km. During this period the vehicle is in micro-gravity and is falling towards the orbit apogee. The final burn circularizes and trims the orbit. The final orbit eccentricity for the optimal simulation run is 0.0004 [49].

The Atlas V active configuration is capable of reaching the ISS orbit with only having used approximately 77 percent of the CM/SM propellant, 840 kg of propellant remains. Since only 500 kg of propellant is required for de-orbit and reentry, this is more than the required amount to de-orbit plus perform any orbit maneuvers that may be required during docking with the ISS. Any residual propellant can also be used to re-boost the ISS after docking. But this remaining propellant is insufficient to achieve the ISS orbit without the first stage CCB.

6.2 Delta IV Heavy CM/SM Passive Configuration

The second configuration in Fig. 6.1 is designated the Delta IV CEV/SM Passive configuration. In this configuration the payload for the Delta IV is replaced by the CM/SM. The configuration assumes that the Orion spacecraft adapter has been modified to interface with the Delta IV upper stage. For this configuration the first and second stages are fully loaded and the propellant is burned to completion during the ascent. This configuration is capable of delivering a fully loaded Orion spacecraft, none of the 3,666 kg of propellant is used to reach the ISS orbit, thus this option is called the "CEV/SM Passive" configuration.

In this configuration the second stage has two burns. The first of these burns is used to reach the desired final ISS orbit while the second is to circularize the final orbit. Since the second burn of the second stage is only used to trim the final orbit the optimized pitch profile is only used until the second stage is cut off after its first burn. The CONOPS

Fig. 6.7: Delta IV Heavy Launch Concept of Operations and Timeline, CM/SM Passive.

and time line for this mission are shown from launch until orbit insertion in Fig. 6.7. As mentioned previously the time line presented does not include the phasing orbit that might be necessary to intercept the ISS.

Fig. 6.8 shows the pitch angle for the optimized trajectory for the Delta IV Heavy “Passive” configuration, Fig. 6.10 plots the altitude versus time, Fig.6.9 shows the velocity, and Fig.6.11 shows the accelerations of the engines. As mentioned for this configuration the endo-atmospheric optimization routine ended after the second stage was turned off for the first time, Fig. 6.11 shows that this happens at about 984 seconds. After the optimized trajectory is terminated it is manually adjusted to zero for the circularization and trim

Fig. 6.8: Optimized Pitch Profile the Delta IV Passive CM/SM Configuration.

burn. As we can see from Fig. 6.10 the final orbit is almost reached during the second burn. Also in Fig. 6.11 we can see that the Keplerian coast phase for this trajectory is much shorter than that of the Atlas V shown in Fig. 6.5. During the first burn of the second stage, 23,300 kg of propellant is burned. In the second burn of the second stage an additional 2,060 kg of propellant is burned these two burns combined account for 98.5 percent of the total second stage. This leaves the CM/SM fully loaded as it is delivered to the ISS, it has sufficient propellant for on orbit operations, including reboosting of the ISS. This option could prove valuable in the case that Russian Progress or the Soyuz spacecraft are no longer available for ISS station keeping and orbit maintenance. This also leaves room to believe that a configuration similar to this is capable of being used for lunar missions, where the SM propellant is used to place the CEV in a translunar orbit.

6.3 Delta IV Heavy CM/SM Active Configuration

From the previous section we see that the Delta IV Heavy is more than capable of lifting the Orion spacecraft to the ISS orbit. This section explores the option of removing the second stage of the Delta IV from the configuration and uses strictly the CCB's and

Fig. 6.9: Velocity Profile for the Delta IV Passive CM/SM Configuration.

Fig. 6.10: Launch Altitude for the Delta IV Passive CM/SM Configuration.

Fig. 6.11: Acceleration Profile for the Delta IV Passive CM/SM Configuration.

the SM propellant to reach the ISS orbit. In this configuration the CM/SM configuration used is the fully loaded configuration that is designed to be used for lunar missions. The main drive in the success of this mission is that this option, if feasible, has the distinct advantage of simplifying the launch stack, which reduces the potential for failure. Also this eliminates the need to man rate the second stage. Since only the Common-Core Booster needs to be human-rated for this configuration, this will potentially cut initial development costs and also shorten the time line for making this system fully operational. Because this configuration uses propellant from the SM this option is deemed the Delta IV “CM/SM Active” configuration.

As discussed the second stage is removed in this configuration, and the CM/SM is mounted to the first stage using the spacecraft adapter. Also as discussed previously a fully loaded CM/SM lunar configuration is used, in this configuration the SM contains 7,910 kg of propellant. In order to reach the ISS orbit with this configuration a large portion of the SM’s propellant is used during launch, but a sufficient amount is reserved for de-orbit and on-orbit maneuvering. The SM is fired twice during launch, the first burn to raise the orbit apogee to the ISS altitude and the second to circularize and trim the final orbit. As

Fig. 6.12: Delta IV Heavy Concept of Operations and Timeline, CM/SM Active Configuration.

mentioned previously the time line presented does not include the phasing orbit that might be necessary to intercept the ISS. The CONOPS and launch time line for this configuration are shown in Fig. 6.12.

Fig. 6.13 shows the pitch angle for the optimized trajectory for the Delta IV Heavy “Active” configuration, Fig. 6.14 plots the altitude versus time, Fig. 6.15 shows the velocity, and Fig. 6.16 shows the accelerations of the engines. For this configuration the endo-atmospheric optimization routine ended after the second stage (SM) was turned off for the first time, Fig. 6.16 shows that this happens at about 990 seconds. After the optimized trajectory is terminated it is manually adjusted to zero for the circularization and trim

Fig. 6.13: Optimized Pitch Profile for the Delta IV Heavy Active Configuration.

burn. Due to the fact that the original Delta IV Heavy second stage is removed in this configuration less mass lies atop the CCB's thus causing greater gravity loads on the vehicle. Thus more down throttling was required for this configuration in order to meet the NASA requirement of gravity loads not exceeding 7.0 g's. This down throttle also causes the first stage to burn longer, this is also notable in Fig. 6.16 and Fig. 6.15. Also from Fig. 6.16 we see a very long Keplerian coast period between the first and second burn of the SM. This is due to the low thrust of the SM engine, the AJ-10. When the optimizer finally reaches an orbit with an apogee of 350 km altitude, the orbit is nearly circular and the spacecraft is just past the apogee of the orbit, thus the spacecraft must wait for almost one complete orbit before it performs its final circularization and trim burn.

