Report of the Horizontal Launch Study

June 2011

Interim Report

DARPA

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NASA

Interim Report

Design, Cost, and Performance Analyses

Executive Summary

This report, jointly sponsored by the Defense Advanced Research Projects Agency (DARPA) and the National Aeronautics and Space Administration (NASA), is the result of a comprehensive study to explore the trade space of horizontal launch system concepts and identify potential near- and mid-term launch system concepts that are capable of delivering approximately 15,000 lbs to low Earth orbit. The Horizontal Launch Study (HLS) has produced a set of launch system concepts that meet this criterion and has identified potential subsonic flight test demonstrators. Based on the results of this study, DARPA has initiated a new program to explore horizontal launch concepts in more depth and to develop, build, and fly a flight test demonstrator that is on the path to reduce development risks for an operational horizontal take-off space launch system.

The intent of this interim report is to extract salient results from the in-process HLS final report that will aid the potential proposers of the DARPA Airborne Launch Assist Space Access (ALASA) program. Near-term results are presented for a range of subsonic system concepts selected for their availability and relatively low development costs. This interim report provides an overview of the study background and assumptions, idealized concepts, point design concepts, and flight test demonstrator concepts. The final report, to be published later this year, will address more details of the study processes, a broader trade space matrix including concepts at higher speed regimes, operational analyses, benefits of targeted technology investments, expanded information on models, and detailed appendices and references.

The study team carried out a three-step analysis to provide the basis for several point designs and flight demonstration system concepts defined to mitigate development risk, operational risk, and cost. These steps were (1) identify representative system concepts from past studies; (2) develop a systematic, normalized basis to compare a variety of approaches with a low analysis fidelity; and (3) further analyze selected system concepts and compare point designs with a representative set of components to demonstrate feasibility.

A number of assumptions and constraints were used to guide the study process. These included the limits of existing runways, current and projected launch rates in various payload classes, and the performance parameters of existing technologies and existing designs. After

considering an array of existing and near-term subsonic carrier aircraft, the study team determined a practical limit of payload mass to low Earth orbit of 50,000 lbs. The DDT&E cost for a subsonic carrier aircraft-based horizontal take-off space launch system to deliver 50,000 lbs to orbit was estimated to be between $4 billion and $8 billion.

For a modified existing or newly-designed subsonic carrier aircraft, payloads of up to 18,000 and 23,000 pounds, respectively, are possible with recurring launch costs as low as $3,000 per pound. Recurring costs include acquisition and production of the carrier aircraft and launch vehicle, operations, facilities, ground support equipment, a 15 percent contingency, and 20 percent program management cost. The nonrecurring cost of modifying an existing or of developing a new subsonic carrier aircraft by a non-traditional provider are both estimated at less than $2 billion. All of these estimates assumed a launch rate of six missions per year over a period of 20 years.

For a nominal reference payload of 15,000 lbs, the study team developed several subsonic carrier aircraft-based reference space launch point design vehicle (PDV) system concepts. One example is a near-term system comprised of a two-stage launch vehicle with a hydrocarbon-fueled first stage and a hydrogen-fueled second stage which is carried to launch by a modified Boeing 747-400F carrier aircraft. This system concept is estimated to require $936 million for DDT&E, and will result in a cost of approximately $9,600 per pound of payload to orbit. Aerial fueling provides further performance and cost benefits by allowing a larger launch vehicle and payload weight while meeting the carrier aircraft's maximum take-off weight. The study team found that existing technologies are sufficient to begin DDT&E on a selected subsonic carrier aircraft-based space launch system concept, and that flight testing of a technology demonstration concept could be initiated immediately.

The study team also identified a viable low-cost flight technology demonstration using the NASA Shuttle Carrier Aircraft, a modified Boeing 747-100, with either a solid rocket or liquid rocket engine launch vehicle mounted on top using existing propulsion subsystems and technologies. It was estimated that a joint DARPA/NASA demonstration program would cost less than $350 million, would take three to four years, and would achieve two to four demonstration flights with up to 5,000 pounds of payload to low Earth orbit.

Acronyms and Symbols

ACES	air collection and enrichment system
AFWAT	Air Force Weight Analysis Tool
APAS	Aerodynamic Preliminary Analysis System
CER	cost estimating relationship
CFD	computational fluid dynamics
CFD	computational fluid dynamics
DARPA	Defense Advanced Research Projects Agency
DDT&E	design, development, test, and evaluation
FEA	finite element analysis
FOM	figure of merit
FTD	Flight Test Demonstrator
FY	fiscal year
HOTOL	horizontal takeoff and landing
HRST	Highly Reusable Space Transportation
HSDTV	hypersonic technology demonstrator vehicle
HTHL	horizontal take-off, horizontal landing
LACE	liquid air cycle engine
lbm	pound mass
LEO	low Earth orbit
LH2	liquid hydrogen
LOM	loss of mission
LOX	liquid oxygen
MAKS	multipurpose aerospace system
NAFCOM	NASA/Air Force Cost Model
NASA	National Aeronautics and Space Administration
NASP	National Aero-Space Plane
NGLT	Next Generation Launch Technology program
PDV	Point Design Vehicle
POST	Program To Optimize Simulated Trajectories
psf	pounds per square foot
q	dynamic pressure
RASV	Reusable Aerodynamic Space Vehicle
RBCC	Rocket Based Combined Cycle
REDTOP	Rocket Engine Design Tool for Optimal Performance
ROSETTA	Reduced Order Simulation For Evaluating Technologies And Transportation Architectures
RP	rocket propellant
SCA	Shuttle Carrier Aircraft
SHABP	Supersonic/Hypersonic Arbitrary Body Program
SSTO	single stage to orbit
TBCC	Turbine Based Combined Cycle
TSTO	two stage to orbit
V	velocity

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Introduction

The potential benefits of horizontal launch begin with the ability to launch from existing runways and extend to the capability to provide a “mobile launch pad” that can cruise above weather, loiter for mission instructions, or achieve precise placement for orbital intercept or reconnaissance. The benefits of horizontal launch are most compelling, however, considering that today’s vertical launch pads are a single earthquake, hurricane, or terrorist attack away from disrupting critical payload delivery capabilities.

Future launch capabilities that provide space access for both civilian and military applications require robust, responsive, reliable, and economical solutions. The promise of horizontal take-off space launch systems fulfilling these requirements has inspired many studies over the years—spanning airbreathing and rocket propulsion, expendable and reusable launch vehicles, and various assisted launch concepts, such as magnetic levitation or sleds. These varied studies are difficult to compare, as each used its own, sometimes unique, figures of merit. Many focused on narrow mission requirements, such as a single payload class, market, maximum gross take-off weight, or staging Mach number. Only a few included the process for and cost of development, testing, production, ground operations, and mission operations.

This study provides a more comprehensive assessment of horizontal launch system concepts for several payload classes. The analysis focused on access to low Earth orbit on a due east launch inclination and considered military, other government, and commercial purposes. Common figures of merit were developed to provide a balanced basis for assessment. These included payload, reliability, and risk as well as the cost of development, production, and operations.

The primary goal of the study was to define potential near-term, horizontal take-off space launch system concepts with payload capability of 15,000 pounds with costs approaching existing launch systems. Because of the broad range of horizontal launch systems that have been designed in the past, a three step process was developed to progressively narrow the range of potential concepts considered in order to allow increased fidelity of the engineering analysis.

- The first step was to identify representative system concepts from past studies that spanned payload and technology development ranges to understand the impact of development risk and cost on payload, operations, and launch costs (or proxies such as system weight or reusability).

- Because no single past system study met the HLS goals, the second step was to explore past concepts and technologies in a systematic way. Thousands of combinations were evaluated and screened in an integrated, low-fidelity, idealized engineering framework.
- The third step further analyzed selected system concepts considering development cost, operations cost, payload, and risk. Point designs were built that used a variety of existing or very near-term propulsion systems to demonstrate the potential to achieve cost, reliability, and operational goals.

The three-step analysis provided the basis for several point designs and flight demonstration system concepts defined to mitigate development risk, operational risk, and cost.

I. Screening Analysis

The process began with a survey of relevant existing studies. More than 130 studies, designs, and concepts were analyzed spanning a period of nearly 60 years, and are listed in Table 1. Comparison analysis factors included the design payload, payload ranges (if scalable), configuration, and concept design maturity. The resulting system concepts were binned according to three payload classes: less than 500 lbs, 500 to 15,000 lbs, and over 15,000 lbs; and three technology development timeframes: 0 to 3 years (near term), 4 to 9 years (mid term), and more than 10 years (far term).

Table 1. List of Systems in Initial Concept Survey

Concept Name	Program Acronym	Sponsoring Organization	Performer	Last year of effort
ABLV-10	ABLV	NASA	Boeing	2003
ABLV-2	ABLV	NASA	Astrox	2003
ABLV-4	ABLV	NASA	NASA MSFC	2003
ABLV-4a	ABLV	NASA	NASA LaRC	2001
ABLV-4b	ABLV	NASA	NASA LaRC	2001
ABLV-4c	ABLV	NASA	NASA LaRC	2001
ABLV-4e	ABLV	NASA	NASA LaRC	2003
ABLV-5	ABLV	NASA	Pratt and Whitney	2003
ABLV-7a	ABLV	NASA	Boeing	2001
ABLV-7c	ABLV	NASA	Boeing	2003
ABLV-7c-LaunchAssist	ABLV	NASA	NASA MSFC	2001
ABLV-8	ABLV	NASA	Lockheed	2003
ABLV-GT	ABLV	NASA + Aerojet	Georgia Tech	2000
AFRL Space Operations Vehicle	AFRL SOV	Air Force	Faulkner Consulting	2003

Concept Name	Program Acronym	Sponsoring Organization	Performer	Last year of effort
Air Launched Sortie Vehicle	ALSV	Air Force	Pratt and Whitney	1981
Air Launched Sortie Vehicle	ALSV	Air Force	Rockwell International	1981
Air Launched Sortie Vehicle	ALSV	Air Force	General Dynamics	1981
ATS Option 3 SSTO (AB/R)	ATS	NASA	NASA LaRC	1994
ATS Option 3 TSTO (AB/R)	ATS	NASA	NASA ARC	1994
Bantam Argus	Bantam-X	NASA	Georgia Tech	1999
KLIN Argus	Bantam-X	NASA	Georgia Tech	1999
PDRE Argus	Bantam-X	NASA	Georgia Tech	1999
Stargazer	Bantam-X	NASA	Georgia Tech	1999
Starsaber	Bantam-X	NASA	Georgia Tech	2001
RBCC	DF-9	Air Force	Boeing	1998
TBCC	DF-9	Air Force	Boeing	1998
Quick Reach	FALCON	DARPA	AirLaunch LLC	2004
Airbourne Microlauncher "MLA"	Foreign	IRAD	Dassault	2008
AVATAR	Foreign	India	DRDO	2001
Dedalus/Global Hawk/WK2	Foreign		CNES/ONERA	
DRLV	Foreign		Israel Inst. Tech.	2008
HAAL Yakovlev	Foreign	Russia	Yakovlev	1994
HOTOL	Foreign	UK	British Aerospace	1982
HSDTV	Foreign	India	DRDO	2007
Interim HOTOL w/AN-225	Foreign	United Kingdom	British Aerospace	1991
MAKS - M	Foreign	USSR	NPO Molniya	1989
MAKS - OS	Foreign	USSR	NPO Molniya	1989
Rafael Light Air Launch (LAL)	Foreign	Israel	Rafael	2006
Sänger 2	Foreign	Germany	MBB	1991
Shenlong Space Plane	Foreign	China		2007
Skylon	Foreign	IRAD	Reaction Engines Ltd.	2010
Spiral 50-50	Foreign	USSR	NPO Molniya	1965
Svitiaz	Foreign	Ukraine	Nat. Space Agen. Ukraine	N/A
Telemaque	Foreign		CNES	N/A
Vozdushny Start	Foreign	Russia	Energia	N/A
Yakovlev Skylifter	Foreign	Russia		N/A
Argus	HRST	NASA	Georgia Tech	1998

Concept Name	Program Acronym	Sponsoring Organization	Performer	Last year of effort
ERJ / LACE SSTO	HRST	NASA	NASA LaRC	1998
HRST SSTO Waverider	HRST	NASA	Rockwell	1998
Hyperion	HRST	NASA	Georgia Tech	1998
Space America Concept	HRST	NASA	Space America	1998
HTS-1 Turbines (+ Tail Rocket) / Rocket	HTS	NASA	Boeing	2000
HTS-2 RBCC / Rocket	HTS	NASA	Boeing	2000
HTS-3 TBCC: Turbines/RJ/SJ (+ Tail Rocket) / Rocket	HTS	NASA	Boeing	2000
HTS-4 ACES/RBCC / Rocket	HTS	NASA	Boeing	2000
HTS-5 Turbines (+ Tail Rocket) / RBCC	HTS	NASA	Boeing	2000
HTS-6 TBCC: Turbines/RJ/SJ (+ Tail Rocket) / RBCC	HTS	NASA	Boeing	2000
HTS-7 Turbines (+ Tail Rocket) w/ 2nd stage Rocket	HTS	NASA	Boeing	2000
747 SCA X-34B	Independent	Orbital Sciences Corp	Orbital Sciences Corp	2000
Advanced Reusable Small Launch System	Independent	NASA	NASA LaRC	1999
Astroliner	Independent	Kelly Aerospace	Kelly Aerospace	
Athena	Independent	NASA	University of Michigan	1994
B-52H Responsive Air Launch	Independent	OSC/DARPA/Schafer corp.	DARPA	2004
Beta	Independent	NASA	Boeing	1991
BETA II	Independent	NASA	LaRC + Boeing	1992
Black Horse	Independent	Pioneer Astronautics	Pioneer Astronautics	2000
BladeRunner	Independent	US Air Force	SMC	2004
Boeing F-15 with TS rocket on back	Independent			
Crossbow	Independent	IRAD	Teledyne Brown	2010
F-15 Microsatellite Launch Vehicle	Independent	US Air Force	Boeing	2003
LauncherOne (WK2 + upperstage)	Independent	Scaled Composites	Scaled Composites	2010
Lazarus	Independent	Georgia Tech	Georgia Tech	2006
Lynx II	Independent	XCOR	XCOR	

Concept Name	Program Acronym	Sponsoring Organization	Performer	Last year of effort
Mustang	Independent	UCF	UCF	2003
NanoLaunch LLC concept	Independent	Premier Space Systems	Premier Space Systems	2010
NanoLauncher Black	Independent	IHI	SEI + IHI	2010
Pathfinder	Independent	Pioneer Rocketplane	Pioneer Rocketplane	1998
Pegasus	Independent	Orbital Sciences	Orbital Sciences	2010
Peregrine	Independent	Andrews Space	Andrews Space	2004
RASV SSTO HTHL Rocket	Independent	Boeing	Boeing	1976
Reusable Orbital Carrier	Independent	Lockheed	NASA	1964
SA-1	Independent	Space Access LLC	Space Access LLC	1998
Sea Argus	Independent	SEI	SEI	2004
StarRunner	Independent	Georgia Tech	Georgia Tech	2004
SuperLACE/ACES Aerospaceplane	Independent		General Dynamics	1962
US Spaceplane	Independent	US Spaceplane Sys.	US Spaceplane Sys.	2010
ICM-2 TSTO HTHL TBCC/Rocket	NGLT	NASA	SAIC + McKinney	2003
ICM-3 TSTO HTHL RBCC/Rocket	NGLT	NASA	SAIC + McKinney	2003
ICM-4 TSTO HTHL Turbine/RBCC/Rocket	NGLT	NASA	SAIC + McKinney	2003
ICM-5 SSTO HTHL TBCC	NGLT	NASA	SAIC + McKinney	2003
RALV-B	JSS	NASA	NASA LaRC	2010
Lockheed-Marquardt TSTO - ESJ	NAS-377	NASA	Lockheed	1967
Lockheed-Marquardt TSTO - LACE	NAS-377	NASA	Lockheed	1967
Early Aerospaceplane (Gregory)	NASP	NASA	NASA LaRC	1970
NASP - GD	NASP	NASA + DoD	General Dynamics	1988
NASP - McDonnell Douglas	NASP	NASA + DoD	McDonnell Douglas	1988
NASP - Rockwell	NASP	NASA + DoD	Rockwell	1988
NASP Derived Vehicle	NASP	NASA	NASA LaRC	1991
FASST	NGLT	NASA	Boeing	2003
Gryphon	NGLT	NASA	Andrews Space	2003
Spaceliner 100	NGLT	NASA	NASA MSFC	1999
RASCAL - Coleman	RASCAL	DARPA	Coleman Team	2003
RASCAL - Delta Velocity	RASCAL	DARPA	Delta Velocity	2003

Concept Name	Program Acronym	Sponsoring Organization	Performer	Last year of effort
RASCAL - Northrop Grumman	RASCAL	DARPA	Northrop Grumman	2003
RASCAL - Pioneer Rocketplane	RASCAL	DARPA	Pioneer Rocketplane	2003
RASCAL - Space Access LLC	RASCAL	DARPA	Space Access LLC	2003
RASCAL - Space Launch Corp.	RASCAL	DARPA	Space Launch Corp.	2003
RASCAL - GT	RASCAL	Georgia Tech	Georgia Tech	2005
Quicksat	Robust Scramjet	US Air Force	SEI	2004
Spiral-2	Robust Scramjet	US Air Force	SEI	2006
Trans-Atmospheric Vehicle (747-Launched 1.5 Stage)	TAV	US Air Force	Rockwell	1984
Trans-Atmospheric Vehicle (GEM-Launched SSTO)	TAV	US Air Force	Rockwell	1984
Aztec	URETI	NASA	Georgia Tech	2004
X-43 Space Access Vision Vehicle	X-43 SAVV	NASA	SAIC + Boeing + McKinney	
ABLV-4c-Derived SSTO TBCC/Rocket w/ MHD	NGLT	NASA	ANSER + Lockheed	1998
Boeing Airlaunch w/747	ORSMV	US Air Force	Boeing	1999
BWB-2 SSTO LACE/DMSJ/Rocket	NASP	NASA + DoD	McDonnell Douglas	1987
NCB-3 SSTO LACE/DMSJ/Rocket	NASP	NASA + DoD	McDonnell Douglas	1987
SSTO HTHL LACE/DMSJ/Rocket				1994

Rather than quantitatively analyze the complete range of concepts and technologies, representative system concepts were selected based on cost and payload capability estimates, taking weight, complexity, operational profiles, and other factors into account via expert judgment. The process was used to identify a range of feasible solutions that could meet the study goals. Further analysis to identify the best solutions is left to future designers.

II. Vehicle Concept Exploration

The goals for the vehicle concept exploration were to identify concepts with useful payloads approaching 15,000 lbs due east to low Earth orbit with development costs in line with current budget estimates and production, and operations costs approaching current launch systems. To ensure military usefulness, the concepts were constrained to a gross takeoff weight less than 1.5 million pounds to meet conventional runway requirements. Minimal ground support was also specified, and flight rates were set at current market projections of approximately six flights per year. A number of additional assumptions that guided the analysis are listed in Table 2.