During the first and second burn of the CM/SM 7305 kg of propellant is used, leaving 604 kg of propellant after the final trim burn. This amount is more than enough to de-orbit the spacecraft, recall that de-orbiting the Orion spacecraft requires 505 kg of propellant. This however does only leave 99 kg of propellant for any orbital maneuvering that might be necessary, but is probably deemed acceptable since the reaction control maneuvering propellant is assumed to have full capacity at this point.

Fig. 6.14: Plot Showing the Launch Altitude for the Delta IV Heavy Active Configuration.

Fig. 6.15: Plot of the Launch Velocity Versus Time for the Delta IV Heavy Active Configuration.

Fig. 6.16: Plot of the Launch Acceleration Versus Time for the Delta IV Heavy Active Configuration.

6.4 Launch Analysis Summary

A Summary of the Analysis discussed in this paper is shown in Table. 6.1. Recall that a ΔV of 105 m/sec (378 ft/sec) is required to de orbit the Orion spacecraft. Table. 6.1 shows that all three of the systems are well above this requirement. Notice that the Delta IV Heavy Passive configuration leaves the most residual propellant, which is to be expected since none of the SM fuel is used during launch. Also it must be stated that the orbit eccentricity shown in Table. 6.1 is a bi-product of the simulation resolution. Higher fidelity simulations with lower time intervals will result in a more precise final orbit eccentricity. The Gross Take-off Mass (GTOM) includes the estimated weights of interstage-adapters, and the LAS system [25].

Table 6.1: Simulation Results Summary for the Three Proposed EELV-Derived Configurations.

Configuration	Atlas V HLV	Delta IV Passive	Delta IV Active
GTOM, kg	883,041	741,959	710,989
Stage-1 Propellant Burned, kg	773,100	598,800	598,800
Stage-2 Propellant Burned, kg	20,830	25,260	–
CM/SM Propellant Burned, kg	2,826	0	7,305
On-Orbit Residual $\Delta V m/sec$	173	692	125
e_{final}	0.0004	0.00017	0.00027
Z_{final} , km	349.94	350.02	349.94

Chapter 7

Reliability Assessment

This analysis did not include a full reliability assessment, but for completeness in this discussion reliability developed by Maggio and Hall are summarized and presented here. The Delta IV Heavy Passive and Delta IV Heavy Active configurations were permutations directly assessed by this team. The data presented for the Delta IV will be extrapolated to give a risk assessment for the Atlas V HLV Active configuration. Discussion will include a loss of mission (LOM) reliability assessment for the launch vehicles, followed by a discussion on loss of crew (LOC) assessment. The LOC reliability assessment take into account the reliability numbers for the Orion LAS system [50].

7.1 Loss of Mission Assessment

The reliability assessment for the Delta IV Heavy Passive configuration is presented in Table 7.1 and Table 7.2. These numbers represent the probability that a mission will be lost due to critical failures, failures due to the first stage being unable to start are not considered. This is because it is assumed that if one of the CCB's fails to start the mission engines will be shut down and the mission will be aborted. Also in this assessment failures of auxiliary power systems, thrust vectoring, throttle control, main power systems, and stage separation are not included. As stated previously in this configuration the Orion is considered to be a passive payload thus the risk assessment is from first stage ignition to the second stage shutoff. The Table 7.1 assessment considers the reliability of the baseline nonhuman rated Delta IV configuration. Table 7.2 presents reliability numbers for the same configuration, only now the components have been human-rated. The reliability numbers for each of the individual components of the launch vehicle was taken directly form Maggio and Hall. As can be expected human rating the components decreases the LOM possibility from one

Table 7.1: Loss of Mission Reliability Assessment for Delta IV Heavy, CEV/SM Passive, Baseline EELV Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 775	1 in 1,204	1 in 357	1 in 357
Strap-CCB	–	1 in 430	1 in 711	1 in 222	1 in 222
RL-10 B-2 Engine	1 in 1,406	1 in 1,406	1 in 3,170	1 in 1,977	1 in 399
Vehicle Total					1 in 102

Table 7.2: Loss of Mission Reliability Assessment for Delta IV Heavy, CEV/SM Passive, Human-Rated Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 1,293	1 in 2,040	1 in 2,668	1 in 610
Strap-CCB	–	1 in 656	1 in 1,093	1 in 2,352	1 in 349
RL-10 B-2 Engine	1 in 2,564	1 in 1,815	1 in 5,593	1 in 3,558	1 in 714
Vehicle Total					1 in 169

failure in every 102 launches to one in every 169 launches, a 40 percent risk reduction.