Table 2. Assumptions Used in Vehicle Concept Exploration

Weights and sizing	Compilation of mass estimating relationships from the StageSizer model, previous studies, and other references
Stage sizing philosophy	All stages were fully parametric; no existing hardware assumed
Stage Thrust-to-weight ratio at ignition	1st Stage: 1.20 2nd Stage: 1.15 3rd Stage: 1.10
Solid motor propellant mass fraction	0.93
Liquid propellant tank unit weight (fuel and oxidizer)	0.8 lbs/ft3
Wing unit weight	5.0 lbs/ft2
Payload density	8.0 lbs/ft3
Payload fairing unit weight	2.75 lbs/ft2
Interstage/intertank unit weight	4.3 lbs/ft2
Propellant reserves, residuals, and start-up losses	1.8% of ideal propellant mass
Propellant ullage	2.0% of required propellant volume
Dry weight growth margin	15%
Propulsion	All engines/motors were fully parametric; no existing hardware assumed
Vacuum specific impulse	Solid motor: 290s (1st stages), 292s (2nd/3rd stages) LOX/RP engine: 346s (1st stages), 354s (2nd/3rd stages) LOX/LH2 engine: 450s (1st stages), 460s (2nd/3rd stages)
Liquid rocket engine vacuum thrust-to-weight	LOX/RP engine: 100 LOX/LH2 engine: 55
Trajectory	Application of rocket equation with assumed total ΔV and losses based on the POST model, OTIS model, and previous analysis of analogous concepts using aerodynamics from wind tunnel, computational fluid dynamics, and APAS, and propulsion data from the REDTOP model, and data from existing systems
Carrier aircraft release conditions	Altitude: 35,000 ft. Mach number: 0.75
Simulation type	3 degrees of freedom, optimized trajectory with constraints
Angle of attack and dynamic pressure constraints	Maximum q: Less than 1,000 psf Maximum Alpha: Less than 15 deg Maximum q-Alpha: Less than 5,000 psf-deg

Acceleration constraints	Maximum wing normal factor: Less than 1.5g Maximum acceleration: Less than 5.0 g
Separation timing	10s delay after separation before engine ignition Minimum of a two second delay after rocket staging event before ignition of next stage engine Wing and tails dropped time chosen by optimizer, must be before first stage burnout and separation Payload fairing release when dynamic pressure falls below 0.1 psf
Orbit	Targeted direct injection into 100 nmi circular due east orbit from a latitude of 28.5 degrees
Aerodynamics	Based on Missile DATCOM analysis, with gradient-based optimization to reach sizing goals
Simulation parameters	Alpha range: -20 to 20 deg Mach range: 0.75 to 30
Wing sizing basis	300 lbm/ ft2 (gross weight wing area)
Release Conditions	Mach 0.7 25,000 ft Lift = 80% of weight at release Carrier aircraft flight conditions Flight path angle 5° Angle-of-attack 2° Rocket mounted at 6° to carrier body X-axis Total angle-of-attack 8°
Configuration	Wing mounted at 5° incidence to rocket body X-axis Horizontal tail control surface set to 50% of chord Vertical tail area and planform same as horizontal tails
Non-recurring costs	All costs in FY10 $M Derived from the TRANSCOST v.8.0 and NAFCOM cost estimating relationship (CER) models, analogies, and other references 90% learning rate applied to all new propulsion 95% production learning rate applied to all existing propulsion
Conventional industry wraps	Contractor fee: 10% Program support: 11% Contingency: 20% Vehicle integration: 4%
Aircraft DDT&E Cost All assumed to be at a TRL of 6; DDT&E costs included only TRL 6+ development	747-100 SCA: $10M (assumes existing structural interfaces) 747-400F: $144M An-225: $20M White Knight X: $125M Dual-fuselage C-5: $2.38B White Knight XX: $400M

Aircraft acquisition cost	747-100 SCA: - 747-400F: $30M An-225: $1.03B White Knight X: - Dual-fuselage C-5: - White Knight XX: -
Aircraft production or modification cost	747-100 SCA: $5M 747-400F: $62M An-225: $10M White Knight X: $50M Dual-fuselage C-5: $1.02B White Knight XX: $180M
Engine cost	Castor 120 unit cost: $8.7 M Castor 30 unit cost: $1.6 M Other proprietary engine costs
Recurring costs (assuming current launch practices)	Facilities cost: Derived from the Facilities and Ground support equipment and discrete event Operations Analysis (FGOA) model Operations cost: Custom implementation of TRANSCOST v8.0 CERs Operations Time Metrics: Based on historical analogies with multipliers
Baseline campaign parameters	Campaign Duration: 20 years Flights Rate: 6 flights per year
Recurring cost contingency	15%
Program management cost	20%
Reliability	Based on event sequence diagram with supporting fault tree analysis derived from compiled historical failure rates
Simulation type	Event sequence diagram supported by fault trees; failure database from historical analogies
Failure rate: booster separation from aircraft	747-100 SCA: 1 in 100 flights 747-400F: 1 in 100 flights An-225: 1 in 125 flights White Knight X: 1 in 1000 flights Dual-fuselage C-5: 1 in 667 flights White Knight XX: 1 in 667 flights
Failure rate: stage separation event	1 in 107 flights
Other event and subsystem failure rates	Based on restricted launch vehicle and aircraft historical data
Engine out capability	None (assume all engines required for all phases of flight)
Commercial viability	Modification of the Cost And Business Analysis Module (CABAM) for project net present value estimation and cash flow analysis

Simulation type	Discounted cash flow analysis supported by custom commercial and military launch demand model
Government contribution to DDT&E cost	$400M (FY10)
Launch market	Both commercial and military
Anticipated inflation rate	2.1%
Tax rate	30%
Flight rate	Calculated according to market demand and capture. Ranges between 3 to 7 flights per year based on payload class.

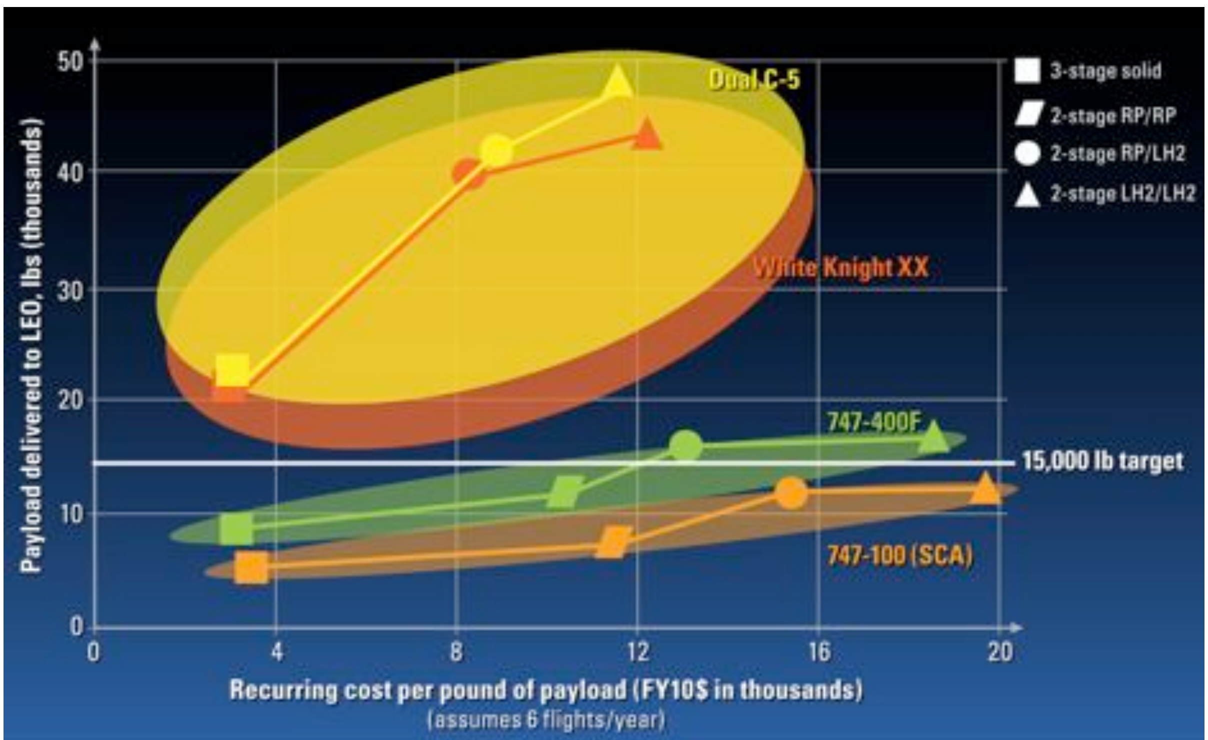


Figure 1. Recurring cost per pound of payload versus payload capability for four system concepts

To trade payload performance with development/operations cost and loss of mission probability, an initial conceptual "Level 0" analysis was used. (See Table 3 for a description of analysis fidelity levels.) This was embodied in an integration framework known as Reduced Order Simulation for Evaluation of Technologies and Transportation Architectures (ROSETTA).[1] ROSETTA is a design simulation that utilizes a multi-disciplinary design optimization modeling process and is intended to represent the design process for a specific system concept.

[1] A Crocker, A Charania, and J Olds, "An Introduction to the ROSETTA Modeling Process for Advanced Space Transportation Technology Investment," AIAA-2001-4625, August 2001.

ROSETTA uses StageSizer that uses historical regressions of subsystem weights based on NASA's Concept Sizer (CONSIZ)[2] and the Air Force Weights Analysis Tool (AFWAT)[3] to compute system bases that meets performance requirements and constraints. System performance is computed using the 3-degree-of-freedom Program to Optimize Simulated Trajectories (POST)[4] to compute the ΔV performance from due east separation to low Earth orbit. POST uses aerodynamic data generated from APAS,[5] using linear potential/ hypersonic impact methods, and propulsion characteristics from REDTOP, a chemical kinetics and cycle sizer.[6]

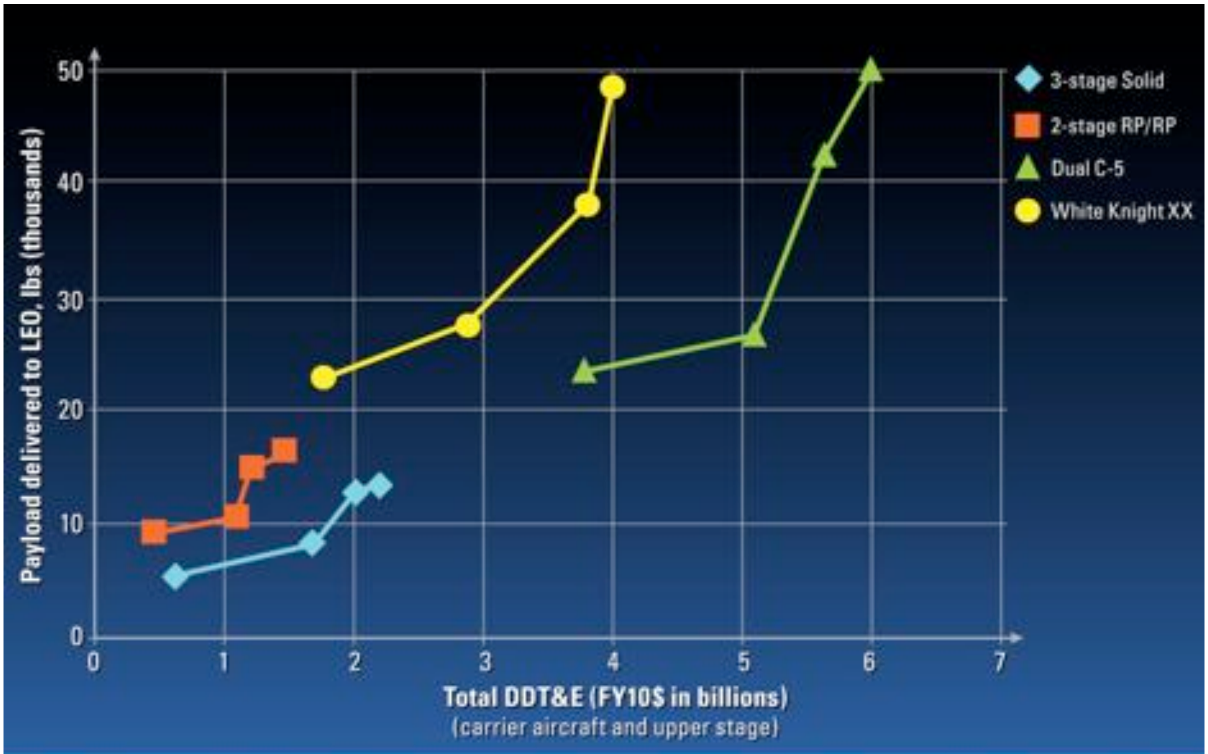


Figure 2. Cost of design, development, test, and evaluation versus payload capability for four system concepts

[2] RA Lepsch Jr, DO Stanley, CI Cruz and SJ Morris Jr, "Utilizing Air-Turborocket and Rocket Propulsion for a Single-Stage-to-Orbit Vehicle," Journal of Spacecraft and Rockets,Vol. 28, No. 5, 1991, pp. 560-566.

[3] AM Faulkner and C Cho, "AFWAT: Air Force Weights Analysis Tool; A Method of Calculating Reusable Launch Vehicle Weights", AFRL-PR-ED-TP-2005-167, June 2005

[4] RW Powell, et al, "Program to Optimize Simulated Trajectories (POST) Utilization Manual, Volume II, Version 5.2," NASA Langley Research Center, Hampton, VA; Martin Marietta Corporation, Denver, CO., October 1997.

[5] D Sova and P Divan, "Aerodynamic Preliminary Analysis System II, Part II", NASA CR 182077, April 1991.

[6] J Bradford, J Charania, and B St Germain, "REDTOP-2: Rocket Engine Design Tool Featuring Engine Performance, Weight, Cost, and Reliability," AIAA-2004-3514, July 2004.

Table 3. Analysis Requirements and Methodology Differences for Performance-Related Disciplines at Various Levels of Analytical Fidelity

Fidelity Level	Configuration, geometry and packaging	Structures and materials	Sizing and closure	Trajectory, GNC and simulation
0	Parametric, empirical or analytical geometry model	Parametric or historical equation adjusted to level 1 or higher for similar technology and vehicle configuration	Weight and volume closure w/ consistent bookkeeping of all propellants and fluids based on commensurate fidelity level inputs from other disciplines; as-flown vehicle photographic scale factor < +/- 15% from as-drawn	Rocket equation or energy methods(path following) simulation
1	External and major internal components modeled such as propellant tanks. Payload bay, propulsion, etc… for volume, area, and key linear dimensions	1D bending loads analysis based on structural theory of beams, shell, etc… with non-optimums based on level 2 or higher results	Weight and volume closure w/ consistent bookkeeping of all propellants and fluids based on commensurate fidelity level inputs from other disciplines;as-flown vehicle photographic scale factor < +/- 10% from as-drawn	Optimized ascent, flyback and re-entry 3-degrees of freedom point mass simulation (untrimmed)
2	All components modeled, packaged, and analyzed for geometric properties including center of gravity. Geometry redrawn and packaged to match closure model	Limited 3D FEA (<20,000 nodes) for all major load cases, structure sized to allowables, non-optimums determined empirically or analytically	Weight and volume closure w/ consistent bookkeeping of all propellants and fluids based on commensurate fidelity level inputs from other disciplines; as-flown vehicle photographic scale factor < +/- 5% from As-Drawn	Optimized ascent, flyback and re-entry 3- degree of freedom (pitch trim) point mass simulation; longitudinal stability and control evaluation
3	All components modeled, packaged, and analyzed for geometric properties including center of gravity and inertia characteristics. Geometry re-drawn and packaged to match closure model	3D FEA (>20,000 nodes) for all major load cases, structure sized to allowables, nonoptimums determined empirically or analytically. Dynamic frequencies estimated.	Weight and volume closure w/ consistent bookkeeping of all propellants and fluids based on commensurate fidelity level inputs from other disciplines; As-Flown vehicle photographic scale factor < +/- 3% from As-Drawn	Optimized ascent, flyback and re-entry 6-degree of freedom simulation; longitudinal, lateral and yaw stability and control evaluation; perfect GNC
4	All components modeled, packaged, and analyzed for geometric properties including center of gravity and inertia characteristics. Geometry re-drawn and packaged to match closure model	3D FEA (>100,000 nodes) for all major load cases, structure sized to allowables, nonoptimums determined empirically or analytically. Dynamic frequencies estimated.	Weight and volume closure w/ consistent bookkeeping of all propellants and fluids based on commensurate fidelity level inputs from other disciplines; as-flown vehicle photographic scale factor < +/- 1% from as-drawn	Optimized ascent, flyback and re-entry 6- degree of freedom simulation; longitudinal, lateral and yaw stability and control evaluation; real GNC w/ gain scheduling (or similar) lags, noise, etc
Fidelity Level	**Propulsion design and performance**	**Aerodynamics and aerotherodynamics**	**Aerothermal and TPS sizing**	**Airframe and engine subsystems**
0	Scaled empirical	Scaled empirical	Parametric or historical	Parametric or historical
1	1D cycle analysis adjusted to level 2 or higher results (military standard or other installation effects included)	Linear/impact methods with all drag increments (empirical) adjusted to level 2 or higher; vehicle satisfies all takeoff/landing speeds, glide path, and runway length requirements	Aerothermal loads based on 1D engineering methods; 1D thru the thickness TPS sizing	Functional definition and evaluation and/or 1D or generic modeling of subsystem
2	2D/3D finite difference inviscid (Euler) flowfield analysis w/ heat conduction / transfer and integral boundary layer analysis. Propulsive moments, installation effects and thermal balance computed.	3D CFD inviscid (Euler) w/ integral boundary layer or potential w/ semi-emperical drag increments or thin layer Navier Stokes w/ semiemperical non-viscous drag increments; vehicle satisfies all takeoff/landing speeds, glide path, runway length, and longitudinal stability requirements	2D/3D Engineering methods or CFD based aerothermal loads w/ quasi-2D TPS sizing	Quantitative thermal and fluid analysis of subsystem; Component Weights estimated w/ empirical, historical or analytical data/analysis
3	2D/3D parabolized Navier-Stokes finite difference / volume flowfield analysis w/ heat conduction / transfer and integral boundary layer analysis. Propulsive moments, installation effects and thermal balance computed. Full mechanical design.	3D CFD parabolized Navier-Stokes (PNS) finite difference / volume flowfield analsis w/ heat conduction / transfer and integral boundary layer analysis; vehicle satifies all takeoff/landing speeds, glide path, runway length, and longitudinal, lateral and yaw stability requirements	2D/3D CFD Based aerothermal loads w/ quasi-2D TPS sizing	Quantitative thermal and fluid analysis of subsystem; Component weights estimated w/ empirical, historical or analytica data/analysis
4	3D full or thin-layer Navier-Stokes (FNS or TLNS) flowfield analysis including pressure feedback, shear stress and heat transfer effects computed directly. Propulsive moments, installation effects and thermal balance computed. Full mechanical design.	3D CFD full or thin layer Navier-Stokes (FNS or TLNS) flowfield analsis including pressure feedback, shear stress and heat transfer effects computed directly; vehicle satisfies all takeoff/landing speeds, glide path, runway length, and longitudinal, lateral and yaw stability requirements	3D CFD based aerothermal loads w/ 3D TPS sizing	Quantitative thermal and fluid analysis of subsystem; Component weights estimated w/ empirical, historical or analytical data/analysis

Thus, ROSETTA assumes an idealized system where the system including propulsion is exactly sized to meet the mission requirements (i.e. rubberized). The cost of a theoretical first unit, along with DDT&E and operations costs are computed using TRANSCOST.[7] Facilities cost are derived from the Facilities and Ground Support Equipment and discrete event Operations Analysis (FGOA)[8] model and reliability is derived from event failure trees and historical component rates. A design-of-experiments method was used to generate response surfaces representative of the tool outputs over the range of expected geometry, propulsion, aerodynamics, trajectory performance, and subsystem weights in order to dramatically decrease the time required to evaluate the concept matrix while maintaining acceptable accuracy. Table 3 provides the assumptions for the range of analysis methods.

ROSETTA was developed to quickly evaluate carrier aircraft and launch vehicle options to identify promising combinations. A matrix of 1,365 combinations was defined that included variables such as carrier aircraft, stage expendability/reusability, propellants, and number of stages and engines. In carrying out the concept analysis, ROSETTA was used to specify the design of each propulsion systems to meet all thrust constraints. These cases were considered new rather than existing designs, and therefore required similar development and acquisition costs for a new system.

Subsonic Carrier Aircraft Options

Characteristics of a number of existing and new large subsonic carrier aircraft are utilized in ROSETTA. The most widely available of these is the Boeing 747-400F, the all-freight version of the commercial airliner that entered into service in 1993. It has a stated maximum payload over 308,000 lb. The Airbus A380-800F is another commercially available airliner. The A380 is a wide-body aircraft with an upper deck that extends along the entire length of the fuselage. It has a stated maximum payload over 320,000 lb.

Several unique carrier aircraft options are also available in ROSETTA. The Antonov An-225 (Mriya) is a Ukranian-built strategic airlift cargo aircraft. It was designed in the 1980s to ferry the Soviet Buran orbiter. It is the world's heaviest aircraft. The NASA Shuttle Carrier Aircraft (747 SCA) is a purpose-modified Boeing 747-100 for piggy-back ferrying of launch vehicles, including the Space Shuttle orbiter.

Designs for modified carrier aircraft are also included, such as a dual-fuselage variant of the C-5 Galaxy strategic airlift carrier aircraft. Two additional derived designs are the White Knight X and White Knight XX, enlarged variants based on the Scaled Composites White

[7] D Koelle, "Handbook of Cost Engineering for Space Transportation Systems including TRANSCOST 8.0", Report No. TCS-TR-190, 2010.
[8] D DePasquale and A Charania, "I-RaCM: A Fully Integrated Risk and Life Cost Model, " AIAA-2004-3514, September 2008.

Knight Two. The X variant is as large again as White Knight Two versus White Knight, and the XX variant is scaled to carry 750,000 lbm.