The reliability assessment for the Delta IV Heavy Active configuration is presented in Table 7.3 and Table 7.4. Remember that in this configuration the Delta IV second stage has been removed and the SM is used as the second stage thus the second stage failure numbers are not included in these tables, and the CM/SM propulsion system failure numbers are presented instead. Maggio and Hall only considered this option for human-rated components, but an additional table using baseline EELV values has been added here for completeness. Because of its simple hypergolic ignition system design the CM/SM is considered to be extremely reliable. Since the CM/SM is being used as the second stage in this configuration and is now an active component of the configuration the probability of failure must be considered in these calculations. The reliability of this modified configuration is only marginally better than the baseline Delta IV Heavy with passive payload. But notice that the overall system for the human-rated configuration is significantly greater. The possibility that the mission will be lost has been reduced by almost twice that of the baseline Delta IV configuration, and approximately 16 percent from the human-rated Delta IV Heavy Passive option presented above. This is significant because as mentioned earlier this may potentially cut initial development costs and cause this configuration to be fully

Table 7.3: Loss of Mission Reliability Assessment for Delta IV, 3 x CCB, CEV/SM Active, Baseline EELV Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 775	1 in 1204	1 in 357	1 in 357
Strap-CCB	–	1 in 430	1 in 711	1 in 222	1 in 222
Orion Service Module	1 in 2,726	1 in 51,800	1 in 12,419	1 in 2,063	1 in 2,063
Vehicle Total					1 in 128

Table 7.4: Loss of Mission Reliability Assessment for Delta IV, 3 x CCB, CEV/SM Active, Human-rated Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 1,293	1 in 2,040	1 in 2,668	1 in 610
Strap-CCB	–	1 in 656	1 in 1,093	1 in 2,352	1 in 349
Orion Service Module	1 in 2,726	1 in 51,800	1 in 54,923	1 in 12,419	1 in 2,063
Vehicle Total					1 in 200

functional sooner.

The reliability assessment for the Atlas V HLV Active configuration is presented in Table 7.5 and Table 7.6 with baseline EELV and human rated components. The numbers presented here are identical to those presented for the Delta IV configurations. Since the CM/SM is only used to perform one of the burns in this configuration, its engine startup failure probability has been reduced by a factor of two. The reliability for the Atlas V configuration is lower than that of the Delta IV, but this is to be expected since this configuration has three active components as opposed to the Delta IV configurations which only have two active components.

Table 7.5: Loss of Mission Reliability Assessment for Atlas V HLV, CEV/SM Active, Baseline EELV Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 775	1 in 1,204	1 in 357	1 in 357
Strap-CCB	–	1 in 430	1 in 711	1 in 222	1 in 222
RL10A-4-2	1 in 1,406	1 in 1,406	1 in 3,170	1 in 1,977	1 in 399
Orion Service Module	1 in 2,726	1 in 51,800	1 in 12,419	1 in 2,063	1 in 2,063
Vehicle Total					1 in 99

Table 7.6: Loss of Mission Reliability Assessment for Atlas V HLV, CEV/SM Active, Human-Rated Components

Component	Startup Failure	Contained	Un-contained	Other	Total
Core Booster	–	1 in 1,293	1 in 2,040	1 in 2,668	1 in 610
Strap-CCB	–	1 in 656	1 in 1,093	1 in 2,352	1 in 349
RL10A-4-2	1 in 2,564	1 in 1,815	1 in 5,593	1 in 3,558	1 in 714
Orion Service Module	1 in 5,452	1 in 51,800	1 in 54,923	1 in 12,419	1 in 3,317
Vehicle Total					1 in 161

7.2 Launch Abort System Reliability Assessment

In presenting this information it is important to understand the fact that a “loss of mission” does not necessarily mean there will be a “loss of crew.” It is imperative that the launch abort system is taken into account when discussing the topic of a loss of crew. The reliability values presented here for the launch abort system once again come from Maggio and Hall. The end to end loss of crew reliability is calculated by taking the reliability for the launch abort system and combining it with that for the EELV configurations presented in the previous section. The Information Laboratory study used the Dynamic Abort Risk Evaluator (DARE) Model to do this assessment. DARE calculates the potential blast yield at the time of booster failure, and then determines the critical distance at which the blast pressure wave will no longer cause a critical Orion failure. If the Orion (after abort) is outside of the critical region, then a successful abort is assumed to have occurred. The numbers to be presented here assume a 100 percent probability that a first stage failure will propagate through the system and cause a failure in the second stage as well. The blast center is assumed to be distributed uniformly over the length of the first stage.

The crew abort effectiveness (CAE) index was established to be 0.83, and the abort effectiveness varies only slightly amongst the configurations in the study done by Maggio and Hall. Thus 83 percent of the missions aborted will be survived by the crew. This value was combined with that of the LOM numbers and we get a rough estimate of the end to end LOC numbers. A summary of these results, for all configurations presented in this paper, are presented in Table 7.7. This analysis was done assuming that the LAS system is jettisoned after the burn out of the first stage and is no longer available afterward, but

Table 7.7: LOM/LOC Summary for Three Proposed EELV Derived Configurations (83 percent Crew Abort Effectiveness)

Configuration	Baseline Components LOM (LOC)	Human Rated LOM (LOC)
Atlas V HLV, Active	1 in 99 (1 in 247)	1 in 161 (1 in 405)
Delta IV Heavy, Passive	1 in 102 (1 in 267)	1 in 169 (1 in 461)
Delta IV Heavy, Active	1 in 128 (1 in 579)	1 in 200 (1 in 937)

obviously if the LAS was not jettisoned until the second stage then the chances of a loss of crew would decrease even more.

It must be noted that the numbers presented in Table 7.7 for the three configurations discussed in this paper are better then the space shuttle operational numbers. To date out of 131 launches the space shuttle has lost two missions, both of which the crew was lost and the shuttle was destroyed. This statistic translates to a LOM and a LOC probability of one in 65.5. In total the USA has launched 158 crewed space missions with the additional loss of the Apollo 13 mission. Thus the actual reliability LOM for the space shuttle is one in 52.7. The USA space LOC probability is one in 79 and the official LOC that NASA claims for the space shuttle is one in 423. NASA claims that the Ares I/Orion configuration will have a LOC value of one in 1,918 launches, but it is still unclear as to how NASA established this assessment [51].

Chapter 8

Augustine Commission

On May 7, 2009 the Office of Science and Technology Policy (OSTP) announced a committee that would review the current U.S. human space flight plans. The committee was named The Review of United States Human Space Flight Plans Committee, also known as the HFS committee, or the Augustine Commission. This committee was formed with the goal of ensuring the nation is on a “vigorous and sustainable path to achieving its boldest aspirations in space” [52].

In October of 2009 the Augustine Commission released the results from their study [53]. This study discussed the future of the Shuttle, ISS, the Constellation Program, and the overall future of the U.S. human space program. In their study they discussed that the Ares I was originally scheduled to be fully operational by 2012, but due to development issues it was soon delayed to 2015, and through their study they discovered that with the current state of the Ares I development it is now probable that the soonest the Ares I will be fully operational is 2017, 2 years after the ISS is scheduled to be de-orbited. The study states that in order for the 2017 Ares I time line to be met NASA will need an additional 3 billion U.S. dollars to the future Ares I budget, but Congress has already stated that no additional funding will be added to the Constellation program.

The Augustine Commission discussed the option of extending the life of the ISS to 2020 in order to support the Ares I, but their findings showed that not only would this cause the U.S. to find additional funding to support the ISS for an extra five years, but it would also financially impact all nations that participate in the ISS missions. If the ISS life time was lengthened out to 2020 this would in turn effect the current budget of the Ares I, as a portion of the future Ares I budget is planned to be pulled from the current ISS budget after the ISS is de-orbited, causing only more budget issues for the Ares I.

In order to shorten the seven year gap from when the Shuttle is scheduled to retire and when the Ares I is operational, the Augustine Commission discussed the possibility of extending the life of the Shuttle for an additional year. They discovered that this optional would also cut into the future Ares I budget as the future Ares I budget also relies absorbing the Shuttle's current budget after it is retired. Thus the conclusion of the Augustine Commission was that the current Constellation program needs to be refined to exclude the current Ares I launch vehicle configuration, and that other options were more viable.

The Augustine Commission discussed the option of having the Orion capsule lifted to a low Earth orbit using the current EELV's. In this discussion they state that the current EELV's could be used as a cost saving option, but that in order to make the current Delta IV Heavy compatible with the Orion capsule there would be an added cost. The Augustine Commission did conclude that using commercial launch vehicle for the future of the U.S. Human Space industry is a viable option.

Chapter 9

Summary and Conclusion

The Space Shuttle is currently scheduled to retire in 2011, and the Ares I is not scheduled for its first manned flight until 2016 during this time the USA will be dependent on the Russian launcher Soyuz. Throughout the design process the Ares I has suffered several setbacks that might delay its initial launch even more. But if the current time line of the Ares I is 100 percent successful there will still exist a five year where the USA will be dependent on other countries for access to the ISS. Any further delays in the Ares I schedule will significantly impact, and possibly terminate the science and mission operations of the ISS. This thesis explores the possibility of using existing Evolved Expendable Launch Vehicles for delivering the Orion spacecraft to the International Space Station. This method will in turn narrow the gap in which the USA does not have direct access to the International Space Station.

Three Launch options are presented the first is the Atlas V HLV. The Atlas V HLV configuration requires propulsion from the Orion service module be used in order to reach the final orbit. The second option presented is the Delta IV Heavy, this configuration launches the Orion as a passive payload and requires no impulsive maneuvering from the service module in order to reach the final orbit. The third configuration presented is comprised of the Delta IV Heavy with three common core boosters as the first stage, and the Orion spacecraft acting as the second stage. This configuration requires a significant impulse from Orion's service module engine to achieve the final orbit. After final orbit insertion all three configurations still had the necessary amount of propellant for de-orbit and re-entry. An equilibrium gas-chemistry code was used to model the combustion products based on mean properties for the specified propellants.

During this analysis extensive simulation studies were performed in order to verify that

the three configurations proposed were capable of performing the mission. The simulations were performed with data gathered from several public domain sources, which were cross referenced for accuracy. As required thermo-chemical data were adjusted to give consistency between the specified engine performance and propulsive stage models. An industry standard code was used to calculate the aerodynamics for each of the configurations. The analysis for this study showed that all three of the proposed configurations are capable of achieving the International Space Station's orbit with sufficient ΔV to allow for de-orbit and re-entry.

Public release documents were used to demonstrate the reliability of the EELV components. These studies showed that when human rated the proposed configurations will be competitive with the NASA proposed Ares I reliability and have better loss of mission and loss of crew reliability when compared to the Space Shuttle. One of the configurations proposed uses only the first stages of the Delta IV Heavy and uses the Orion service module, fully loaded with the propellant used for a lunar mission, as the second stage. This configuration simplifies the stack of the launch vehicle, thus reducing the risk of failures. Also this configuration eliminates the need to human-rate the current second stage of the Delta IV Heavy, thus since only the Common Core Booster of the first stage would need to be human-rated, this configuration leaves the possibilities for schedule buy down and cost savings. When discussing all three configurations, because the EELVs are already operational and the launch procedures and infrastructure are well established, there is a high probability that using these systems will result in a considerably compressed schedule when compared to the development schedule of the Ares I.

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Appendices

Appendix A

Rocket Thrust Equation Derivation

A.1 Newton's Third Law Method

Newton's Second Law states, "The acceleration produced by a force is directly proportional to the force and inversely proportional to the mass which is being accelerated", as shown in equation A.1. Where $\bar{F}$ is the force vector, m is the mass, and $\bar{a}$ is the acceleration vector. But Newton soon realized that this law only holds when mass is constant. Newton modified the second law accordingly, equation A.2. Which is equal to equation A.3 $\bar{P}$ is the momentum vector. This modification to the Second Law became Newton's Third Law, The Conservation of Momentum, which states that for every action, there is an equal and opposite reaction or equation A.5.