Launch Vehicle Options

The first decision to be made regarding the launch vehicle is the placement on the carrier aircraft, and a range of configurations are available to the designer. Launch vehicles may be carried externally on the top or on bottom of the carrier aircraft, stored internally, or towed. All have advantages and disadvantages, discussed as follows:

- Internally stowed launch vehicles have the highest and fastest staging condition for a given carrier aircraft; however, the size of the launch vehicle size is severely limited by the internal payload configuration of the carrier aircraft. The carrier aircraft must be rear-loaded for separation and the launch vehicle may need deployable aerodynamic surfaces.
- Towed launch vehicles have the fewest modifications and the easiest separation conditions, but require attachments and wings designed for takeoff, attachments for the dropped takeoff gear, and must be designed for the dynamic loads from the tow line. Towing offers larger payloads than internal stowing, and could achieve the HLS goal of 15,000 lbs of payload to orbit in some configurations.
- Top-mounted launch vehicles on commercially-available carrier aircraft can carry up to 50,000 pounds of payload to due-east orbit.
- Bottom-mounted launch vehicles such as the Pegasus rocket on the Lockheed L-1011 Stargazer, are limited by ground clearance that restricts the size of the launch vehicle. A dramatic advantage is presented by high-wing dual-fuselage designs, such as a dual-fuselage C-5 or White Knight carrier aircraft, which can carry a launch vehicle bottom-mounted on the center wing. This configuration can be tailored to meet almost any payload requirement and a wide range of ascent trajectories. These advantages are offset by the need to develop and operate a one-of-a-kind carrier aircraft, and limits on basing flexibility due to the wingspan and associated takeoff and landing gear of twin fuselage aircraft.

In ROSETTA, launch vehicles may have one, two, or three engines or motors, may be either expendable or reusable, and operate with or without a drop tank. Launch vehicles are optimized within size and gross weight constraints depending on the carrier aircraft, but are not constrained to existing available engines and solid motors. System concept solutions are tailored to match the maximum load carrying capability of the carrier aircraft, and the payload capability is therefore maximized assuming a packaging density of 8 pounds per cubic foot.

Propellant options included solid rockets, liquid engines with liquid oxygen (LOX), rocket propellant (RP), liquid hydrogen (LH2), or combinations of these. Because liquid engines may have different basing flexibility and development risks as compared to solid rockets, some characteristics are weighted differently in the model.

Results of the Screening Analysis

Table 4 tallies the 1,365 system concepts that were analyzed based on combinations of available carrier aircraft and propulsion components; 94 percent of these yielded positive payload delivery. The remaining 6 percent did not achieve performance closure owing to low specific impulse and too few stages, i.e., a two-stage solid launch vehicle. The analysis demonstrates that existing carrier aircraft can deliver launch vehicles with useful payloads exceeding 15,000 lbs and as high as 20,000 lbs for an existing, older 747-400F aircraft, and up to 52,000 lbs for the dual-fuselage C-5 design. Preliminary separation analysis resulted in no major issues.

Table 4. System Concept Configurations Examined in the Study

Carrier aircraft	Cases analyzed	Cases with positive payload	Launch vehicle gross weight available (pounds)	Maximum LEO payload (pounds)
White Knight X	195	180	176,000	11,180
747-100 (SCA)	195	185	240,000	15,440
A380	195	184	264,550	17,090
747-400F	195	185	308,000	20,000
An-225	195	185	440,925	30,380
White Knight XX	195	185	750,000	49,940
Dual-fuselage C-5	195	185	771,618	52,290
Total	**1,365**	**1,289**		

Based on these initial results, the following three carrier aircraft were not continued for further analysis for the following reasons:

- White Knight X – low payload capability compared to existing commercial aircraft
- An-225 – only one unit currently exists, and the risks of purchasing and maintaining such a unique specimen were very high
- A380 – duplication of the 747 capabilities and more expensive to acquire

Characteristics of the carrier aircraft included for further analysis are listed in Table 5, and the relationships between payload and cost for the array of configurations are shown in Figures 1 and 2. It is of interest to note that the system concepts with three-stage solid rockets generally had the lowest recurring cost per pound of payload for all systems and the lowest development costs, and the system concepts with two-stage all LOX/LH2 engines generally had the highest payload capability and the highest development costs. Specifically, the payload capability of a 747-400F ranges from 9,000 lbs with solid rockets to 16,000 lbs with liquid engines. A 747-400F with a two-stage all LOX/RP launch vehicle approaches the 15,000 lb mark while avoiding the operational complexities of storage and handling of liquid hydrogen.

Table 5. System Concepts Results

Carrier aircraft	747-400F	747-400F	747-100 SCA	White Knight XX
Launch vehicle	3-stage expendable solid	2-stage expendable LOX/RP+LOX/LH2	2-stage expendable solid	3-stage expendable solid
Payload	9,390 lbs.	16,210 lbs.	5,540 lbs.	23,060 lbs.
Recurring cost/lb	$4,730	$14,739	$5,885	$2,980
DDT&E	$1B	$2.1B	$0.7B	$1.8B
Military utility	moderate	high	low	high
Mission flexibility	high	high	high	high
Basing flexibility	high	moderate	high	low
Development risk	low	moderate	low	moderate

The analysis of the dual-fuselage design options presented a a wider spread of results with greater uncertainty. A point to note is the lower cost of the White Knight XX versus the dual-fuselage C-5, in spite of the fact that the C-5 is an existing production aircraft. This cost assumption was made based on the development costs of the White Knight as compared to similar commercial and military aircraft development.

III. Point Design Vehicle Analysis

Several point design vehicles (PDVs) were derived from the vehicle concept exploration as example system concepts intended to validate the designs at the next level of analysis fidelity, Level 1 (See Table 3). The analyses optimized the various design parameters in order to maximize the payload delivered to orbit, and determined the best existing rocket motor or engine and other critical subsystems to reduce the development risk and uncertainty of system weight and cost predictions.

Based on the previous analyses, the 747-400F was selected as the carrier aircraft due to its availability, acquisition cost, and the capability to deliver over 15,000 pounds of payload to orbit. The four configurations analyzed were:

- PDV-1 747-400F with 3-stage solid launch vehicle
- PDV-2 747-400F with 2-stage liquid LOX/RP+LOX/LH2 launch vehicle
- PDV-2M 747-400F with 2 stage liquid LOX/RP launch vehicle with aerial fueling
- PDV-3 747-400F with 2-stage liquid LOX/LH2 launch vehicle

PDV-1 was selected to represent the lowest DDT&E costs, and PDV-3 for the highest payload. The other two vehicles, PDV-2 and PDV-2M, were selected as compromises between performance and cost.

To select the best existing rockets and other subsystems while optimizing the payload of the system, an array of analysis tools were integrated into a framework to link control variables such as thrust-to-weight ratio, wing loading, and diameter, as seen in Figure 3. A parametric geometry model scaled the wing geometry based on wing loading constraints, stage length and diameter based on propellant requirements, and carrier aircraft constraints. Aerodynamics were predicted using the Missile DATCOM code and the Supersonic/Hypersonic Arbitrary Body Program (SHABP).[9] Vehicle weights were predicted using physics-based models for structures and historically-driven weight estimating relationships for subsystems that were not selected from the list of existing subsystems. Aerodynamics, rocket performance, and system weight were communicated to the Program to Optimize Simulated Trajectories (POST) program to maximize payload. If the launch vehicle did not meet all the constraints, the vehicle geometry (diameter and length), aerodynamic surfaces, and thrust were resized until the payload was optimized and all constraints were met.

[9] G Gentry, D Smyth, and W Oliver, "The Mark IV Supersonic-Hypersonic Arbitrary-Body Program", AFFDL-TR-73-159, Vols I-III, Nov. 1973.

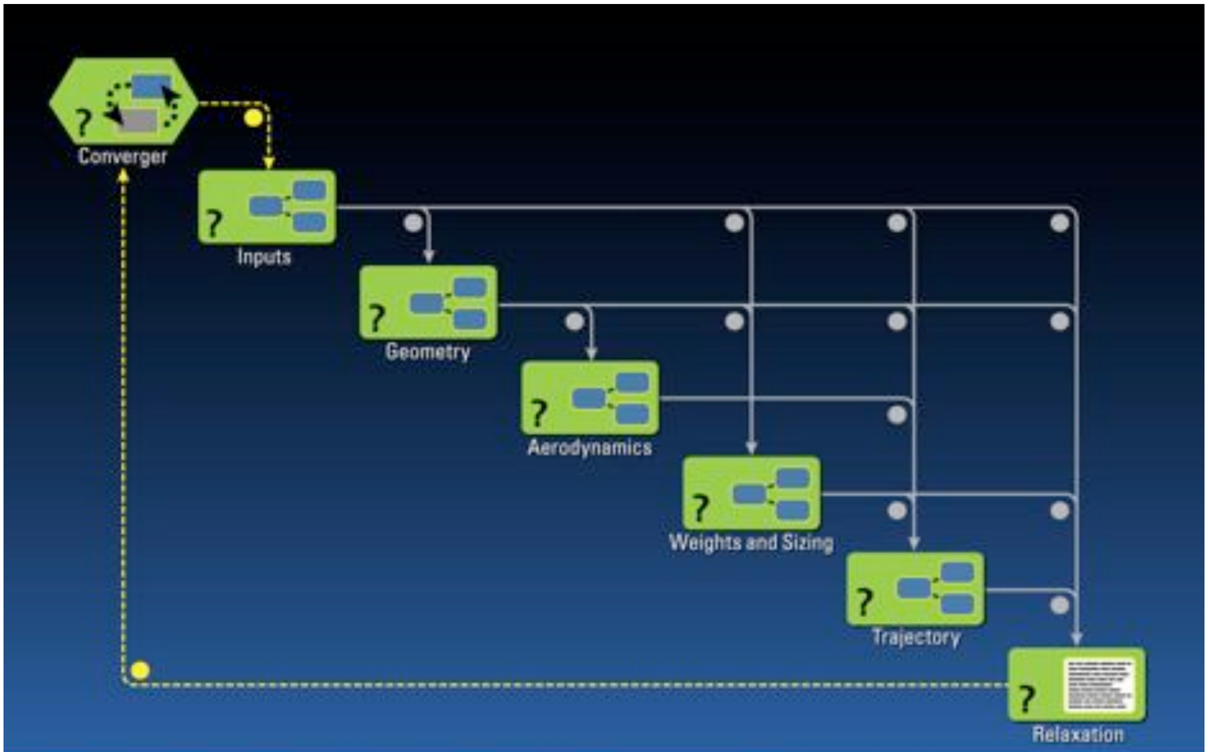


Figure 3. Analysis framework used to communicate control of variables

Costs were calculated for each of the point design vehicle system concepts using the NAFCOM program to obtain DDT&E and theoretical first unit (TFU) costs.[10] The Facilities and Ground Support Equipment and discrete event Operations Analysis (FGOA) code was used for facilities cost and to generate event sequence diagrams and historical failure rates for reliability and mission success. Market projections over the next decade were used to set six flights per year at 15,000 lbs of payload, and an additional three flights at 5,000 and 10,000 lbs, assuming dual manifesting.

Point Design Vehicle 1

PDV-1 consists of the 747-400F carrier aircraft and a three-stage solid rocket launch vehicle, as seen in Figures 4 and 5. The solid rocket motors selected for the maximum payload are two Castor 120 motors for stages one and two and a Castor 30 motor for stage three. The wing and empennage are attached to the first stage with a winged "strongback", a non-integral structural interface connecting the aerodynamic surfaces to the launch vehicle. The intertanks, interstages, and aerodynamic surfaces are made with graphite/epoxy composite materials. Power and attitude control subsystems are based on the Lockheed Athena orbit adjust module.

[10] C Smart, "NAFCOM Cost Risk Module", The NASA Engineering Cost Group and Science Application International Corporation, Huntsville, AL., pages 2-8, 2004.

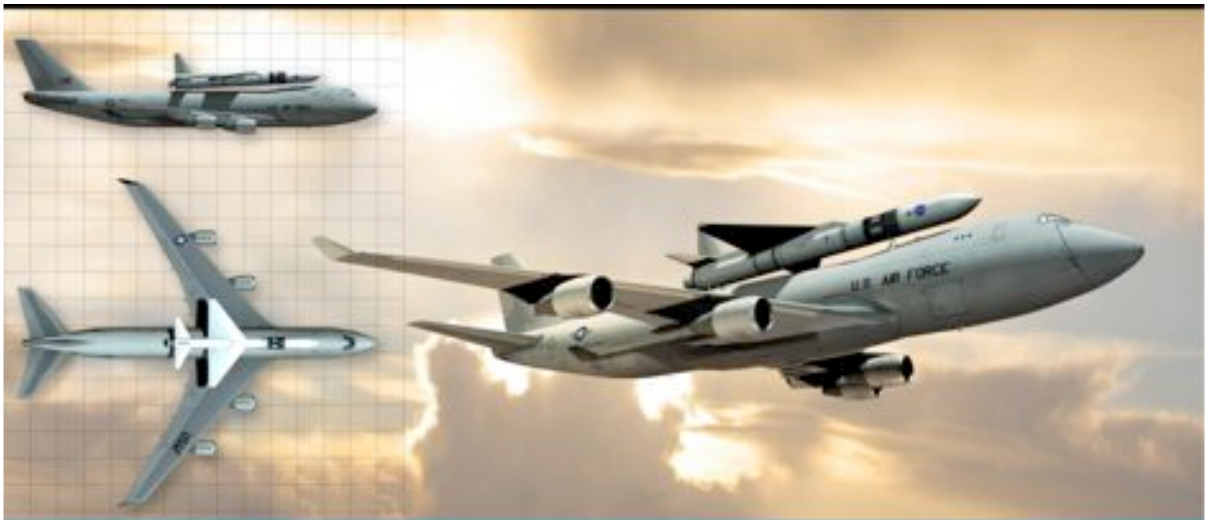


Figure 4. Configuration of the PDV-1 system concept

Figure 5. Expanded configuration of the PDV-1 system concept

With the selection of the three existing solid rocket motors, the optimized gross weight of PDV-1 was less than the maximum the 747-400F could carry, as seen in Table 6. This allowed a reduction in the internal structural modifications and lowered the development costs of the 747 significantly.

The schematic in Figure 6 and dimensions in Table 7 were used for aerodynamic analysis of PDV-1. The resulting trajectory shows the launch vehicle separates from the carrier aircraft at Mach 0.7 with a flight path angle of 5 degrees and an angle of attack at 8 degrees, as seen in

Table 8. The dynamic pressure is 270 psf at separation and the wing area is sized for a nominal wing loading of 300 psf. Once the launch vehicle attains a typical vertical flight profile, the aerodynamic surfaces are jettisoned. The launch vehicle reaches a maximum dynamic pressure of 835 psf at moderate angle of attack which is similar to most launch vehicles.

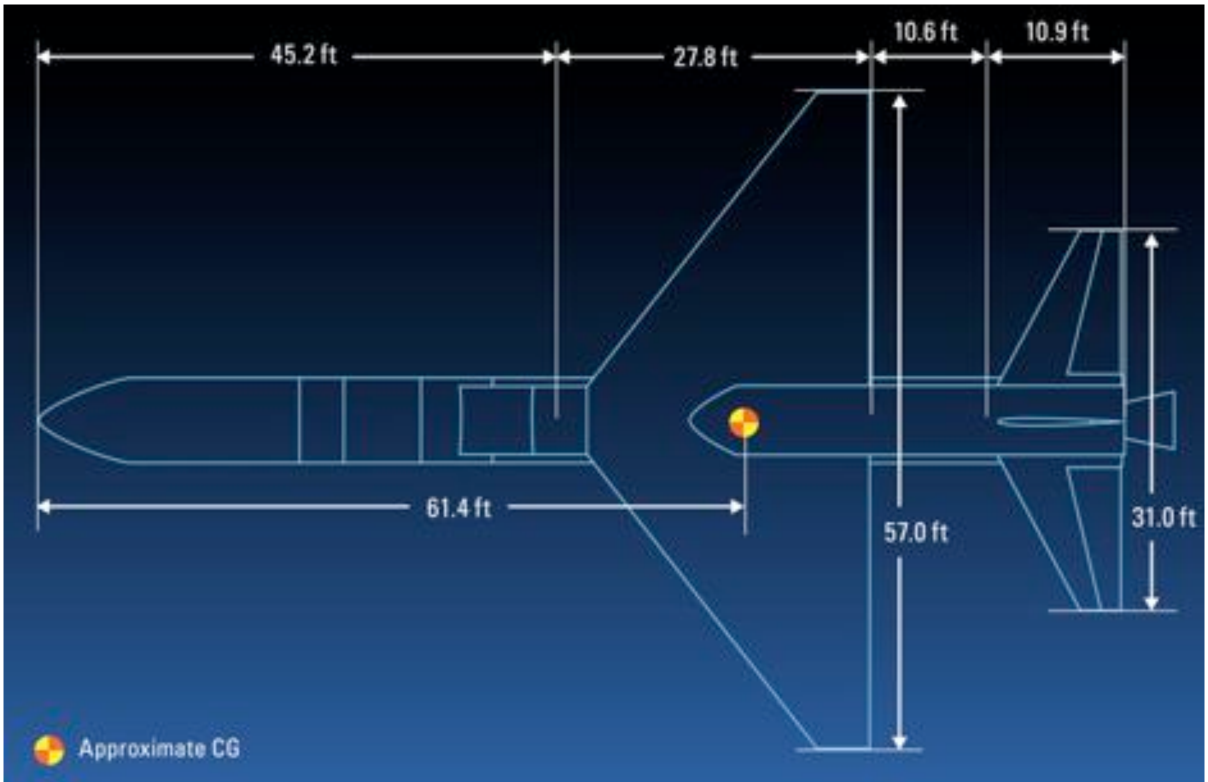


Figure 6. Schematic used in aerodynamic analysis for the PDV-1 system concept

Table 6. Summary Dimensions and Weights of the PDV-1 System Concept

Carrier aircraft	747-400F (modified)
External Payload Capacity	290,000 lbs.*
Total Length	231 ft.
Fuselage Diameter	21 ft.
Wing Span	211 ft.
Launch Vehicle	3-stage-solid
Total Gross Weight	288,483 lbs.†
Payload to LEO	5,662 lbs.
Total Length	100 ft.
Maximum Diameter	7.75 ft.
Wing Span	57 ft.

* Reduced external carriage capacity results in lower aircraft DDT&E
† Includes dry mass margin

Table 7. Dimensions Used in Aerodynamic Analysis for the PDV-1 System Concept

Parameter	Wing	Horizontal/Vertical Tails (each)
Aspect Ratio	3.5	4.0
Taper Ratio	0.17	0.43
LE Sweep Angle	38.5°	21.8°
Planform Area	923 ft^2	120 ft^2
Thickness-to-Chord Ratio	0.1	0.1
Wing Loading at 1g	310 lbs/ft^2	
Wing Incidence Angle	5°	

Although the integrated aerodynamic surface module is more than two times the payload weight, this module is jettisoned early in the trajectory when the launch vehicle reaches the appropriate flight path angle and thus has only a few hundred pounds of impact on the payload delivery capability. The payload delivered by PDV-1 was computed to be 5,662 lbs, as shown near the bottom of Table 9.

Using current integration and checkout practices for launch vehicles, the crew size was minimized to turn the vehicle around in 36 hours, as shown in Figure 7, with a margin of 20 percent with a crew of 21 technicians.

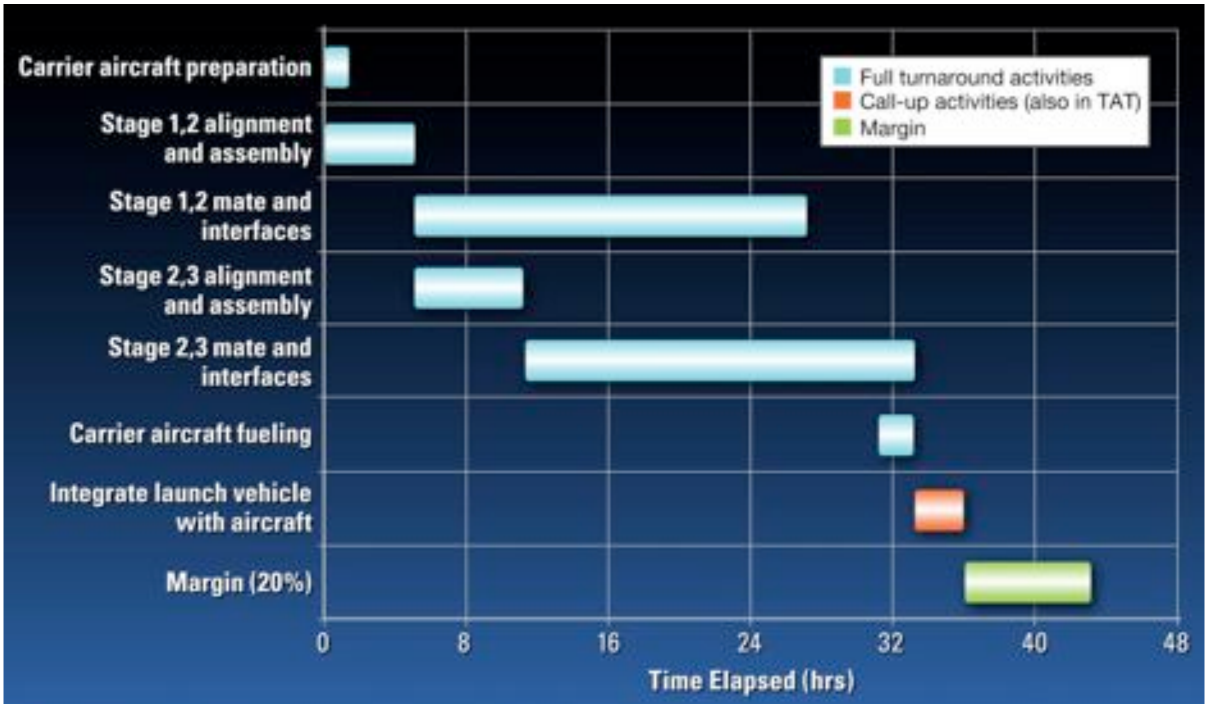


Figure 7. Results of the operational analysis for the PDV-1 system concept

Using failure rates of existing systems and the reliability exponential growth history of past systems, the loss of mission (LOM) probabilities were found to be 6.5 percent for the fourth flight, and 1.6 percent for the 16th mission, shown in Table 10. Note this LOM prediction does not include the two recent consecutive payload fairing separation mishaps of the Taurus rocket. As shown in the list of important elements, there are no dominating unreliable components.

Program costs, listed in Table 11, were estimated using a number of assumptions. Chief among these was the acquisition and modification of a used 747-400F for $86 million. This was based on current market price and past Boeing AirLaunch studies. Market price was also used for the solid rocket stages and included a 15 percent cost for modifications. Development costs of the strongback aerosurface module were based on traditional aerospace practices modeled with the NAFCOM model; however recent commercial efforts have reported cost factors 3 to 8 times lower than the model predicts. In addition, the typical government oversight costs for the program are based on previous manned system development, as are the use of government facilities (and their associated costs) for testing and demonstration. With a nominal 20 percent contingency, the total cost per flight and cost per pound of payload were calculated to be $51 million and $8,934 per pound of payload, respectively.

Point Design Vehicle 2

The second point design vehicle, PDV-2 as shown in Figure 8, consists of the 747-400F carrier aircraft and a two-stage launch vehicle with a LOX/RP first stage and a LOX/LH2 second stage.

Figure 8. Design of the PDV-2 system concept

The PDV-2 configuration in Figure 9 reveals three LOX/RP Merlin 1C engines from SpaceX on the first stage. The second stage has three RL10A-4-2 LOX/LH2 engines from Pratt&Whitney Rocketdyne. The wing and empennage are attached to the first stage with a strongback. All interstages, fairings and aerodynamic surface are graphite/epoxy composite materials, as shown in Figure 10.

The summary dimensions and weights of the PDV-2 in Table 12 show that the selection of the liquid engines brings the system closer to the maximum external payload limit of the 747-400F. The gross weight of the launch vehicle equals 308,000 lbs (minus 3,000 lbs of integration weight). The dry weight also includes a 15 percent weight growth margin. The schematic in Figure 11 and dimensions in Table 13 were used for aerodynamic analysis of PDV-2 using the DATCOM code. The results are shown in Figure 12.

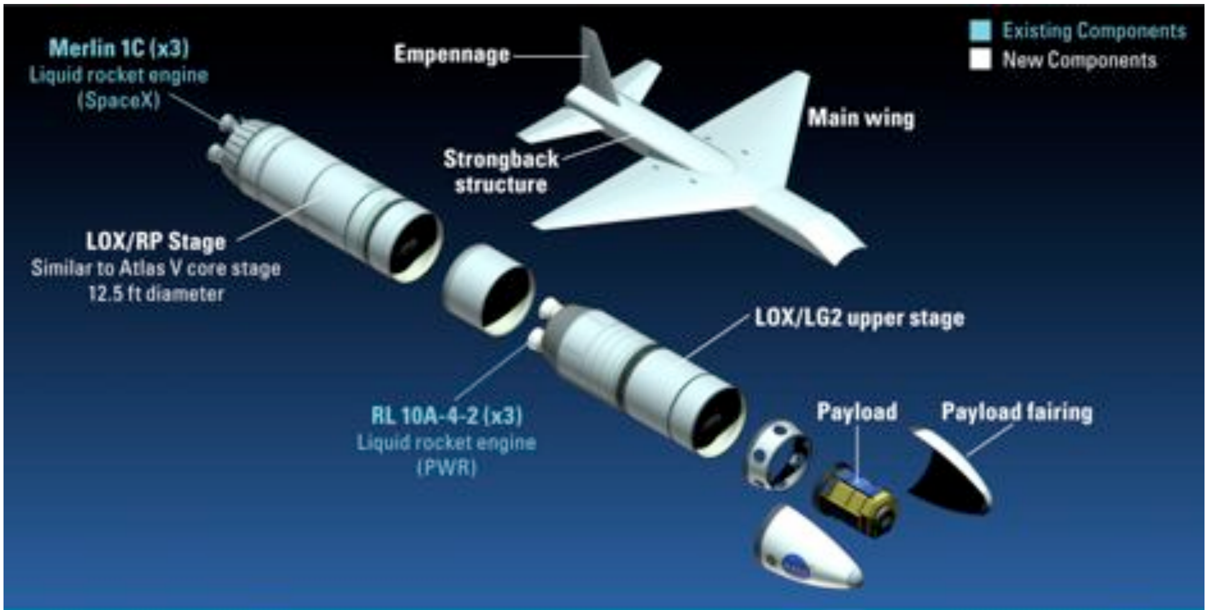


Figure 9. Expanded configuration of the the PDV-2 system concept

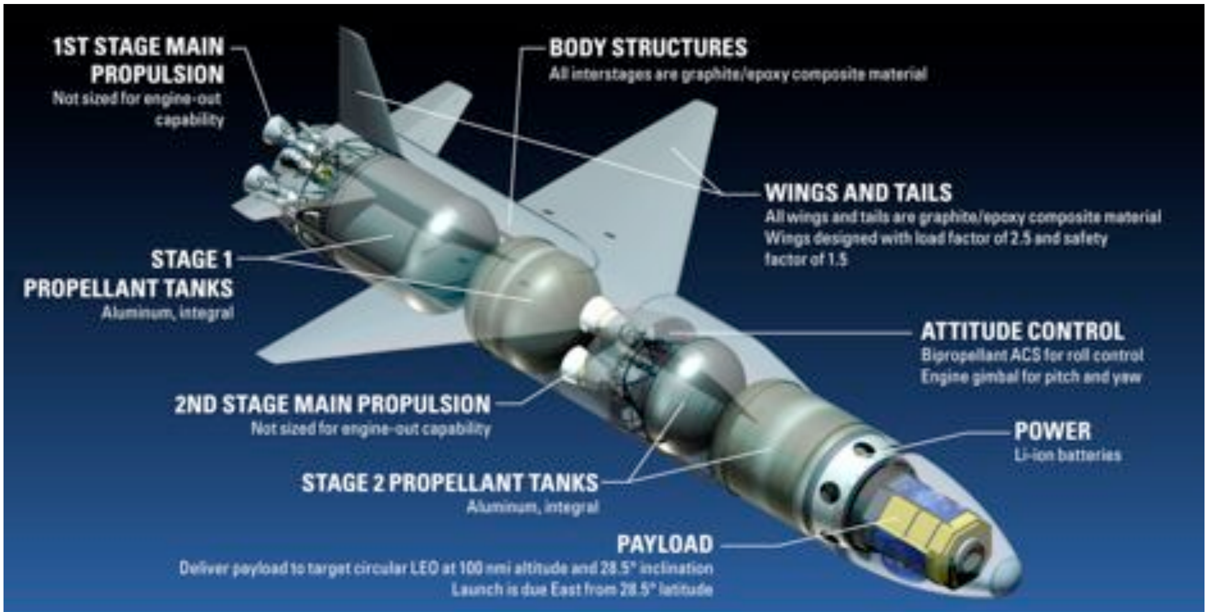


Figure 10. Materials and systems in the PDV-2 system concept

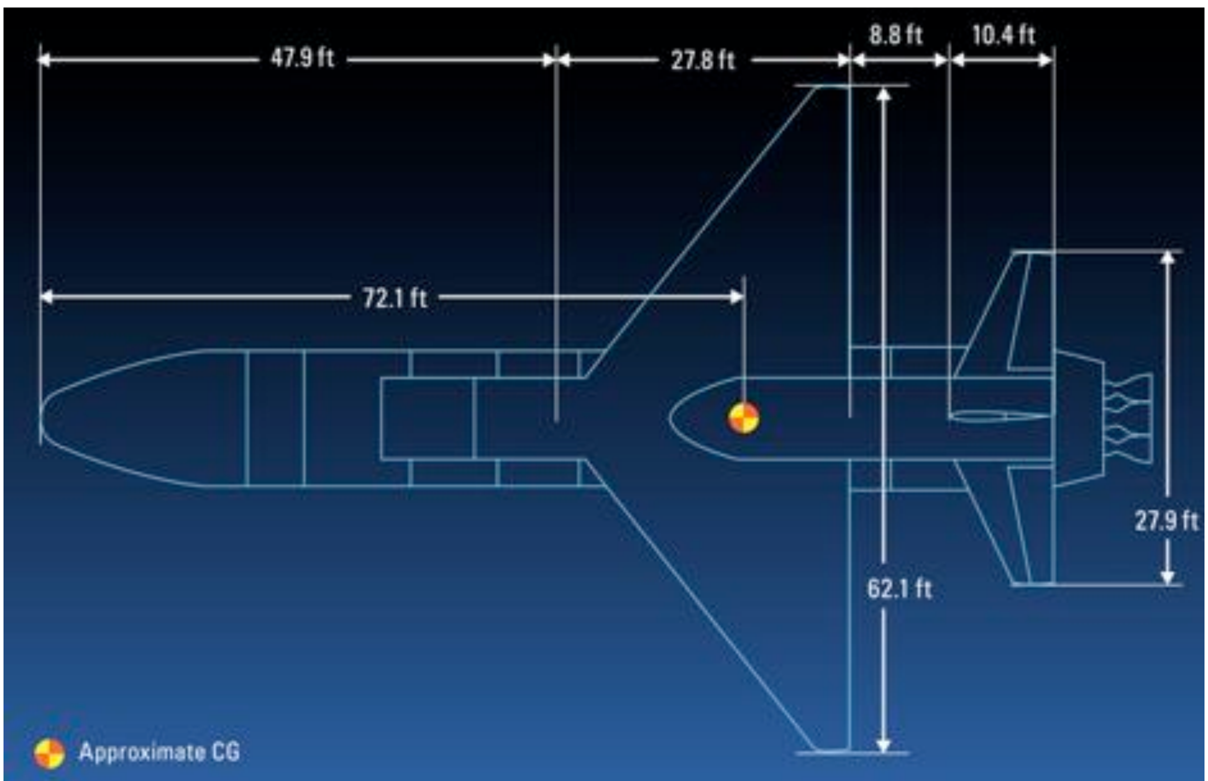


Figure 11. Schematic used in aerodynamic analysis for the PDV-2 system concept

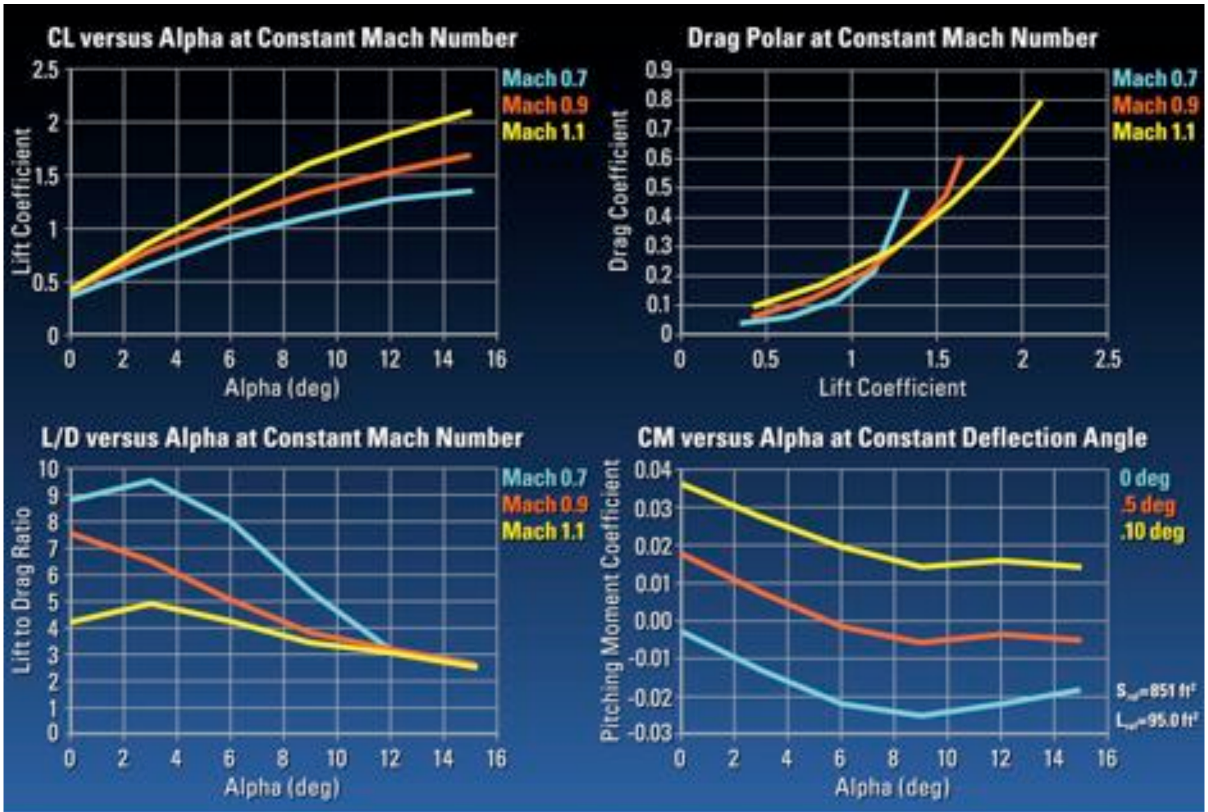
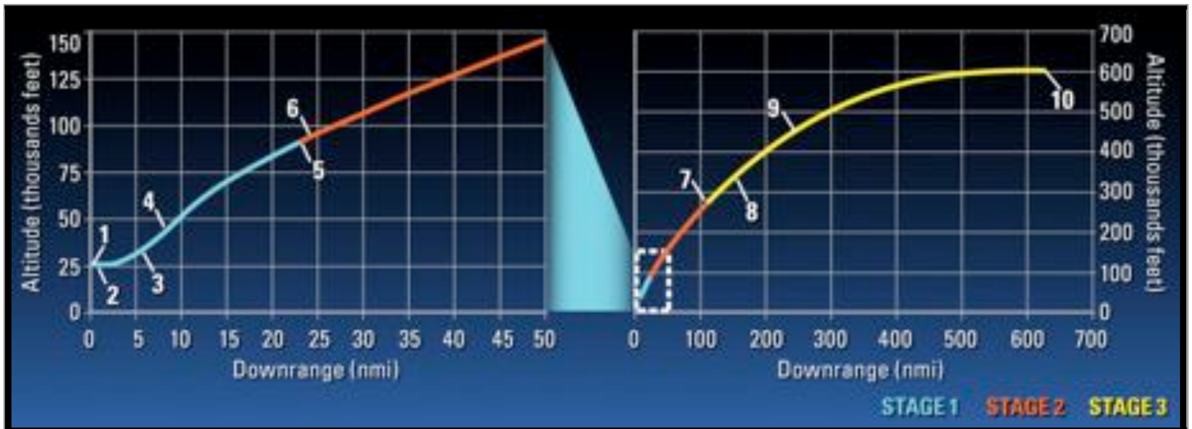


Figure 12. Aerodynamic analysis forthe PDV-2 system concept

Table 8. Trajectory Analysis for the PDV-1 System Concept



ID	Event	Time (s)	Weight (lbm)	Altitude (ft)	Velocity (fps)	Mach Number	Dynamic Pressure (psf)	Gamma (deg)	Alpha (deg)
1	Aircraft separation	0	288,490	25,000	711	0.7	270	5.0	8.0
2	Stage 1 ignition	10	288,490	25,244	674	0.7	240	-1.4	8.0
3	Maximum dynamic pressure	38	248,198	33,459	1,460	1.5	835	32.7	6.0
4	Aerosurface jettison	47	234,596	41,864	1,668	1.7	747	39.0	6.0
5	Stage 1 burnout and separation	89	168,166	90,407	3,298	3.3	283	21.1	11.8
6	Stage 2 ignition	92	158,077	93,288	3,297	3.3	247	20.1	12.1
7	Stage 2 burnout and separation	171	50,117	263,736	12,606	13.6	2.7	16.5	6.3
8	Fairing jettison	190	40,073	327,786	12,446	13.9	0.1	14.7	3.9
9	Stage 3 ignition	235	38,650	446,839	12,162	13.6	0.0	10.1	-1.4
10	Stage 3 burnout and separation	376	10,396	605,877	24,189	27.0	0.0	0.0	-10.5

Table 9. Weight Breakdown Statement for PDV-1

Item	Weight (lbm)
STAGE 1: CASTOR 120	
Motor Inert Mass	8,976
Subsystems	249
Interstage	747
Propellant	108,038
Stage 1 Loaded Mass	118,010
STAGE 2: CASTOR 120	
Motor Inert Mass	8,976
Subsystems	354
Interstage	637
Propellant	108,038
Stage 2 Loaded Mass	118,004
STAGE 3: CASTOR 30	
Motor Inert Mass	2,698
Subsystems	128
Propellant	28,300
Stage 3 Loaded Mass	31,126
AEROSURFACE MODULE	
Wing	6,833
Fins	2,032
Actuators	1,276
Strongback Structure	2,261
Aerosurface Module Total Mass	12,402
CONTROL MODULE	
Structures and tanks	469
Subsystems	263
Reaction control systems	206
Propellant	925
Control Module Total Mass	1,863
Fairing and adapter	1,423
Payload	5,662
Total Launch Vehicle Gross Mass	288,490

Table 10. Loss of Mission (LOM) Probabilities for the PDV-1 System Concept

End state values	4th flight		16th flight	
	Probability	Mean flights between failure	Probability	Mean flights between failure
Stage 1 LOM contribution	2.07%	48	0.50%	199
Stage 2 LOM contribution	1.35%	74	0.32%	300
Stage 3 LOM contribution	2.09%	48	0.50%	191
Takeoff through rocket release LOM contribution	1.00%	101	0.24%	419
Total LOM	6.52%	15	1.56%	64
747-400F Loss of Vehicle	0.0058%	17,241	0.0014%	71,428

Most important elements		Probability	Importance
1	Off-nominal orbit insertion	1.0%	0.208
2	Stage 1 castor 120 off-nominal performance	0.67%	0.139
3	Expendable aerosurface separation	0.67%	0.139
4	Stage 2 castor 120 off-nominal performance	0.67%	0.083
5	Stage 3 castor 30 off-nominal performance	0.67%	0.083
6	Fairing separation	0.28%	0.058
7	Stage 2 separation	0.23%	0.048
8	Payload separation	0.23%	0.048
9	Stage 1 separation	0.23%	0.048
10	Human error	0.16%	0.032

Table 11. Projected Costs for the PDV-1 System Concept, in FY10 Dollars, Assuming 6 Flights Per Year

DDT&E and Facilities Costs	
747-400F	$122 M
Stage 1	$48 M
Stage 2	$48 M
Stage 3	$13 M
Aerosurfaces	$104 M
Attitude control system and fairing	$31 M
Facilities and ground service equipment	$109 M
Subtotal for DDT&E and facilities costs	$475 M
Acquisition and Production	
747-400F Acquisition and modifications	$86 M
Subtotal reusable first unit costs	$86 M
Stage 1 average production cost	$13 M
Stage 2 average production cost	$13 M
Stage 3 average production cost	$2 M
Aerosurfaces average production cost	$10 M
Attitude control system and fairing average production cost	$3 M
Subtotal expendable average production cost	$41 M
Total recurring cost per pound of payload	$8,934 / lb
Total recurring cost	$51M / flight

Table 12. Summary Dimensions and Weights of the PDV-2 System Concept

Carrier aircraft	747-400F (modified)
External payload capacity	305,000 lbs.
Total length	231 ft.
Fuselage diameter	21 ft.
Wing span	211 ft.
Launch vehicle	2-stage liquid LOX/RP+LOX/LH2
Total gross weight	305,000 lbs.†
Payload to LEO	12,575 lbs.
Total length	102 ft.
Maximum diameter	12.5 ft.
Wing span	62 ft.