$$\bar{F} = m\bar{a} = m \frac{d\bar{V}}{dt} \quad (\text{A.1})$$

$$\bar{F} = \frac{d(m\bar{V})}{dt} = \frac{d(P)}{dt} \quad (\text{A.2})$$

$$\bar{P} = m\bar{V} \quad (\text{A.3})$$

$$M_1 V_1 = M_2 V_2 \quad (\text{A.4})$$

Equation A.5 can also be written in the form shown in equation ??, where ΔM and ΔV are the change in mass and velocity, and U_{ex} is the exit velocity of the mass leaving the rocket, also known as the effective exhaust velocity. Fig. A.1 shows a good depiction

Fig. A.1: Depiction of Newton's Third Law as It Applies to a Rocket.

of this.

$$MV = (M - \Delta M)(V + \Delta V) + \Delta M(V - U_{ex}) \quad (\text{A.5})$$

By manipulating equation A.5 we can extract the rocket equation as shown below.

$$\begin{aligned} MV &= (M - \Delta M)(V + \Delta V) + \Delta M(V - U_{ex}) \\ MV &= MV - \Delta MV + M\Delta V - \Delta M\Delta V + \Delta MV - \Delta MU_{ex} \\ M\Delta V &= \Delta MU_{ex} + \Delta M\Delta V \end{aligned}$$

Finally if divide by time, Δt and evaluate the limit, A.6 we get the reaction force on the rocket or what is also known as the rocket thrust equation A.7.

$$\frac{M \frac{\Delta V}{\Delta t} = \frac{\Delta M}{\Delta t} U_{ex} + \Delta M \frac{\Delta V}{\Delta t}}{\lim(\Delta M, \Delta V, \Delta t)} \quad (\text{A.6})$$

$$F = M \frac{dV}{dt} = \frac{dM}{dt} U_{ex} \quad (\text{A.7})$$

And if we take the equation for the effective exhaust velocity A.8 and plug it into equation A.7 we get the more conventional rocket equation A.9.

$$U_{ex} = V_e + \frac{p_e A_e - p_\infty A_e}{\dot{m}_e} \quad (\text{A.8})$$

$$F_{thrust} = \dot{m}_e V_e + (p_e A_e - p_\infty A_e) \quad (\text{A.9})$$

A.2 Conservation of Mass and Momentum Method

Conservation of Mass

The Thrust Equation can also be derived through the conservation of mass and momentum. Assume an arbitrary volume (CV), fixed in space, with fluid flux across its boundaries as shown in Fig. A.2. The outer surface of the control volume we will call the *ControlSurface* (CS). And the mass within the control volume at any instant in time t we define as the *System*. Now mass within the control volume must obey the laws of mass and momentum conservation.

The conservation of mass law states that mass contained within the control volume is conserved at any instant in time, and is shown in equation A.10. And within an elemental piece of the control volume, dv , as shown in Fig. A.2, and the element density is equal to an elemental piece of the mass, equation A.11. Thus the total contained mass is described in equation A.12.

$$\frac{\delta}{\delta t}(m_{CV}) = \int \int \int_{CV} \dot{m}_{in} - \int \int \int_{CV} \dot{m}_{out} \quad (\text{A.10})$$

$$dm_{CV} = \rho dv \quad (\text{A.11})$$

$$m_{CV} = \int \int \int_{CV} \rho dv \quad (\text{A.12})$$

Fig. A.2: Conservation of Mass and Momentum System.

The mass flow out of the CV across an elemental piece of the control surface, ds , is shown in equation A.13. And the incremental mass flow into the CV is described in equation A.14. If we integrate this over the entire control surface equation A.15, and applying the principle of quasi one dimensional steady flow, which states that mass flow in is equal to mass flow out, Fig. A.25, we get equation A.16, or the Continuity Equation.

$$d(\dot{m}_{out}) = \rho V_{out} \cos(\theta) ds \quad (A.13)$$

$$d(\dot{m}_{in}) = -\rho V_{in} \cos(\theta) ds \quad (A.14)$$

$$\dot{m}_{CV} = - \int \int_{CV} (\rho \vec{V} \bullet \vec{ds}) \quad (A.15)$$

$$\frac{d}{dt} \left(\int \int \int_{CV} \rho dv \right) = 0 = - \int \int_{CS} (\rho \vec{V} \bullet \vec{ds}) \longrightarrow \rho_1 V_1 A_1 = \rho_2 V_2 A_2 \quad (A.16)$$

Fig. A.3: One-Dimensional Flow.

Fig. A.4: Control Volume Element.

Conservation of Momentum

The rate of momentum of change within the control volume, Fig. A.4, is shown in equation A.17. Thus the total net rate of momentum change within the control volume can be described by equation A.18.

$$\frac{d}{dt}(m\vec{V}) \quad (\text{A.17})$$

$$\frac{d}{dt}(m\vec{V})_{CV} = \frac{d}{dt} \iiint_{CV} (\rho dv \vec{V}) = \iiint_{CV} \frac{d}{dt} (\rho \vec{V}) dv \quad (\text{A.18})$$

Then if we apply Newton's Second Law, to account for the forces acting on the control

volume; body forces, equation A.22, pressure forces, equation A.20, and the viscous forces, equation A.21, we get equation A.23.

$$(\sum \vec{F})_{body} = \int \int \int_{CV} \rho \vec{f}_b dv \quad (A.19)$$

$$(\sum \vec{F})_{body} = - \int \int_{CS} p \vec{dS} \quad (A.20)$$

$$(\sum \vec{F})_{friction} = \int \int_{CS} |\vec{f} \times \vec{dS}| \frac{\vec{dS}}{S_{CS}} \quad (A.21)$$

$$(\sum \vec{F})_{body} = \int \int \int_{CV} \rho \vec{f}_b dv \quad (A.22)$$

$$\int \int_{CS} (\rho \vec{V} \bullet \vec{ds}) \vec{V} + \int \int \int_{CV} \frac{d}{dt} (\rho \vec{V}) dv \quad (A.23)$$

Assuming steady, inviscous flow, and that the body forces are negligible, this equation can be reduced to equation A.24.

$$- \int \int_{CS} p \vec{dS} = \int \int_{CS} (\rho \vec{V} \bullet \vec{ds}) \vec{V} \quad (A.24)$$

and from applying the principles of one dimensional flow shown in Fig. A.25, get equation ??.

$$p_1 A_1 V_1 + \rho_1 V_1^2 A_1 + \int p \vec{ds} \bullet \vec{i}_x = p_2 A_2 V_2 + \rho_2 V_2^2 A_2 \quad (A.25)$$

Then from this equation if we apply the conservation laws to the engine thrust model, shown in Fig. A.5, Fig. A.6, and integrating the pressure forces acting on the external and internal surface of the engine wall we get equation A.26.

$$\int_{wall} P_{wall} dA_{wall} + p_{\infty} (A_i - A_e) = \dot{m}_e V_e - \dot{m}_i V_i + (p_e A_e - p_{\infty} A_e) \quad (A.26)$$

Fig. A.5: One-Dimensional Engine Thrust Model Showing Pressure Distribution on External and Internal Walls of Engine.

Fig. A.6: One-Dimensional Engine Thrust Model Showing Pressure Distribution, Velocity, and Area of Engine Model.

The left side of equation A.26 is the thrust of the engine, and thus this equation can be reduced to equation A.27, or the Rocket Thrust Equation.

$$F_{thrust} = \dot{m}_e V_e + (p_e A_e - p_\infty A_e) \quad (\text{A.27})$$

Appendix B

Flight Path Equations of Motion

The following are the equations of motion, used in the launch simulation, as derived by Schulz, Prado, and Morales [34].

FLIGHT PATH EQUATIONS OF MOTION (Rotating Oblate Earth and Zonal Gravity Model)

The first-order flight path equations of motion relative to a rotating oblate Earth and a fourth-order zonal gravity model are as follows:

Geocentric radius

$$\dot{r} = \frac{dr}{dt} = V \sin \gamma$$

Geographic longitude

$$\dot{\lambda} = \frac{d\lambda}{dt} = V \frac{\cos \gamma \sin \psi}{r \cos \delta}$$

Geocentric declination

$$\dot{\delta} = \frac{d\delta}{dt} = V \frac{\cos \gamma \cos \psi}{r}$$

Speed

$$\begin{aligned} \dot{V} = \frac{dV}{dt} = & \frac{(T \cos \alpha - D)}{m} - g_r \sin \gamma + g_\phi \cos \gamma \cos \psi \\ & + \omega_e^2 r \cos \delta (\sin \gamma \cos \delta - \sin \delta \cos \gamma \cos \psi) \end{aligned}$$

Flight path angle

$$\begin{aligned} \dot{\gamma} = \frac{d\gamma}{dt} = & \frac{V}{r} \cos \gamma + \left(\frac{T \sin \alpha + L}{mV} \right) \cos \beta - \frac{g_r \cos \gamma}{V} - \frac{g_\phi \sin \gamma \cos \psi}{V} \\ & + 2\omega_e \sin \psi \cos \delta + \omega_e^2 \frac{r}{V} \cos \delta (\cos \psi \sin \gamma \sin \delta + \cos \gamma \cos \delta) \end{aligned}$$

Flight azimuth

$$\begin{aligned} \dot{\psi} = \frac{d\psi}{dt} = & \frac{V}{r} \tan \delta \sin \psi \cos \gamma + \left(\frac{T \sin \alpha + L}{mV \cos \gamma} \right) \sin \beta - \frac{\sin \psi}{V \cos \gamma} g_\phi \\ & + 2\omega_e (\sin \delta - \cos \psi \cos \delta \tan \gamma) + \frac{r}{V \cos \gamma} \omega_e^2 \sin \psi \cos \delta \sin \delta \end{aligned}$$

where

- r = geocentric radius
- v = speed
- γ = flight path angle
- δ = geocentric declination
- λ = longitude (+ east)
- ψ = flight azimuth (+ clockwise from north)
- β = bank angle (+ for a right turn)
- α = angle of attack
- r_e = Earth equatorial radius
- ω_e = Earth inertial rotation rate
- μ = Earth gravitational constant
- L = aerodynamic lift force $= \frac{1}{2} \rho v^2 C_L S$
- D = aerodynamic drag force $= \frac{1}{2} \rho v^2 C_D S$
- T = propulsive thrust
- m = spacecraft mass
- C_L = lift coefficient (non-dimensional)
- C_D = drag coefficient (non-dimensional)
- S = aerodynamic reference area
- ρ = atmospheric density $= f(h)$
- h = geodetic altitude

The bank angle is the angle between the lift vector and the projection of the lift vector on the plane formed by the vehicle's relative velocity vector and the local vertical direction. Bank angle is measured positive clockwise looking forward in the direction of motion.

The components of the gravity vector are determined from the gradient of the potential function according to

$$\mathbf{F}_G = \nabla U = \begin{Bmatrix} \frac{1}{r} \frac{\partial U}{\partial \phi} \\ 0 \\ -\frac{\partial U}{\partial r} \end{Bmatrix} = \begin{Bmatrix} g_\phi \\ 0 \\ g_r \end{Bmatrix}$$

where

$$U = \frac{\mu}{r} \left[1 - \sum_{j=1}^{\infty} \left(\frac{r_e}{r} \right)^j J_j P_{j0}(\sin \phi) \right]$$

$$\frac{\partial U}{\partial r} = \frac{\mu}{r^2} \left[-1 + \sum_{l=2}^{\infty} (l+1) \left(\frac{r_e}{r} \right)^l J_l P_{l0}(\sin \phi) \right]$$

$$\frac{1}{r} \frac{\partial U}{\partial \phi} = -\frac{\mu}{r^2} \sum_{l=2}^{\infty} \left(\frac{r_e}{r} \right)^l J_l \frac{\partial P_{l0}}{\partial \phi}$$