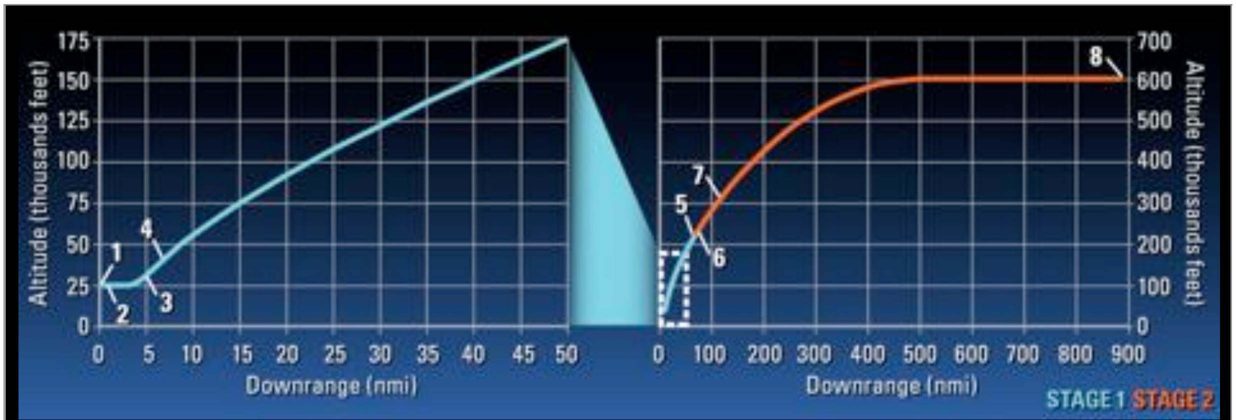
†Includes dry mass margin

Table 13. Dimensions Used in Aerodynamic Analysis for the PDV-2 System Concept

Parameter	Wing	Horizontal/vertical tails (each)
Aspect Ratio	4.1	4.0
Taper Ratio	0.09	0.34
LE Sweep Angle	38.3	26.2
Platform Area	940 $ft.^2$	97 $ft.^2$
Thickness-to-chord ratio	0.1	0.1
Wing Loading at 1g	320 $lbs./ft.^2$	
Wing Incidence Angle	5°	

Using the thrust histories of the liquid rocket engines and the launch vehicle aerodynamics, the resulting POST trajectories were calculated. The launch separation state for the launch vehicle is at Mach 0.7, flight path angle of 5 degrees, angle of attack at 8 degrees, as shown in Table 14. The dynamic pressure at separation is 270 psf and the wing area is sized for a nominal wing loading of 300 psf. Once the launch vehicle attains a typical vertical launch vehicle flight profile, the aerodynamic surfaces are jettisoned. The launch vehicle reaches a maximum dynamic pressure (q) of 715 psf at a moderate angle of attack, similar to most launch vehicles.

Table 14. Trajectory Analysis for the PDV-2 System Concept



ID	Event	Time (s)	Weight (lbm)	Altitude (ft)	Relative Velocity (ft/s)	Mach Number	Dynamic Pressure (psf)	Gamma (deg)	Alpha (deg)
1	Aircraft separation	0	305,000	25,000	711	0.7	270	5.0	8.0
2	Stage 1 ignition	10	305,000	25,243	673	0.7	239	-1.5	8.0
3	Maximum dynamic pressure	33	273,006	30,176	1,271	1.3	715	30.6	6.7
4	Aerosurface jettison	46	255,852	40,848	1,461	1.5	602	44.4	6.7
5	Stage 1 main engine cut off and separation	152	99,303	223,283	8,251	8.4	7.2	20.8	9.0
6	Stage 2 ignition	154	78,657	229,081	8,229	8.4	5.6	20.4	8.9
7	Fairing jettison	185	73,836	312,473	8,791	9.8	0.1	16.2	12.1
8	Stage 2 main engine cut off	513	22,969	606,934	24,189	27	0.0	0.0	2.5

The resulting weight statement reveals that the integrated aerodynamic surface module is approximately equal to the payload weight. However, these surfaces are jettisoned early in the trajectory and thus have only a few hundred pounds of impact on the payload. Payload delivery was computed to be 12,575 lbs for the closed vehicle, as shown at the bottom of Table 15.

Table 15. Weight Breakdown Statement for the PDV-2 System Concept

	Weight (lbm)
STAGE 1	
Structure	8,974
Propulsion	9,190
Thermal control	328
Power	164
Avionics	46
Stage 1 Dry Mass	18,701
Residuals, reserves, consumables	1,944
Main propellants	193,440
Start-up losses	580
Stage 1 Wet Mass	214,666
STAGE 2	
Structure	6,112
Propulsion	2,144
Thermal Control	283
Power	164
Avionics	484
Stage 2 Dry Mass	9,187
Residuals, reserves, consumables	813
Reaction control system propellants	443
Main propellants	53,394
Start-up losses	160
Stage 2 Wet Mass	63,997
Aerosurface module	
Wing	6,863
Fins	1,649
Actuators	1,035
Strongback structure	2,129
Aerosurface module total mass	11,677
Summary	
Fairing and adapter	2,086
Payload	12,575
Total gross mass	305,000

Using current integration and checkout practices for launch vehicles, the crew size was minimized to turn the vehicle around in 46 hours, as shown in Figure 13, with a margin of 20 percent and a crew of 23 technicians.

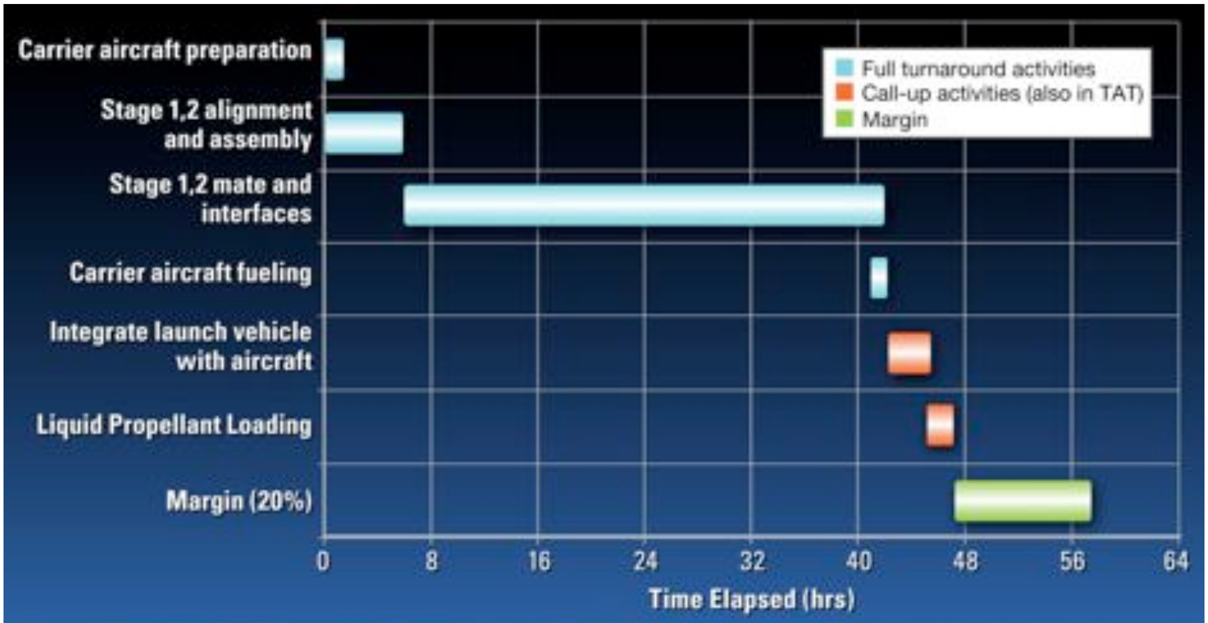


Figure 13. Results of the operational analysis for the PDV-2 system concept

Using failure rates of existing systems and the reliability exponential growth history of past systems, the LOM probabilities are 5.1 percent for the fourth flight and 1.2 percent for the 16th mission, as shown in Table 16. This LOM prediction does not include failures related to the Taurus rocket. As shown in the list of important elements, there are no dominating unreliable elements.

Table 16. Loss of Mission (LOM) Probabilities for the PDV-2 System Concept

End state values	4th flight		16th flight	
	Probability	**Mean flights between failure**	**Probability**	**Mean flights between failure**
Stage 1 LOM contribution	1.72%	58	0.41%	240
Stage 2 LOM contribution	2.37%	42	0.57%	172
Takeoff through rocket release LOM contribution	1.00%	101	0.24%	419
Total LOM	5.09%	20	1.22%	82
747-400F Loss of Vehicle	0.0058%	17,241	0.0014%	71,428
Most important elements			**Probability**	**Importance**
1	Off-nominal payload insertion		1.00%	0.266
2	Merlin 1-C off-nominal performance		0.40%	0.106
3	RL 10A-4-2 off-nominal performance		0.40%	0.106
4	Fairing separation		0.28%	0.074
5	Stage 1 separation		0.23%	0.062
6	Payload separation		0.23%	0.062
7	Exterior structure		0.16%	0.043
8	Human error		0.16%	0.041
9	Software		0.15%	0.040
10	Expendable aerosurface separation		0.09%	0.023

Program costs, shown in Table 17, were estimated assuming that a used 747-400F could be acquired and modified for $86 million. The Merlin 1C prices were assumed from the advertised cost of the SpaceX Falcon 1 and Falcon 9, and it was also assumed that SpaceX would sell their engines to the program. Development costs of the aerosurface module were based on traditional aerospace practices modeled with NAFCOM; however small commercial companies have shown factors of 3 to 8 lower than these traditional costs. In addition, the typical government oversight costs for the program are based on previous manned system development, and government facilities (and their associated costs) are used for testing and demonstration. Using a nominal 20 percent contingency, the total cost per flight and cost per pound of payload are estimated at $120 million and $9,555/lbs, respectively.

Table 17. Projected Costs for the PDV-2 System Concept, in FY10 Dollars, Assuming 6 Flights Per Year

DDT&E and Facilities Costs	
747-400F	$122 M
Stage 1	$272 M
Stage 2	$305 M
Aerosurfaces	$103 M
Facilities and ground service equipment	$134 M
Subtotal for DDT&E and facilities costs	$936 M
Acquisition and Production	
747-400F Acquisition and modifications	$86 M
Subtotal reusable first unit costs	$86 M
Stage 1 average production cost	$34 M
Stage 2 average production cost	$67 M
Aerosurfaces average production cost	$11 M
Subtotal expendable average production cost	$112M
Total recurring cost per pound of payload	$9,555 / lb.
Total recurring cost per flight	$120M / flight
Operations	
Turn-around time	68 hours
Call-up time	5.9 hours

Point Design Vehicle 2M

As an additional data point, the PDV-2 system concept was modified such that the second stage utilized LOX/RP propulsion, labeled PDV-2M. With the integrated specific impulse of an all LOX/RP vehicle, sizing the payload to a 305,000 lbs maximum gross weight will be less than the LOX/RP+LOX/LH2 PDV-2. However, if the carrier aircraft takes off with limited fuel or the launch vehicle without any RP, a much larger launch vehicle can be carried, but the carrier aircraft, the launch vehicle, or both, will need to be aerial fueled. This will require a separate tanker, complicating the concept of operations and adding an additional vehicle to the overall cost.

The PVD-2M grows from 77 ft to 112 ft in length and the payload to 16,742 lbs when aerial fueling is used. Using a SpaceX Merlin 1C engine, estimated to be 25 times cheaper than an RL10 based on thrust per cost, means the resulting cost per flight and cost per pound of payload is significantly reduced. In addition, the ground handling and launch vehicle storage of propellants is greatly reduced over any system with liquid hydrogen fuel.

Point Design Vehicle 3

The third point design vehicle, PDV-3, consists of the 747-400F and a two-stage all LOX/LH2 launch vehicle, is shown in Figure 14. With the large diameter of this PDV compared to the others, lateral directional stability and dynamic loads from buffet required more in depth analyses.

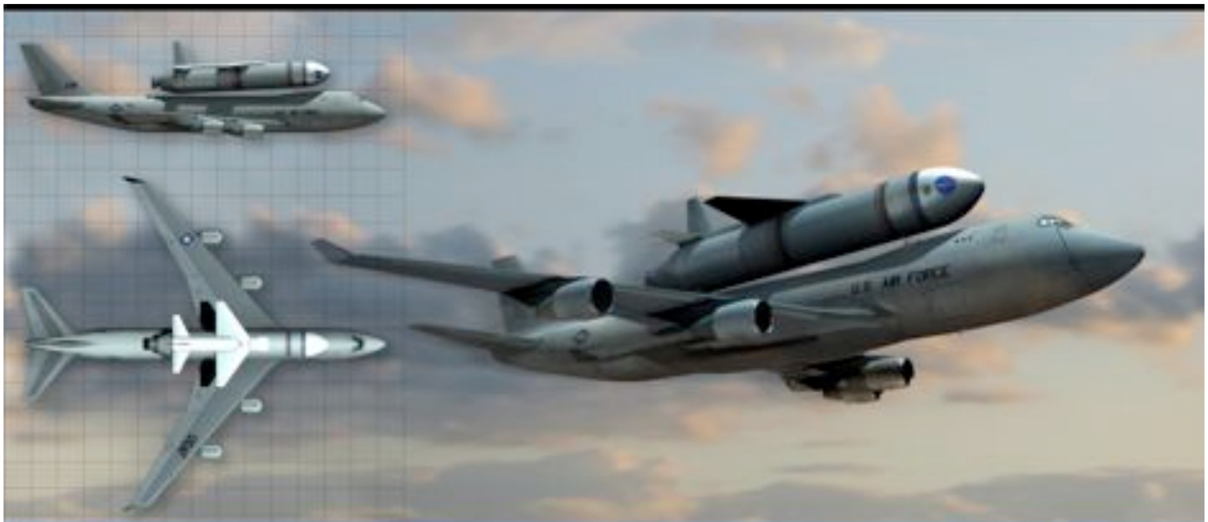


Figure 14. Design of the PDV-3 system concept

The first stage has one RS-25E (an air-start, expendable Space Shuttle Main Engine), and the second stage has three RL10A-4-2 LOX/H2 engines, shown in Figure 15. The wing and empennage are attached to the first stage with a strongback. All interstages, fairings and aerodynamic surface are graphite/epoxy composite materials, detailed in Figure 16.

Figure 15. Expanded configuration of the PDV-3 system concept

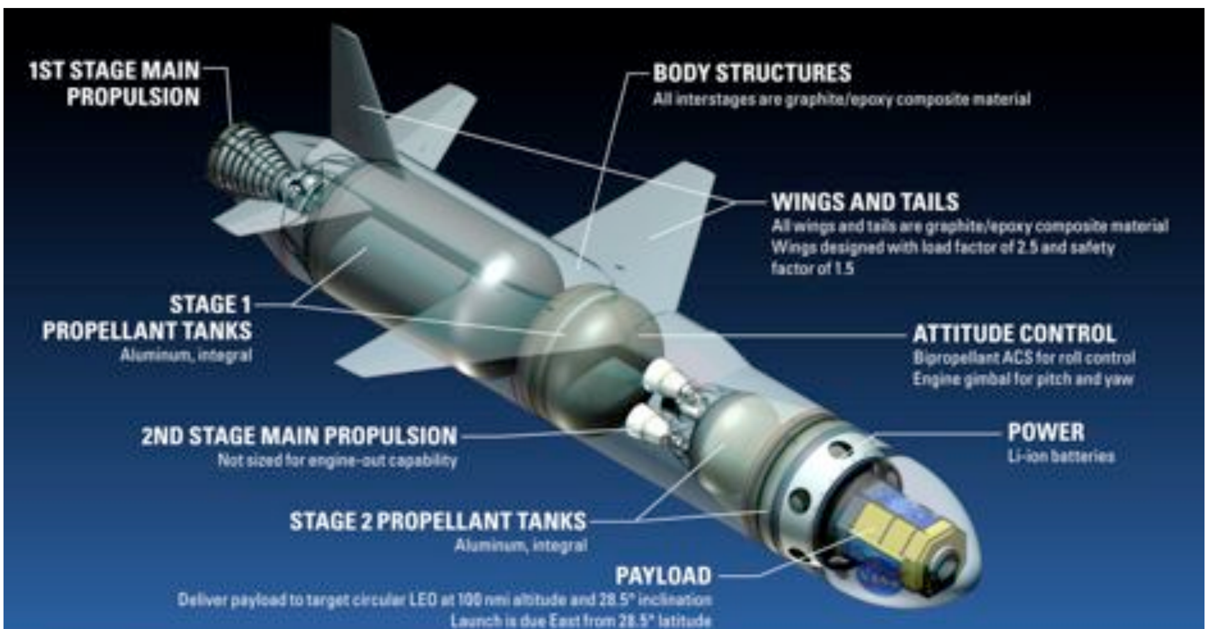


Figure 16. Materials and systems in the PDV-3 system concept

The summary dimensions and weights of the PDV-3 are shown in Table 18. With the selection of the liquid engines, the optimized payload weight limit of PDV-3 was within the gross weight of the launch vehicle equaling 308,000 lbs (minus 3,000 lbs of integration weight), which is the 747-400F maximum external payload limit. The PDV-3 dry weight also includes a 15 percent weight growth margin. This PDV has the highest payload of the four at 17,813 lbs.

Table 18. Summary Dimensions and Weights of the PDV-3 System Concept

Carrier aircraft	747-400F (modified)
External payload capacity	305,000 lbs.
Total length	231 ft.
Fuselage diameter	21 ft.
Wing span	211 ft.
Launch vehicle	2-stage liquid LOX/LH2
Total gross weight	305,000 lbs.†
Payload to LEO	17,812 lbs.
Total length	113.7 ft.
Maximum diameter	16.4 ft.
Wing span	53 ft.

† Includes dry mass margin

Like the other PDVs, only the isolated aerodynamics of the launch vehicle were computed in DATCOM using estimated dimensions shown in Table 19 and the schematic in Figure 17. This resulted in the aerodynamics analysis shown in Figure 18.

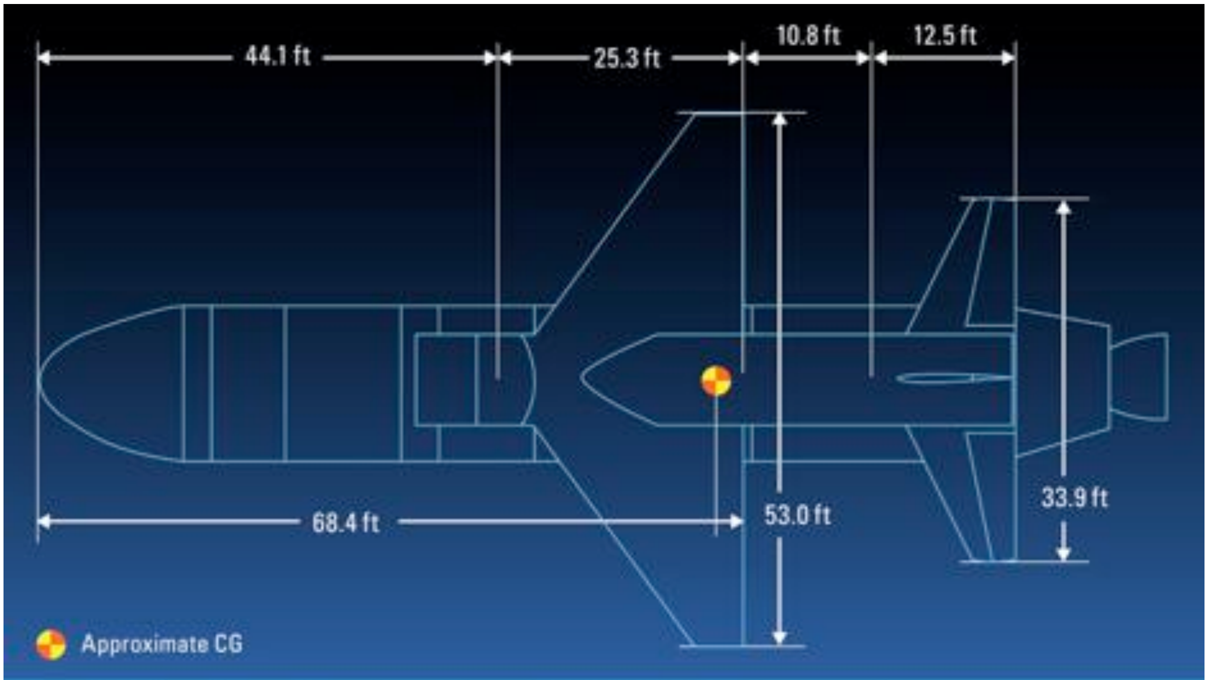


Figure 17. Schematic used in aerodynamic analysis for the PDV-3 system concept

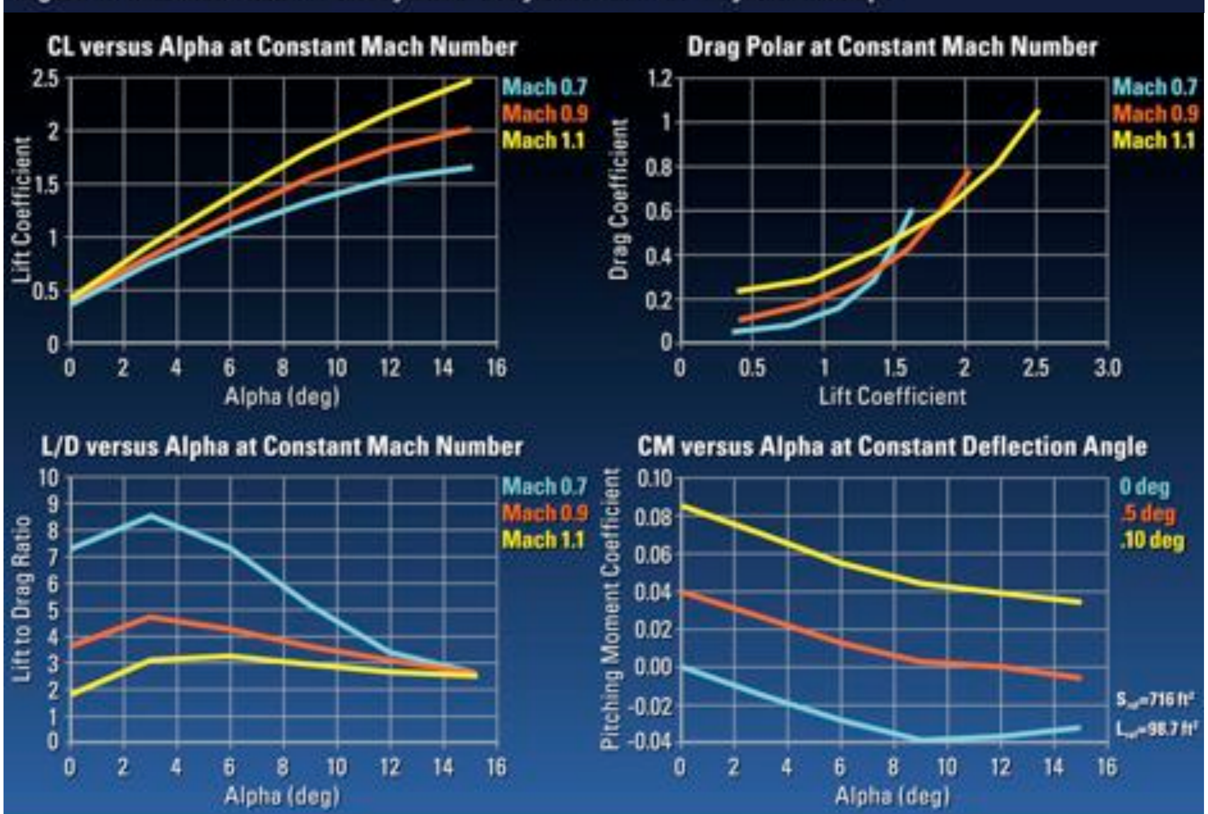


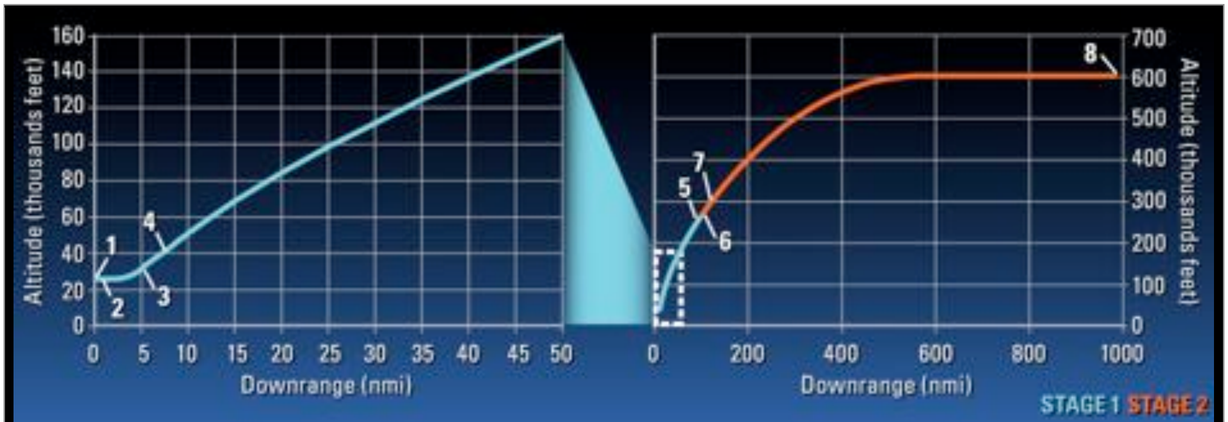
Figure 18. Aerodynamic analysis of the PDV-3 system concept

Table 19. Dimensions Used in Aerodynamic Analysis for the PDV-3 System Concept

Parameter	Wing	Horizontal/vertical tails (each)
Aspect ratio	3.5	4.0
Taper ratio	0.2	0.36
LE sweep angle	36.4	25.4
Platform area	803 ft^2	145 ft^2
Thickness-to-Chord ratio	0.1	0.1
Wing loading at 1g	380 lbs/ft^2	
Wing incidence angle	5°	

Using the thrust histories of the liquid rocket engines and the launch vehicle aerodynamics, the resulting POST trajectory is shown in Table 20. The launch separation state for the launch vehicle is at Mach 0.7, flight path angle of 5 degrees, angle of attack at 8 degrees. The dynamic pressure at separation is 270 psf and the wing area is sized for a nominal wing loading of 300 psf. Once the launch vehicle attains a typical vertical flight profile, the aerodynamic surfaces are jettisoned. The launch vehicle reaches a maximum dynamic pressure (q) of 794 psf at moderate angle of attack which is similar to most launch vehicles.

Table 20. Trajectory Analysis for the PDV-3 System Concept



ID	Event	Time (s)	Weight (lbm)	Altitude (ft)	Relative Velocity (ft/s)	Mach Number	Dynamic Pressure (psf)	Gamma (deg)	Alpha (deg)
1	Aircraft separation	0	305,000	25,000	711	0.7	270	5.0	8.0
2	Stage 1 ignition	10	305,000	25,238	670	0.7	237	-1.6	8.0
3	Maximum dynamic pressure	35	277,274	31,292	1,368	1.4	794	28.4	6.1
4	Aerosurface jettison	48	251,295	41,676	1,594	1.6	688	37.9	6.8
5	Stage 1 main engine cut off and separation	175	112,991	266,870	10,749	11.7	1.7	15.7	6.3
6	Stage 2 ignition	177	82,339	272,644	10,732	11.8	1.3	15.5	6.2
7	Fairing jettison	195	77,169	320,843	11,060	12.4	0.1	13.6	7.8
8	Stage 2 main engine cut off	516	29,437	607,161	24,118	27.0	0.0	0.0	2.5

The resulting weight breakdown statement is shown in Table 21. Although the integrated aerodynamic surface module is shown to be the same order of magnitude as the payload weight, these surfaces are jettisoned early in the trajectory and thus have only a few hundred pounds of impact on the payload. Payload delivery was computed to be 17,813 lbs for the closed vehicle.

Table 21. Weight Breakdown Statement for the PDV-3 System Concept

	Weight (lbm)
STAGE 1	
Structure	16,614
Propulsion	11,491
Thermal Control	516
Power	164
Avionics	46
Stage 1 dry mass	28,830
Residuals, reserves, consumables	1,821
Main Propellants	179,159
Start-up Losses	537
Stage 1 wet mass	210,347
STAGE 2	
Structure	7,341
Propulsion	2,166
Thermal control	183
Power	164
Avionics	484
Stage 2 dry mass	10,339
Residuals, reserves, consumables	767
Reaction control system propellants	567
Main propellants	50,308
Start-up losses	151
Stage 2 wet mass	62,132
Aerosurface module	
Wing	6,089
Fins	2,445
Actuators	1,535
Strongback structure	2,245
Aerosurface module total mass	12,313
Fairing and adapter	2,396
Payload	17,813
Total gross mass	305,000

Figure 19 shows the results of the operational analysis. Using current integration and checkout practices for launch vehicles, the crew size was minimized at 21 technicians to turn the vehicle around in 67 hours, with a margin of 20 percent.

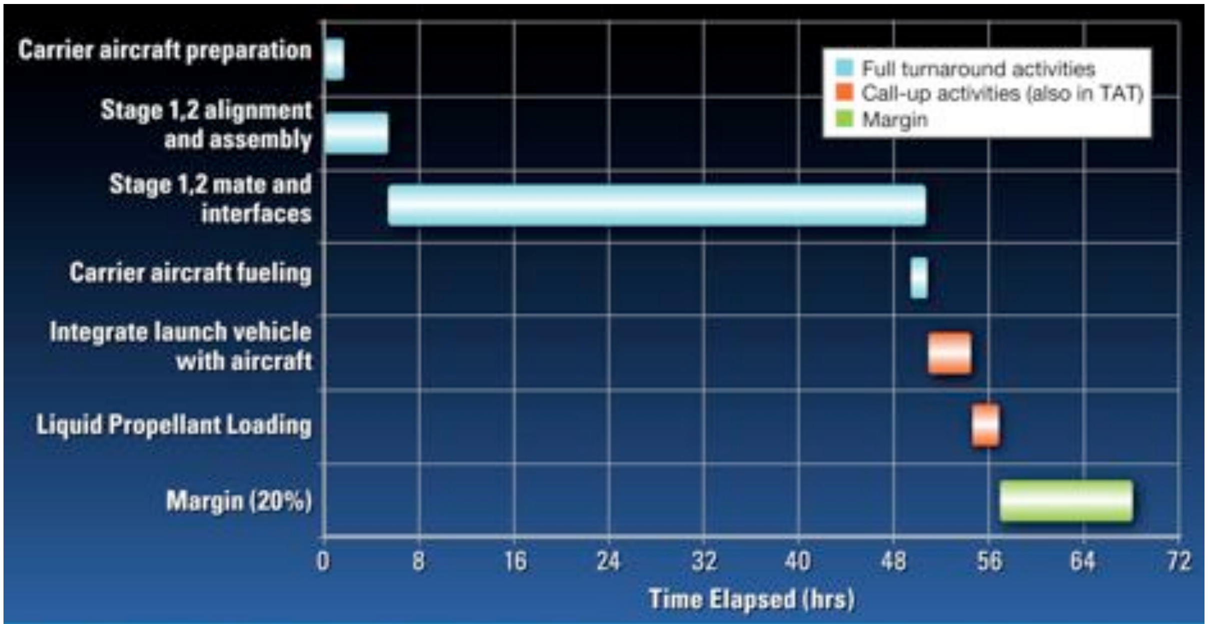


Figure 19. Results of the operational analysis for the PDV-3 system concept

Using failure rates of existing systems and the reliability exponential growth history of past systems, the LOM probabilities are shown in Table 22 at 5.3 percent for the fourth flight and 1.3 percent for the 16th mission. This LOM prediction did not incorporate two recent consecutive payload fairing separation mishaps of the Taurus rocket. As shown in the list of important elements, there are no dominating unreliable elements.

Table 22. Loss of Mission (LOM) Probabilities for the PDV-3 System Concept

End state values	4th flight		16th flight	
	Probability	Mean flights before failure	Probability	Mean flights before failure
Stage 1 LOM contribution	1.95%	50	0.47%	212
Stage 2 LOM contribution	2.36%	41	0.56%	178
Takeoff through rocket release LOM contribution	1.00%	101	0.24%	419
Total LOM	5.32%	19	1.27%	79
747-400F Loss of Vehicle	0.0058%	17,241	0.0014%	71,428

Most important elements		Probability	Importance
1	Off-nominal payload insertion	1.00%	0.250
2	RS-25E off-nominal performance	0.40%	0.100
3	RL 10A-4-2 off-nominal performance	0.40%	0.100
4	RS-25E contained failure	0.39%	0.097
5	Fairing separation	0.28%	0.070
6	Stage 1 separation	0.23%	0.058
7	Payload separation	0.23%	0.058
8	Exterior structure	0.16%	0.040
9	Human error	0.16%	0.039
10	Software	0.15%	0.038

Program costs, shown in Table 23, were estimated assuming that a used 747-400F could be acquired and modified for $86 million. The RS-25E costs were estimated by the NASA Constellation Program and a 15 percent integration cost was added. Note that the RS-25E air-start technology development cost (primarily for air-start capability) has not been added to DDT&E. Development costs of the aerosurface module were based on traditional aerospace practices modeled by NAFCOM; however small commercial companies have shown factors of 3 to 8 lower than these traditional costs. In addition, the typical government oversight for the program is based on previous manned system development, and government facilities (and their associated costs) are used for testing and demonstration. Using a nominal 20 percent contingency, the total cost per flight and cost per pound of payload are $130 million and $7,299/lbs, respectively.

Table 23. Projected Costs for the PDV-3 System Concept, in FY10 Dollars, at 6 Flights Per Year

DDT&E and facilities	
747-400F	$122 M
Stage 1	$1.78 B
Stage 2	$295 M
Aerosurfaces	$106 M
Facilities and GSE	$132 M
Subtotal for DDT&E and facilities costs	$2.44 B
Acquisition and Production	
747-400F acquisition and modifications	$86 M
Subtotal reusable first unit costs	$86 M
Stage 1 average production cost	$51 M
Stage 2 average production cost	$58 M
Aerosurfaces average production cost	$12 M
Total expendable average production cost	$120 M
Subtotal expendable average production cost	
Total recurring cost per pound of payload	$7,299 / lb
Total recurring cost	$130M / flight
Operations	
Turn-around time	57 hours
Call-up time	7.2 hours

Discussion of Results for Point Design Vehicles

The results of the analysis in Table 24 showed that even with the constraint of using existing engines, the resulting systems still produced good payload performance, cost, and reliability. The three-stage solid, PDV-1, had the lowest DDT&E cost. The all LOX/LH2 two-stage PDV-3 had the highest payload delivery. The all LOX/RP two-stage, PDV-2M, takes off without RP in the launch vehicle and with reduced fuel in the carrier aircraft and is fueled in flight. This concept enables the LOX/RP vehicle to approach the payload of the all LOX/LH2 vehicle. The concept of operations is more complicated with aerial fueling; however the ground and flight operations are simplified by not handling hydrogen. In addition, a Merlin 1C LOX/RP engine reduces the cost per pound of PDV-2M to the lowest value for all concepts.

Table 24. Comparison of PDV System Concepts

Metric	PDV-1	PDV-2	PDV-2M	PDV-3
Carrier aircraft	747-400F	747-400F	747-400F	747-400F
Launch vehicle	3-stage solid	2-stage LOX/RP+ LOX/LH2	2-stage LOX/RP+ aerial refuel	2-stage LOX/LH2
Payload to LEO	5,662 lbs.	12,575 lbs.	16,742 lbs.	17,812 lbs.
Total launch vehicle gross weight	288, 483 lbs.	305,000 lbs.	305,000 lbs.	305,000 lbs.
Total length	100 ft.	102 ft.	112 ft.	114 ft.
Maximum diameter	7.75 ft.	12.5 ft.	12.5 ft.	16.4 ft.
DDT&E and facilities cost	$475 M	$935 M	$950 M	$2,440 M
Recurring cost/payload	$8,934/ lb.	$9,555/lb.	$5,027/lb.	$7,299/lb.
Recurring cost/flight	$52M/flight	$120M/flight	$101M/flight	$130M/flight
4th flight LOM probability	6.52%	5.09%		5.32%

IV. Flight Test Demonstrator Analysis

The goal of the Flight Test Demonstrator (FTD) is to reduce development risk of a mission-capable horizontal take-off space launch system, by executing a realistic and achievable program that is directly traceable to ultimate performance, operations, and cost goals. While several existing systems have demonstrated various aspects of air-launch technologies, demonstrations of separation dynamics with large, fully-loaded launch vehicles, ground operations, efficient system integration, and safe handling of propellant are required to validate the technologies and procedures needed for the DDT&E of a routine, safe, and cost effective horizontal launch system.

The major technical challenges to be demonstrated include the following:

- Efficient and low-cost design, development, mission, and ground and flight operations of a horizontal take-off space launch system
- In-flight command and control of the launch vehicle
- Loads and structural interfaces between the carrier aircraft and launch vehicle at takeoff, climb, cruise, and launch
- Launch altitude, velocity, and flight path angle
- Separation physical mechanism and aerodynamics
- Launch vehicle transition from initial separated state to the optimum ascent trajectory
- Validation of cost and operations models
- Cryogenic handling and storage

A critical aspect of the flight technology demonstration is operations. Current launch costs range from $30,000 per pound for the Pegasus, to $4,000 to $5,000 per pound for expendable evolved launch vehicles, and to $2,500 per pound for a Falcon 9 (assuming the full payload capability is used for each). The factors that drive this large range include hardware acquisition, system integration, test and evaluation, mission planning, and operations—but perhaps the most important factor is annual launch rate.

The realities of the factor are shown in Table 25 for the Space Shuttle, where the launch rate ranged from a high of 7 to a low of zero launches per year. Given annual funding of approximately $3 billion per year calculates an average cost of $13,000 per pound of payload to orbit. The recurring launch costs for the Space Shuttle reveal an opportunity: only 9 percent of the cost is accounted in hardware acquisition, integration, and system turnaround, and only 22 percent is in indirect system support. The majority, almost 70 percent, is attributed to management support. Thus, a key driver for any planned flight test is to demonstrate a change to the traditional processes that contributed to the staggering overhead burden. These will include changes to not only management oversight methods, but to quality control, logistics support, traffic and flight control approaches, and launch and support infrastructure.

Table 25. Space Shuttle Cost Analysis

Operations function	Total cost (M$) (1994 $)	Percent of total
Hardware acquisition, integration, turnaround		9%
Element receipt and acceptance	1.4	
Landing and recovery	19.6	
Vehicle assembly and integration	27.1	
Launch	51.5	
Offline payload and crew	75.9	
Turnaround	112.3	
Indirect system support		22%
Vehicle depot maintenance	237.5	
Traffic and flight control	199.4	
Operations support infrastructure	318.6	
Management support		69%
Concept-unique logistics	842.7	
Operations planning and management	1,477.4	
Total	3,363.4	100%

Source: Study on Access to Space, 1994.

Two flight test demonstration system concepts were developed to evaluate potential technology and performance approaches to meeting program goals. Program costs for both were estimated assuming that the 747-100 Shuttle Carrier Aircraft (SCA) would be available at the end of the Shuttle program. The same analysis methods used for the point design vehicles to estimate system performance and life cycle costs are used here; however, the purpose of this analysis is only to provide first order feasibility to effectively evaluate test flight options, and not to offer a final solution.

Flight Test Demonstrator 1

The first concept flight test demonstrator (FTD-1), shown in Figure 20, consists of the 747-100 SCA and a four-stage solid launch vehicle. This approach was designed to minimize DDT&E costs.

Figure 20. Configuration of the FTD-1 system concept

The four-stage launch vehicle configuration, seen in Figure 21, is intentionally similar to a Taurus rocket and consists of a Castor 120 first stage, an Orion 50S XLG second stage, an Orion 50XL third stage, and an Orion 38 fourth stage. The wing and empennage are attached to the first stage with a strongback. All interstages, fairings and aerodynamic surfaces are composite materials, as shown in Figure 22. Power and attitude control subsystems are based on existing systems.

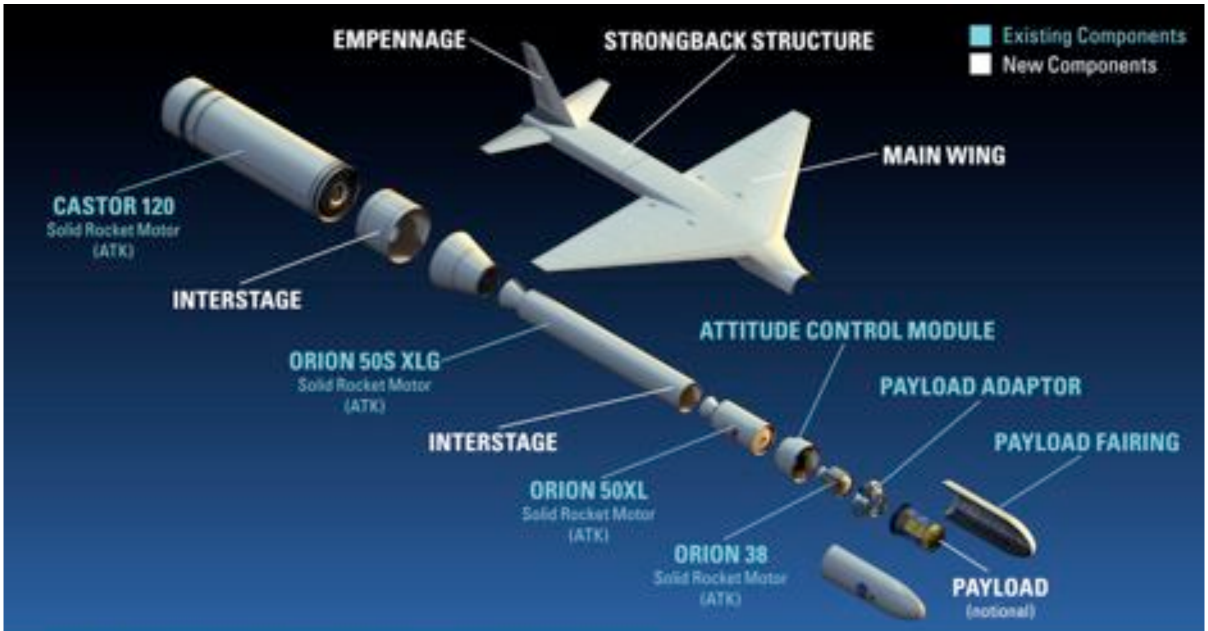


Figure 21. Configuration details of the FTD-1 system concept

Figure 22. Materials and systems for the FTD-1 system concept

Because the gross weight of the launch vehicle is more than 40,000 lbs lighter than the Space Shuttle, structural modifications to the fuselage of the SCA may not be required. However, because the vehicle is longer than the space shuttle, the attachment points may have to be moved. Further analysis will be needed to determine whether an active separation mechanism will be needed.