$$P_{j0}(\sin \phi) = \sum_{t=0}^k T_{jt} \sin^{j-2t} \phi$$

$$T_{jt} = (-1)^t (2j-2t)! / 2t! (j-t)! (j-2t)!$$

For a *zonal-only* gravity model of order four, the Legendre functions and their partial derivatives are given by

$$P_{20} = \frac{1}{2} (3 \sin^2 \phi - 1)$$

$$P_{30} = \frac{1}{2} (5 \sin^3 \phi - 3 \sin \phi)$$

$$P_{40} = \frac{1}{8} (35 \sin^4 \phi - 30 \sin^2 \phi + 3)$$

$$\frac{\partial P_{20}}{\partial \phi} = 3 \sin \phi \cos \phi$$

$$\frac{\partial P_{30}}{\partial \phi} = \frac{3}{2} (5 \sin^2 \phi - 1) \cos \phi$$

$$\frac{\partial P_{40}}{\partial \phi} = \frac{5}{2} (7 \sin^2 \phi - 3) \sin \phi \cos \phi$$

Load Factors

“total” (wind axis coordinate system)

$$n = \frac{\sqrt{(L + T \sin \alpha)^2 + (D - T \cos \alpha)^2}}{W}$$

axial component (body frame coordinate system)

$$n = \frac{-L \sin \alpha + D \cos \alpha - T}{W}$$

normal component (body frame coordinate system)

$$n = \frac{L \cos \alpha + D \sin \alpha}{W}$$

Axial and Normal Coefficients (body axis coordinate system)

$$C_A = -C_L \sin \alpha + C_D \cos \alpha - T$$

$$C_N = C_L \cos \alpha + C_D \sin \alpha$$

Lift and Drag Coefficients (body axis coordinate system)

$$C_L = -C_A \sin \alpha + C_N \cos \alpha$$

$$C_D = C_A \cos \alpha + C_N \sin \alpha$$

Axial and Normal Force Components (body axis coordinate system)

$$\mathbf{f} = \frac{1}{2} \rho V^2 S \begin{Bmatrix} -C_A \\ 0 \\ -C_N \end{Bmatrix}$$

Steady-State Flight Conditions

$$T \cos \alpha - D - W \sin \gamma = 0$$

$$T \sin \alpha + L - W \cos \gamma = 0$$

Aerodynamic Characteristics

General Form of Drag Polar

$$C_D = C_{D_0} + k|C_L|^n$$

Lift-to-Drag Ratio

$$E = \frac{L}{D} = \frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + kC_L^n}$$

Maximum Lift-to-Drag Ratio

$$E^* = \frac{dE}{dC_L} = \frac{(C_{D_0} + kC_L^n) - C_L(nkC_L^{n-1})}{(C_{D_0} + kC_L^n)^2} = 0$$

$$C_L^* = \sqrt[n]{\frac{C_{D_0}}{k(n-1)}}$$

$$C_D^* = \frac{nC_{D_0}}{(n-1)}$$

In general,

$$E^* = \frac{C_L^*}{C_D^*} = \frac{\sqrt[n]{(n-1)^{n-1}}}{n\sqrt[n]{kC_{D_0}^{n-1}}}$$

For a parabolic drag polar ($n = 2$),

$$C_D = C_{D_0} + kC_L^2$$

where

C_D = drag coefficient

C_{D_0} = drag coefficient at 0° angle-of-attack

C_L = lift coefficient

k = constant

and

$$C_D^* = 2C_{D_0} = \text{drag coefficient at maximum } L/D$$

$$C_L^* = \sqrt{C_{D_0} / k} = \text{lift coefficient at maximum } L/D$$

$$E^* = \left(\frac{C_L}{C_D} \right)_{\max} = \frac{1}{2\sqrt{kC_{D_0}}} = \text{maximum } L/D$$

Chapman's Stagnation Point Heat Rate Equation

$$q = \frac{dq}{dt} = \frac{17,600}{\sqrt{R_N}} \left(\frac{v}{v_0} \right)^{3.15} \sqrt{\frac{\rho}{\rho_0}} \quad \left(\frac{\text{BTU}}{\text{ft}^2 - \text{sec}} \right)$$

where

R_N = nose radius (feet)

v = relative velocity at the spacecraft location (feet/second)

v_0 = "local circular velocity" at the Earth's surface = $\sqrt{\frac{\mu}{r_e}}$ (feet/second)

ρ = atmospheric density at the spacecraft location (slugs/feet³)

ρ_0 = atmospheric density at the Earth's surface (slugs/feet³)

μ = gravitational constant of the Earth (feet³/second²)

r_e = radius of the Earth (feet)

Crossrange and Downrange Calculations (Spherical Earth)

The crossrange angle is determined from the following expression:

$$\sin \nu = -\sin \psi_1 \sin \phi_2 \cos \phi_1 \cos \Delta\lambda - \cos \psi_1 \cos \phi_1 \sin \Delta\lambda + \sin \psi_1 \cos \phi_2 \sin \phi_1$$

The downrange angle is determined from the following three equations:

$$\sin \mu = -\cos \psi_1 \sin \phi_2 \cos \phi_1 \cos \Delta\lambda + \sin \psi_1 \cos \phi_1 \sin \Delta\lambda + \cos \psi_1 \cos \phi_2 \sin \phi_1$$

$$\cos \mu = \cos \phi_2 \cos \phi_1 \cos \Delta\lambda + \sin \phi_2 \sin \phi_1$$

$$\mu = \tan^{-1}(\sin \mu, \cos \mu)$$

where

ϕ_1 = geocentric latitude of the initial point

ψ_1 = flight azimuth at the initial point

ϕ_2 = geocentric latitude of the final point

$\Delta\lambda = \lambda_2 - \lambda_1$

λ_1 = east longitude of the initial point

λ_2 = east longitude of the final point

The crossrange distance d_c and downrange distance d_d are determined from

$$d_c = r_e \nu$$

$$d_d = r_e \mu$$

where r_e is the radius of the Earth. The flight azimuth is measured positive clockwise from north. Also note that the inverse tangent above is a four quadrant form.

UTILITY TRANSFORMATIONS

Conversion of ECI state vector to spherical (ADBARV) coordinates

The components of the ADBARV coordinate system are as follows:

Alpha = right ascension

Delta = geocentric declination

Beta = conjugate flight path angle

A = azimuth

R = position magnitude

V = velocity magnitude

The following diagram illustrates the geometry of the ADBARV coordinates. In this picture α is the right ascension, δ is the geocentric declination and β is the conjugate flight path angle.

The mathematical relationships between ADBARV elements and the components of the ECI position and velocity vectors are as follows:

$$r = \sqrt{r_x^2 + r_y^2 + r_z^2}$$

$$v = \sqrt{v_x^2 + v_y^2 + v_z^2}$$

$$\alpha = \tan^{-1}(r_y, r_x)$$

$$\delta = \tan^{-1}\left(r_z, \sqrt{r_x^2 + r_y^2}\right)$$

$$\beta = \cos^{-1}\left(\frac{\mathbf{r} \bullet \mathbf{v}}{|\mathbf{r} \bullet \mathbf{v}|}\right)$$

$$A = \tan^{-1}\left[r(r_x v_y - r_y v_x), r_y(r_y v_z - r_z v_y) - r_x(r_z v_x - r_x v_z)\right]$$

Conversion of spherical (ADBARV) coordinates to ECI state vector

The inertial position and velocity vectors can be determined from the ADBARV elements with the following set of equations:

$$r_x = r \cos \delta \cos \alpha$$

$$r_y = r \cos \delta \sin \alpha$$

$$r_z = r \sin \delta$$

$$v_x = v \left[\cos \alpha (-\cos A \sin \beta \sin \delta + \cos \beta \cos \delta) - \sin A \sin \beta \sin \alpha \right]$$

$$v_y = v \left[\sin \alpha (-\cos A \sin \beta \sin \delta + \cos \beta \cos \delta) + \sin A \sin \beta \cos \alpha \right]$$

$$v_z = v (\cos A \cos \delta \sin \beta + \cos \beta \cos \delta)$$

The inertial speed can also be computed from the following expression

$$v_i = \sqrt{v^2 + 2vr\omega \cos \gamma \sin \psi \cos \delta + r^2 \omega^2 \cos^2 \delta}$$

The inertial flight path angle can be computed from

$$\cos \gamma_i = \frac{v^2 \cos^2 \gamma + 2vr\omega \cos \gamma \cos \psi \cos \delta + r^2 \omega^2 \cos^2 \delta}{v^2 + 2vr\omega \cos \gamma \cos \psi \cos \delta + r^2 \omega^2 \cos^2 \delta}$$

The inertial azimuth can be computed from

$$\cos \psi_i = \frac{v \cos \gamma \cos \psi + r\omega \cos \delta}{\sqrt{v^2 \cos^2 \gamma + 2vr\omega \cos \gamma \cos \psi \cos \delta + r^2 \omega^2 \cos^2 \delta}}$$

where all coordinates on the right-hand-side of these equations are relative to a rotating Earth.

Conversion of geocentric radius and declination to geodetic altitude and latitude

The following diagram illustrates the geometric relationship between geocentric and geodetic coordinates for an oblate spheroid.

In this diagram, δ is the *geocentric* declination, ϕ is the *geodetic* latitude, r is the geocentric radius, and h is the geodetic altitude. The exact mathematical relationship between geocentric and geodetic coordinates is given by the following system of two nonlinear equations

$$(c + h) \cos \phi - r \cos \delta = 0$$

$$(s + h) \sin \phi - r \sin \delta = 0$$

where the geodetic constants c and s are given by

$$c = \frac{r_{eq}}{\sqrt{1 - (2f - f^2) \sin^2 \phi}}$$

$$s = c (1 - f)^2$$

and r_{eq} is the Earth equatorial radius (6378.14 kilometers) and f is the flattening factor for the Earth (1/298.257).

The geodetic latitude is determined using the following expression:

$$\phi = \delta + \left(\frac{\sin 2\delta}{\rho} \right) f + \left[\left(\frac{1}{\rho^2} - \frac{1}{4\rho} \right) \sin 4\delta \right] f^2$$

The geodetic altitude is calculated from

$$\hat{h} = (\hat{r} - 1) + \left\{ \left(\frac{1 - \cos 2\delta}{2} \right) f + \left[\left(\frac{1}{4\rho} - \frac{1}{16} \right) (1 - \cos 4\delta) \right] f^2 \right\}$$

In these equations, ρ is the geocentric distance of the vehicle, $\hat{h} = h / r_{eq}$ and $\hat{r} = \rho / r_{eq}$.

Conversion of geodetic latitude and altitude to geocentric radius and geocentric declination

The equations for this coordinate conversion are as follows:

$$\delta = \phi + \left(\frac{-\sin 2\phi}{\hat{h} + 1} \right) f + \left\{ \frac{-\sin 2\phi}{2(\hat{h} + 1)^2} + \left[\frac{1}{4(\hat{h} + 1)^2} + \frac{1}{4(\hat{h} + 1)} \right] \sin 4\phi \right\} f^2$$

and

$$\hat{\rho} = (\hat{h} + 1) + \left(\frac{\cos 2\phi - 1}{2} \right) f + \left\{ \left[\frac{1}{4(\hat{h} + 1)} + \frac{1}{16} \right] (1 - \cos 4\phi) \right\} f^2$$

where the geocentric radius r and geodetic altitude h have been normalized by $\hat{\rho} = r / r_{eq}$ and $\hat{h} = h / r_{eq}$, respectively, and r_{eq} is the equatorial radius of the Earth.