The summary dimensions and weights of the FTD-1 are shown in Table 26. As with the point design vehicles, only the isolated aerodynamics of the launch vehicle was computed in DATCOM using the estimated schematic in Figure 23 and dimensions in Table 27. The predicted aerodynamics are shown in Figure 24.

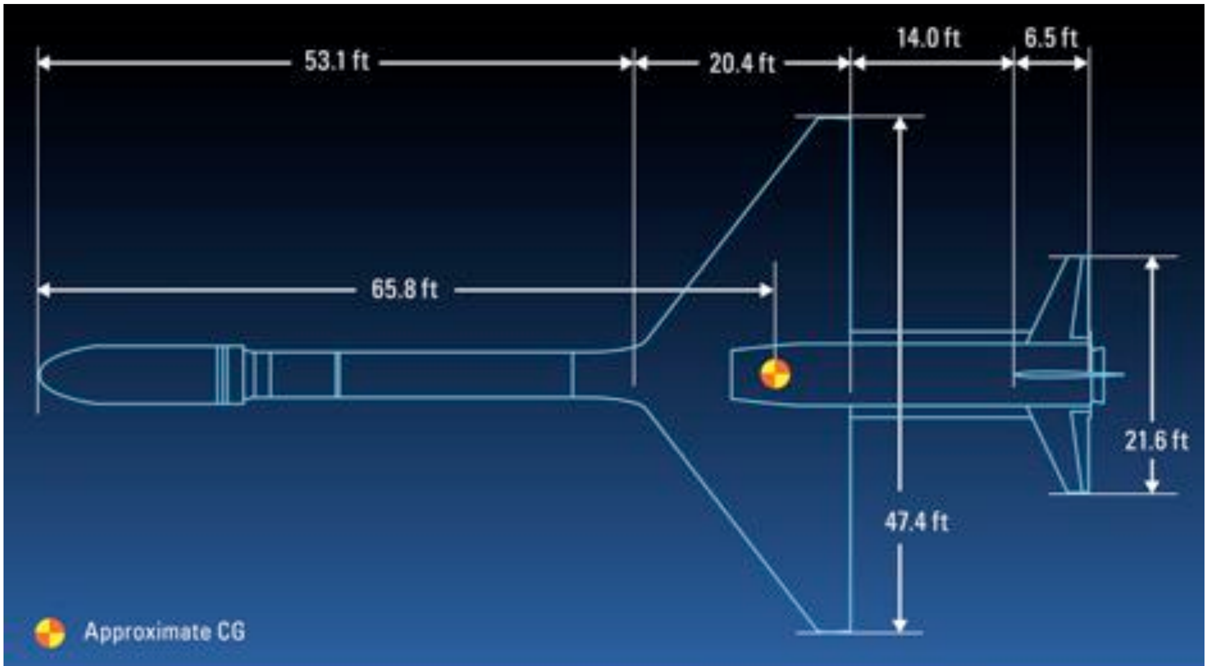


Figure 23. Schematic used for aerodynamics calculations for the FTD-1 system concept

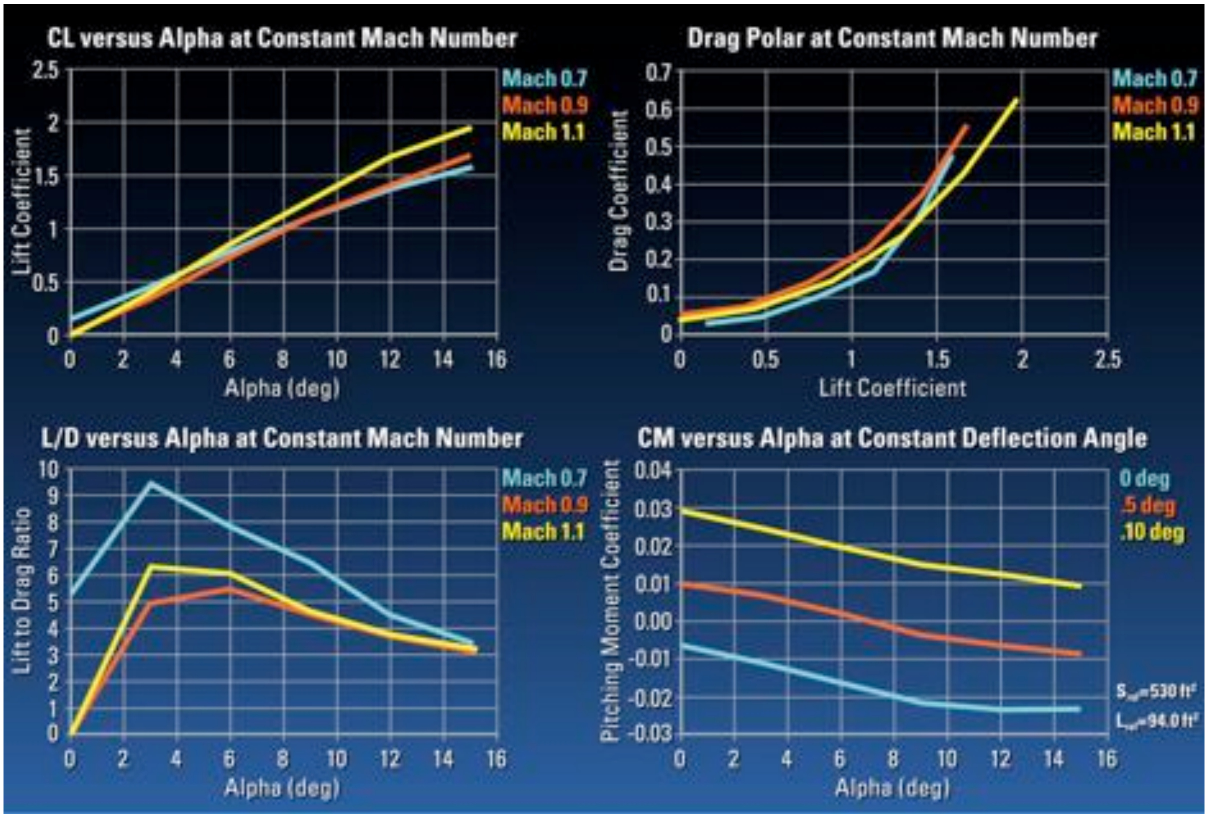


Figure 24. Aerodynamic analysis results for the FTD-1 System Concept

Using the thrust histories of the solid rocket motors and the launch vehicle aerodynamics, the resulting POST trajectory is shown in Table 28. The separation state for the launch vehicle is at Mach 0.7, flight path angle of 5 degrees, angle of attack at 8 degrees. The dynamic pressure at separation is 270 psf and the wing area is sized for a nominal wing loading of 300 psf. Once the launch vehicle attains a typical vertical flight profile, the aerodynamic surfaces are jettisoned. The launch vehicle reaches a maximum dynamic pressure (q) of 2,248 psf which is very aggressive compared with a nominal launch vehicle maximum q of approximately 800 psf; however, the X-43 was designed to an upper limit of 2,000 psf. No trade studies of payload performance versus constrained maximum q were done in this analysis which may result in the loss of hundreds of pounds of payload. The non-constrained trajectory payload delivery was computed to be 4,562 lbs.

Table 26. Performance Summary for the FTD-1 System Concept

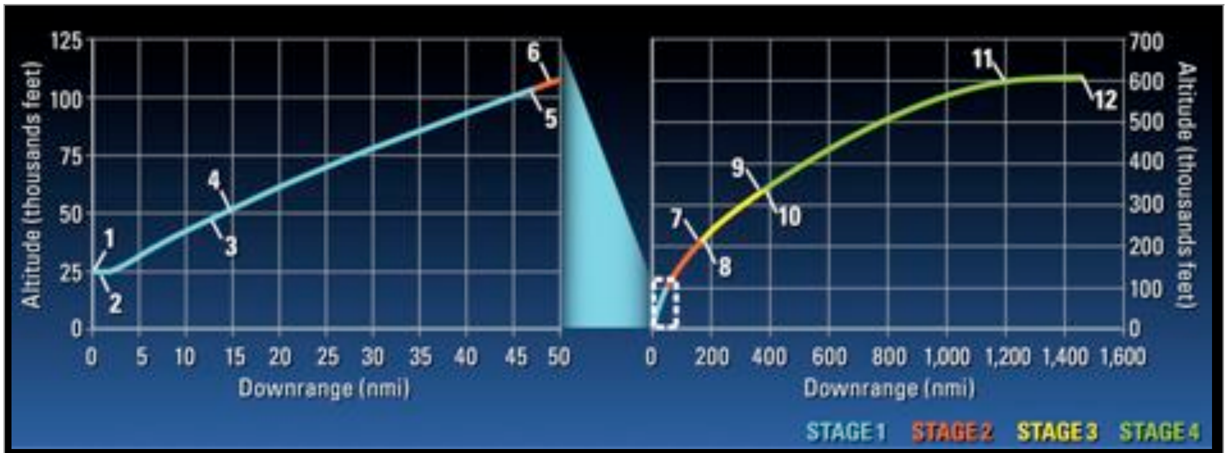
Carrier aircraft	747-100 SCA
External carriage capacity	192,000 lbs.
Total length	231 ft.
Fuselage diameter	21 ft.
Wing span	196 ft.
Launch vehicle	
Total gross weight	179,474 lbs.†
Payload to LEO	4,562 lbs.
Total length	99 ft.
Maximum fuselage diameter	7.75 ft.
Wing span	47.4 ft.

† Includes dry mass margin

Table 27. Dimensions Used for Aerodynamics Design for the FTD-1 System Concept

	Wing	Horizontal tails (each)	Vertical tail
Aspect ratio	3.76	4.58	3.69
Taper ratio	0.15	0.30	0.35
LE sweep angle	37.5°	25.0°	40.0°
Platform area	597.8 ft^2	51.2 ft^2	51.2 ft^2
Thickness-to-Chord ratio	0.1	0.1	0.1
Wing incidence angle	0°	-	-

Table 28. Trajectory Analysis Data Summary for the FTD-1 System Concept



ID	Event	Time (s)	Weight (lbm)	Altitude (ft)	Relative Velocity (ft/s)	Mach Number	Dynamic Pressure (psf)	Gamma (deg)	Alpha (deg)
1	Aircraft separation	0	179469	25,000	711	0.7	270	5.0	8.0
2	Stage 1 ignition	10	179468	25,322	657	0.6	228	-0.4	8.0
3	Aerosurface jettison	50	121005	47,690	3,314	3.4	2,232	19.0	2.2
4	Maximum dynamic pressure	53	109764	50,736	3,577	3.7	2,248	18.3	2.2
5	Stage 1 burnout and separation	90	64215	102,540	7,715	7.8	877	12.6	4.1
6	Stage 2 ignition	92	53666	105,871	7,687	7.7	745	12.3	4.3
7	Stage 2 burnout and separation	159	20561	215,168	16,106	16.1	38.0	6.7	1.8
8	Stage 3 ignition	161	17691	218,903	16,098	16.2	32.6	6.6	1.6
9	Stage 3 burnout and separation	229	9083	335,239	22,128	24.7	0.2	4.5	0.4
10	Fairing jettison	236	7950	347,474	22,111	24.7	0.1	4.4	0.2
11	Stage 4 ignition	462	7150	593,720	21,768	24.3	0.0	1.2	-7.5
12	Stage 4 burnout and separation	528	5460	607,444	24,187	27.0	0.0	0.0	-6.5

Table 29. Weight Breakdown Statement for the FTD-1 System Concept

Item	Weight (lbm)
STAGE 1: CASTOR 120	
Motor inert mass	8,976
Subsystems	216
Interstage	1,357
Propellant	108,038
Stage 1 loaded mass	118,587
STAGE 2: Orion 50S XLG	
Motor inert mass	2,599
Subsystems	72
Interstage	199
Propellant	33,105
Stage 2 loaded mass	35,975
STAGE 3: Orion 50XL	
Motor inert mass	870
Subsystems	67
Interstage	154
Propellant	8,650
Stage 3 loaded mass	9,741
STAGE 4: Orion 38	
Motor inert mass	267
Subsystems	13
Propellant	1,699
Stage 4 loaded mass	1,979
Aerosurface module	
Wing	4,775
Fins and actuators	1,126
Strongback structure	1,316
Aerosurface module total mass	7,218
Attitude control module and power	613
Fairing and payload adapter	800
Payload	4,562
Total vehicle gross mass	179,474

The resulting weight statement in Table 29 shows similar weights to the Taurus rocket except for the aerodynamic surfaces. As shown in the trajectory above, these surfaces are jettisoned early in the trajectory and thus have a relatively small impact on the payload.

Using failure rates of existing systems, the success probabilities are shown in Figure 25 and Table 30. As shown at the end of the trajectory (and also in the failure ranking), fairing separation has been determined to be a higher probability of failure event because of two recent consecutive Taurus rocket fairing failures. As shown in the rankings of failures, the fairing separation is an order of magnitude higher than all other propulsion and human error events. The predicted reliability improves with each flight based on historical reliability growth curves for past systems. Because these issues are expected to be resolved for the Taurus rocket for future missions, the reliability predictions presented here may be considered very conservative.

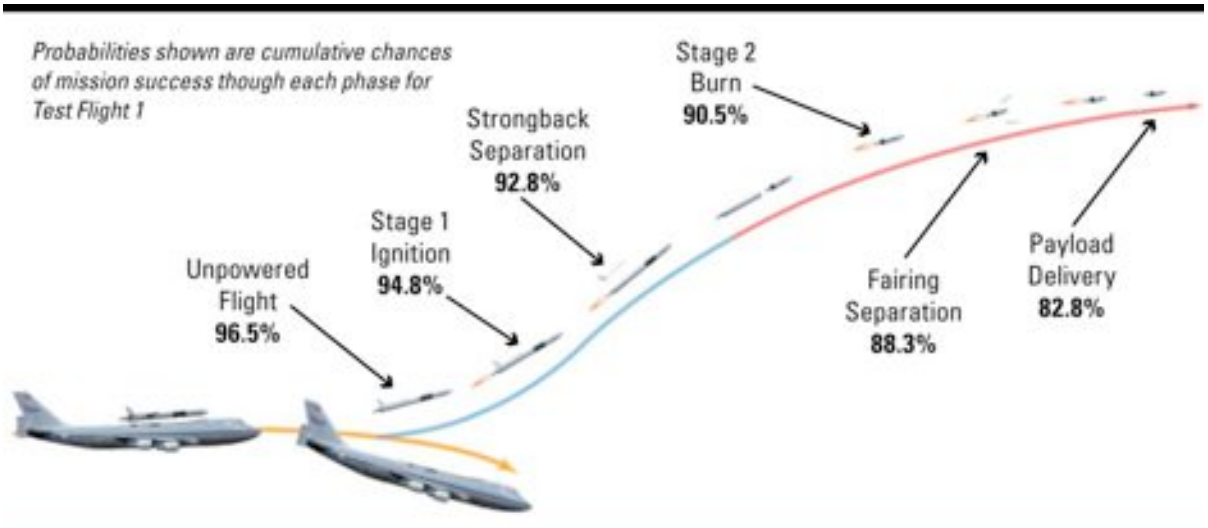


Figure 25. Cumulative chances of mission success through each phase for the FTD-1 system concept

Development costs, shown in Table 31, were based on traditional aerospace practices modeled in NAFCOM. Traditional government oversight and existing government facilities were modeled for testing and demonstration. A nominal 20 percent contingency is used bringing the total estimated cost of four flights to $320 million (FY2010 dollars).

Table 30. Reliability Assessment for the FTD-1 System Concept

	Probability
Demonstrated historical reliability	67%
Predicted reliability—Test 1	77.72%
Predicted reliability—Test 2	79.81%
Most important elements	Importance
Fairing separation	0.803
Off-nominal payload insertion	0.047
Off-nominal propulsive performance (stages 2, 3, 4)	0.019
Off-nominal propulsive performance (stage 1)	0.017
Stage 1 separation	0.011
Stage 2 separation	0.011
Stage 3 separation	0.011
Payload separation	0.011

Table 31. Program Cost Summary for the FTD-1 System Concept

Development phase costs	$91 M
Test program phase costs	$109 M
Total government team and program management	$67 M
Total contingency (20%)	$53 M
Total test program cost	$320 M

Flight Test Demonstrator 2

The second concept flight test demonstrator (FTD-2), shown in Figure 26, is comprised of the 747-100 Shuttle Carrier Aircraft (SCA) and a two-stage launch vehicle with LOX/RP propulsion on both stages similar to the Falcon 1e. The Falcon 1e was selected as a convenient example of a low-cost, low-risk demonstrator.

The two-stage liquid engine launch vehicle configuration is shown in Figure 27. The first stage is equipped with a LOX/RP Merlin 1C engine and the second stage with a Kestrel engine, both developed by SpaceX.

Figure 26. Configuration of the FTD-2 system concept

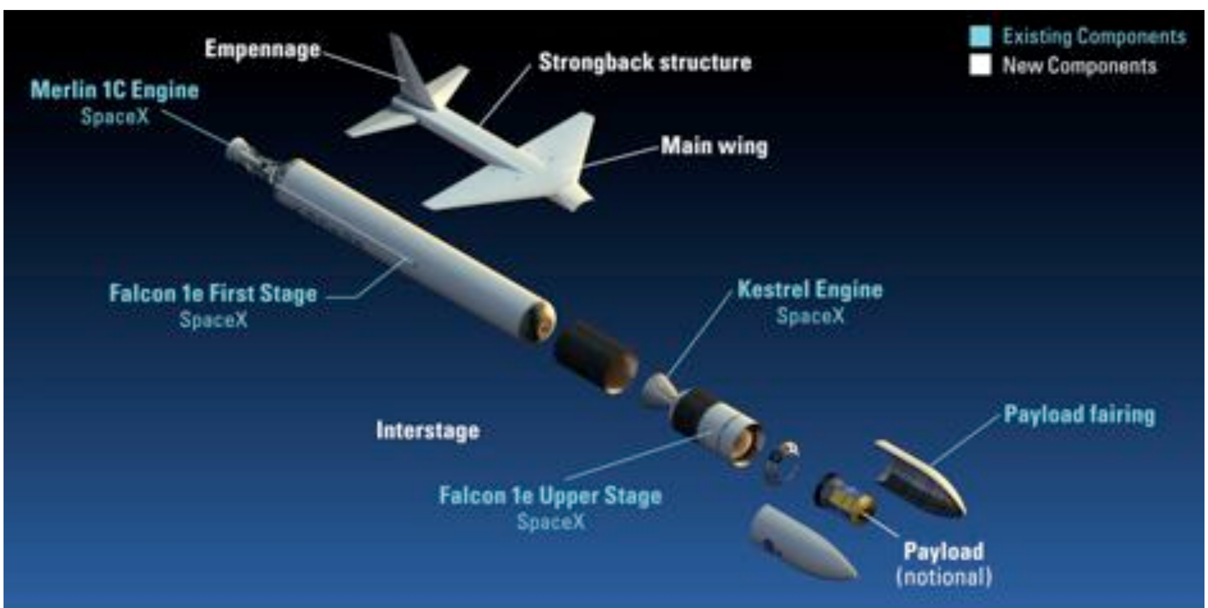


Figure 27. Expanded configuration of the FTD-2 system concept

Because the launch vehicle is less than half the weight of the Space Shuttle, fuselage structural modifications may not be required. However, because the vehicle is substantially shorter than the shuttle, the attachment points will have to be moved and an active separation mechanism may have to be added.

The wing and empennage are attached to the first stage with a strongback. All interstages, fairings and aerodynamic surface are composite materials, as shown in Figure 28. Power and attitude control subsystems are based on existing subsystems. The performance summary of the FTD-2 is shown in Table 32.

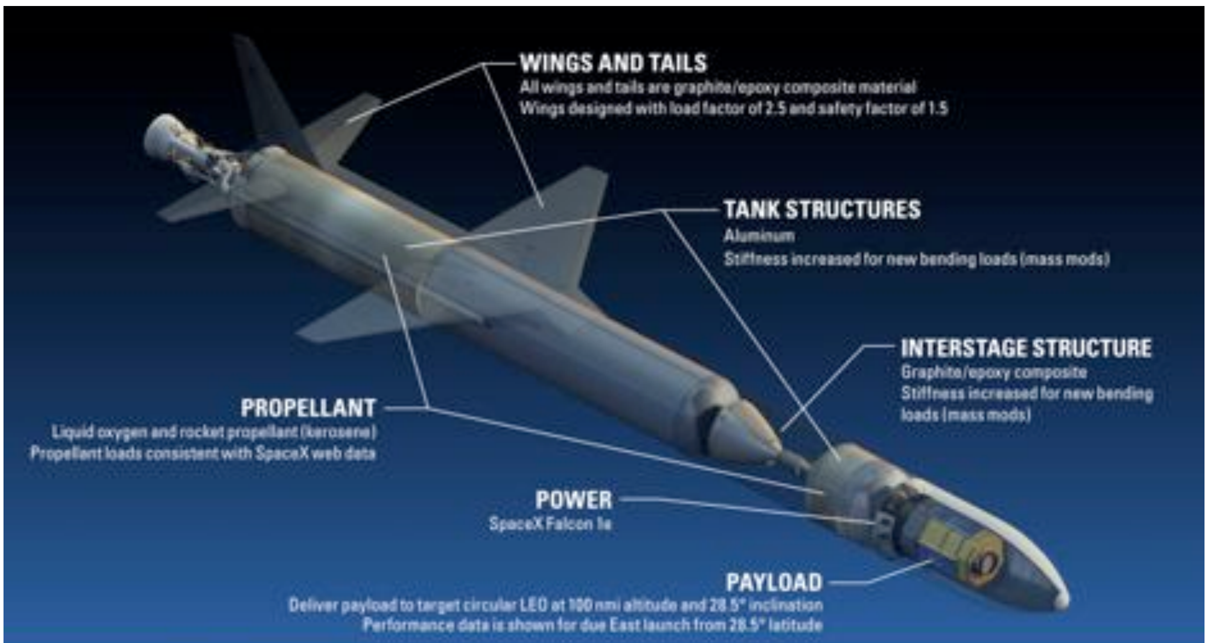


Figure 28. Materials and structures in the FTD-2 system concept

Like the point design vehicles, only the isolated aerodynamics of the launch vehicle were computed in DATCOM using the schematic in Figure 29 and dimensions in Table 33 to produce the predicted aerodynamics characteristics in Figure 30.

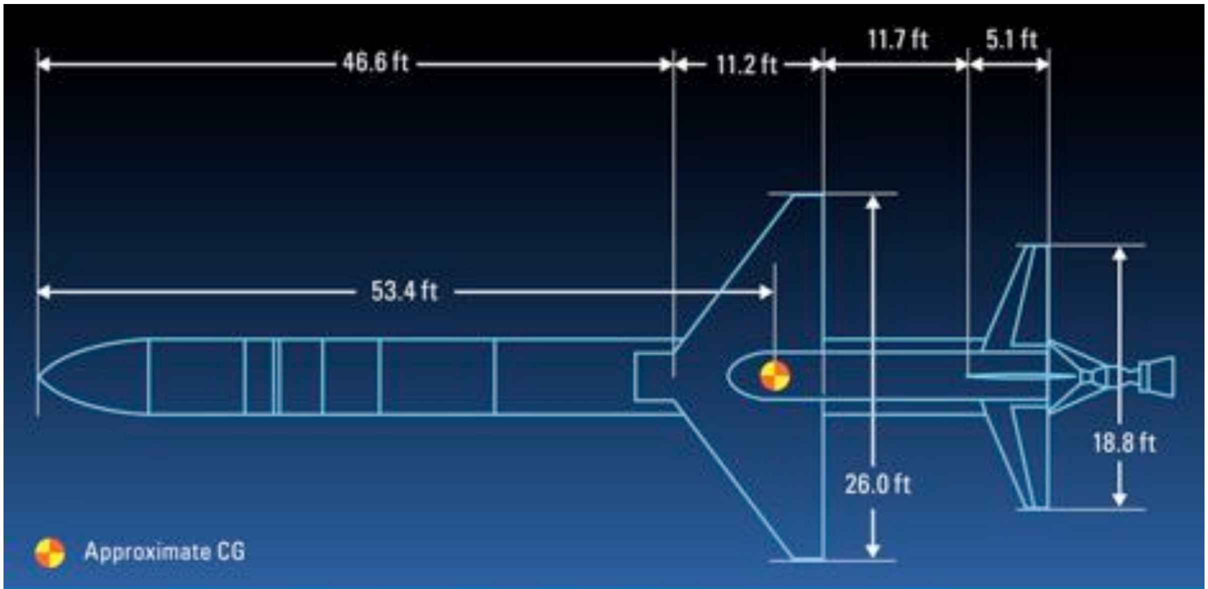


Figure 29. Schematic used for aerodynamics analysis for the FTD-2 system concept

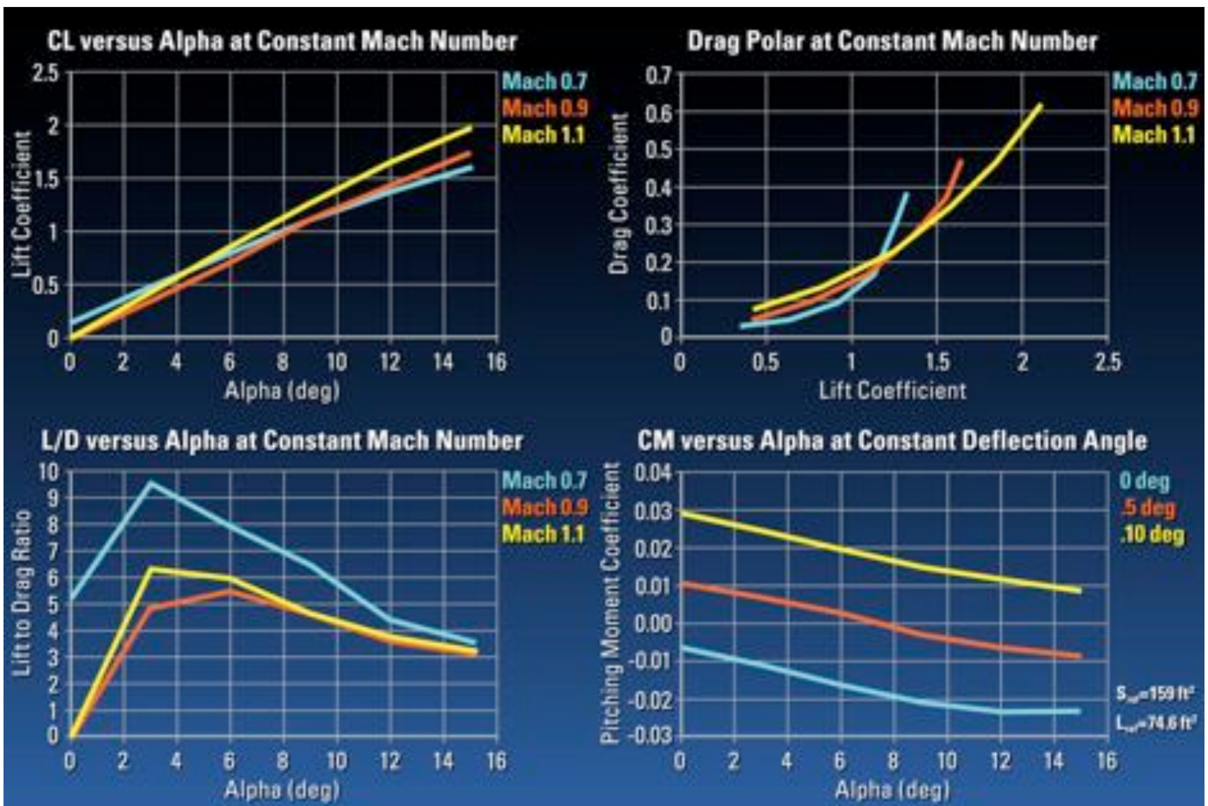


Figure 30. Results of aerodynamics analysis for the FTD-2 system concept

Using the thrust histories of the launch vehicle aerodynamics, the resulting POST trajectory is shown in Table 34. The launch separation state for the launch vehicle is at Mach 0.7, flight path angle of 5 degrees, angle of attack at 8 degrees. The dynamic pressure at separation is 270 psf and the wing area is sized for a nominal wing loading of 300 psf. Preliminary separation analysis indicates that this separation scenario is adequate, but further detailed analysis must be conducted for verification. Once the launch vehicle attains a typical vertical flight profile, the aerodynamic surfaces are jettisoned. The launch vehicle reaches a maximum dynamic pressure (q) of 980 psf.

Table 32. Performance Summary for the FTD-2 System Concept

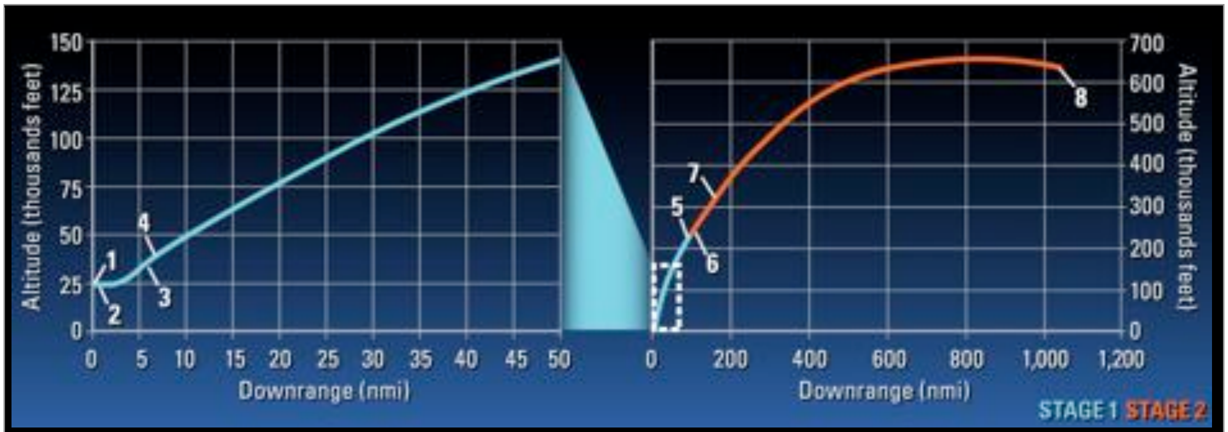
Carrier aircraft	747-100 SCA
External carriage capacity	192,000 lbs.
Total length	231 ft.
Fuselage diameter	21 ft.
Wing span	196 ft.
Launch vehicle	
Total gross weight	81,987 lbs.†
Payload to LEO	2,749 lbs.
Total length	81 ft.
Maximum fuselage diameter	5.5 ft.
Wing span	26 ft.

† Includes dry mass margin

Table 33. Dimensions Used for Aerodynamics Design for the FTD-2 System Concept

	Wing	Horizontal Tails (each)	Vertical Tail
Aspect ratio	3.51	5.00	3.77
Taper ratio	0.19	0.30	0.30
LE sweep angle	37.3°	23.1°	40.0°
Planform area	193.2 ft.^{2}	35.5 ft.^{2}	35.2 ft.^{2}
Thickness-to-Chord ratio	0.1	0.1	0.1
Wing incidence angle	5°	-	-

Table 34. Trajectory Analysis Data Summary for the FTD-2 System Concept



ID	Event	Time (s)	Weight (lbm)	Altitude (ft)	Velocity (fps)	Mach Number	Dynamic Pressure (psf)	Gamma (deg)	Alpha (deg)
1	Aircraft separation	0	81,914	25,000	711	0.7	270	5.0	8.0
2	Stage 1 ignition	10	81,914	25,250	665	0.7	233	-1.5	8.0
3	Aerosurface jettison	40	68,249	37,264	1,703	1.8	970	35.1	5.1
4	Maximum dynamic pressure	44	62,673	41,662	1,902	2.0	980	32.7	5.1
5	Stage 1 main engine cut off and separation	144	17,141	228,996	13,125	13.5	14.2	15.5	5.2
6	Stage 2 ignition	146	13,259	235,960	13,108	13.6	10.5	15.3	5.2
7	Fairing jettison	175	12,676	329,295	13,333	14.9	0.1	12.9	7.2
8	Stage 2 main engine cut off and separation	612	3,980	606,306	24,189	27.0	0.0	0.0	10.8

Table 35. Weight Breakdown Statement for the FTD-2 System Concept

Item	Weight (lbm)
STAGE 1	
Structure	2,032
Propulsion	1,520
Thermal control	34
Power	42
Avionics	20
Stage 1 dry mass	3,648
Residuals, reserves, consumables	234
Main propellants	61,011
Start-up losses	183
Stage 1 wet mass	65,076
STAGE 2	
Structure (including payload adapter)	573
Propulsion	240
Thermal control	16
Power	187
Avionics	139
Stage 2 dry mass	1,155
Residuals, reserves, consumables	138
Reaction control system propellants	13
Main propellants	8,946
Start-up losses	27
Stage 2 wet mass	10,279
Aerosurface module	
Wing	1,704
Fins and actuators	1,222
Strongback structure	652
Aerosurface module total mass	3,578
Summary	
Fairing	305
Payload	2,749
Total gross mass	81,987

The resulting weight breakdown statement in Table 35 shows similar weights to the Falcon 1e except for the aerodynamic surfaces. As shown in the trajectory analysis, these surfaces are jettisoned early in the trajectory and thus have only a few hundred pound impact on the payload. Payload was computed to be 2,749 lbs.

Using failure rates of existing systems, the success probabilities are shown in Table 36 and Figure 31. Because the Falcon 1e has had similar flight test performance as previous liquid rocket systems, the ending probability of success is somewhat higher than the four-stage solid case. The payload fairing failure percentage is much lower in this case and it is expected that the four-stage solid configuration will have a similar reliability once the payload fairing failure mechanism is found and corrected.

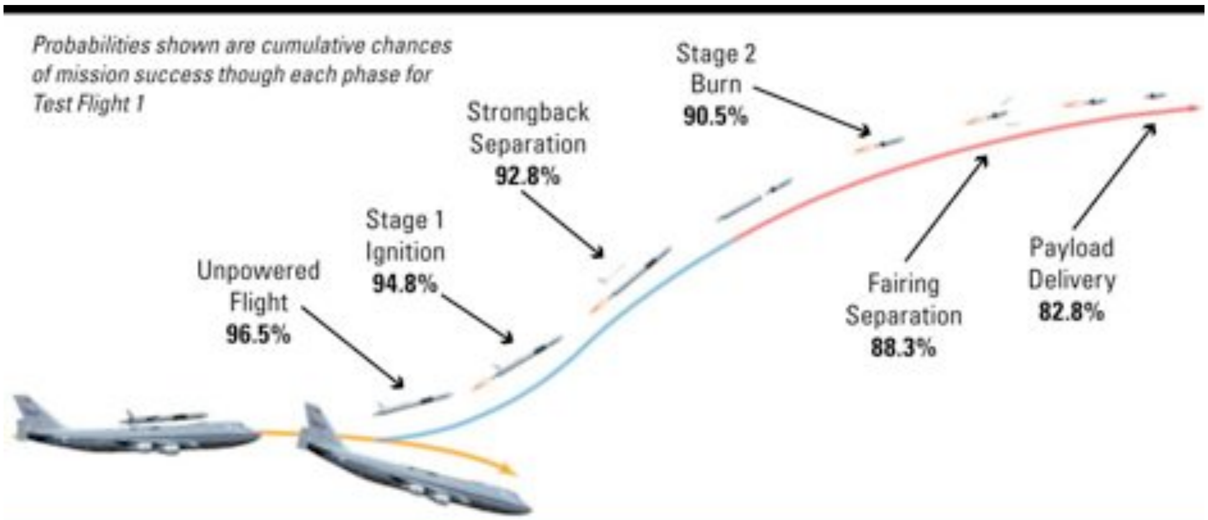


Figure 31. Cumulative chances of mission success through each phase for the FTD-2 system concept

Table 36. Reliability Assessment for the FTD-2 System Concept

	Probability
Demonstrated historical reliability	60%
Predicted reliability—Test 1	82.82%
Predicted reliability—Test 2	85.35%
Most important elements	Importance
Off-nominal payload insertion	0.274
Fairing separation	0.077
Stage 1 separation	0.064
Payload separation	0.064
Vehicle structure	0.059
Human error	0.057
Software	0.055
Kestrel performance	0.053
Merlin 1C performance	0.030
Strongback separation	0.024

Program costs for FTD-1, shown in Table 37, were estimated assuming that the 747 Shuttle Carrier Aircraft would be available at the current funding levels. Development costs were based on commercial space engineering and production practices which have shown factors of 3 to 8 lower than traditional aerospace industry costs. In addition, the traditional government oversight and insight into the program is assumed and government facilities (and their associated costs) are used for testing and demonstration. A nominal 20 percent contingency is used bringing the total estimated cost of four flight, four year demonstrator program to $245 million (FY2010 dollars), less than the FTD-1 four-stage solid configuration. Thus, the FTD-2 results in a low-cost program based on the assumptions of a highly-efficient commercial space company.

Table 37. Program Cost Summary for the FTD-2 System Concept

Development phase costs	$85 M
Test program phase costs	$52 M
Total government team and program management	$67 M
Total contingency (20%)	$41 M
Total test program cost	$245 M

Flight Test Demonstrator Summary

The two examples of a flight test demonstrator provide traceability to the concept vehicles in terms of cost and performance model validation and optimization for aerodynamics, separation, carrier vehicle control and structural loads, and payload to orbit. The flight tests are intended to demonstrate operability including turnaround time, crew size, launch vehicle integration, in-air propulsion start, and on-board mission and flight control. In addition, the two-stage liquid FTD-2 can demonstrate cryogenic handling of liquid oxygen on the ground and launch vehicle storage during the cruise to separation phase.

Table 38 compares the four-stage solid and two-stage liquid flight test demonstrators. The four-stage solid FTD-1 has a higher payload and an estimated higher development cost compared to FTD-2. Based on the assumption that the two-stage liquid FTD-2 can be developed at the same cost as the SpaceX Falcon family of launch vehicles, total cost favors the two-stage liquid FTD-2. The failure risk for the solid FTD-1 is higher but is based on the recent history of payload separation failures of the Taurus launch vehicle. If these problems are solved and typical failure rates prevail, the risks are similar for the two configurations.

The two-stage liquid FTD-2 has a number of advantages for a demonstration. For example, the FTD-2 demonstrates all the necessary operational needs including LOX logistics, storage on ground, and storage in flight. In addition, it is anticipated that the payload could be increased if desired by lengthening the stages according to limits determined in a structural bending loads analysis.

Table 38. Summary of the Flight Test Demonstrator System Concepts

	FTD-1	FTD-2
Launch vehicle propulsion	Solid rockets (Taurus-based)	LOX/RP engines (Falcon-based)
Launch vehicle gross weight	179,474 lb.	81,987 lb.
Payload to LEO	4,562 lb.	2,749 lb.
Launch vehicle total length	99 ft.	81 ft.
Launch vehicle maximum diameter	7.75 ft.	5.5 ft.
Wing span	47.4 ft.	26 ft.
Total costs	$320 M	$245 M
Reliability for flight test 1	77.77%	82.82%
Biggest risk factor	Fairing separation	Off nominal payload insertion

Summary and Conclusions

An analysis of subsonic horizontal launch concepts using existing aircraft designs and technology and utilizing mature and existing launch vehicle designs and technology has been completed. Basic mission objectives for payload, mobility, and responsiveness are achievable, and two flight demonstration approaches are recommended.

Based on the point design vehicle results, payloads up to 15,000 lbs can be obtained using a much larger two-stage system that is empty of fuel at takeoff and utilizing a tanker for in-air fueling of the carrier or the launch vehicle.

The major cost of a subsonic horizontal take-off space launch is the launch vehicle. To use these for horizontal launch, aerodynamic surfaces and other structures are added to enable separation and pull-up maneuvers. These additions, along with the need for a carrier aircraft, have the potential to make horizontal launch a more expensive option. However, horizontal launch provides the potential for improved basing flexibility, covert launch, weather avoidance, and offset launch for orbital intercept and reconnaissance that may outweigh any increased cost.

In conclusion, a useful flight demonstration is possible within a funding profile of less than $350 million over four years. This plan utilizes NASA's Shuttle Carrier Aircraft that will be available after Shuttle's last flight. Following successful demonstration flights, even lower cost horizontal take-off space launch may be possible if, for example, increased flight rates lower the amortization costs, new technology development is funded by government investments, or improved development and operational practices are implemented that follow the examples of SpaceX and Scaled Composites.