Next Generation Spacecraft, Crew Exploration Vehicle

A Special Bibliography From the
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Includes research on reusable launch vehicles, aerospace planes, shuttle replacement, crew/cargo transfer vehicle, related X-craft, orbital space plane, and next generation launch technology.

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20040004384 Nebraska Univ., Lincoln, NE, USA

Design of Experiments for the Thermal Characterization of Metallic Foam

Crittenden, Paul E.; Cole, Kevin D.; Optimal Experiment Design for Thermal Characterization of Functionally Graded Materials; August 18, 2003; In English

Contract(s)/Grant(s): NAG1-01087; No Copyright; Avail: CASI; A03, Hardcopy

Metallic foams are being investigated for possible use in the thermal protection systems of reusable launch vehicles. As a result, the performance of these materials needs to be characterized over a wide range of temperatures and pressures. In this paper a radiation/conduction model is presented for heat transfer in metallic foams. Candidates for the optimal transient experiment to determine the intrinsic properties of the model are found by two methods. First, an optimality criterion is used to find an experiment to find all of the parameters using one heating event. Second, a pair of heating events is used to determine the parameters in which one heating event is optimal for finding the parameters related to conduction, while the other heating event is optimal for finding the parameters associated with radiation. Simulated data containing random noise was analyzed to determine the parameters using both methods. In all cases the parameter estimates could be improved by analyzing a larger data record than suggested by the optimality criterion.

Author

Metal Foams; Heat Transfer

20040003713 NASA Langley Research Center, Hampton, VA, USA

Development of Structural Health Management Technology for Aerospace Vehicles

Prosser, W. H.; [2003]; In English; JANNAF 39th Combustion/27th Airbreathing Propulsion/21st Propulsion Systems Harzards/3rd Modeling and Simulation Joint Subcommittee Meeting, 1-5 Sep. 2003, Colorado Springs, CO, USA; Original contains color illustrations; No Copyright; Avail: CASI; A02, Hardcopy

As part of the overall goal of developing Integrated Vehicle Health Management (IVHM) systems for aerospace vehicles, NASA has focused considerable resources on the development of technologies for Structural Health Management (SHM). The motivations for these efforts are to increase the safety and reliability of aerospace structural systems, while at the same time decreasing operating and maintenance costs. Research and development of SHM technologies has been supported under a variety of programs for both aircraft and spacecraft including the Space Launch Initiative, X-33, Next Generation Launch Technology, and Aviation Safety Program. The major focus of much of the research to date has been on the development and testing of sensor technologies. A wide range of sensor technologies are under consideration including fiber-optic sensors, active and passive acoustic sensors, electromagnetic sensors, wireless sensing systems, MEMS, and nanosensors. Because of their numerous advantages for aerospace applications, most notably being extremely light weight, fiber-optic sensors are one of the leading candidates and have received considerable attention.

Author

Aerospace Vehicles; Systems Health Monitoring; Systems Engineering; Technology Utilization; Remote Sensors

20040001677 Aerospace Corp., USA

Liquid Engine Test Facilities Assessment

Adams, Michael J.; Emdee, Jeffery L.; George, Daweel J.; Peinemann, Manfred; May 03, 2002; In English

Contract(s)/Grant(s): F04701-00-C-0009

Report No.(s): NASA/SE-2002-05-00041-SSC; No Copyright; Avail: CASI; A04, Hardcopy

The John C. Stennis Space Center (SSC) requested The Aerospace Corporation to examine the current testing capability of all existing large liquid engine test facilities located in the USA. That information along with projected liquid rocket engine development was used to examine future liquid rocket engine testing facilities needs in the coming decade. Current domestic liquid engine test facilities capabilities, when examined against engine concepts for the coming decade, indicate there are

ample facilities offering altitude simulation during test. In addition, it was observed that many contractor facilities have limited ambient test capability of larger thrust engines under current consideration. Finally, it was concluded that diminished contractor participation engine development testing will drive this activity to the government sector. Only three facilities are seen as key contributors to engine testing in the coming decade, namely John C. Stennis Space Center (SSC), Marshall Space Flight Center (MSFC), and Air Force Research Laboratory (AFRL). Past rocket engine test experience was evaluated as a possible resource for projecting future engine test needs. A database comprised of various engine models and the level of testing performed to flight qualify those systems for their first flight was constructed. For comparison purposes in this study, development and qualification efforts were totaled and treated as one test program. Based on experience with past Air Force programs, the time on the test stand accounts for typically 50% or more of the total program time. Historical data show that the time to design and develop new engines has increased over the last 40 years, most likely due to scarcer resources in today's funding environment.
Derived from text
Engine Tests; Liquid Propellant Rocket Engines; Test Facilities; Aerospace Systems

20040001153 NASA Langley Research Center, Hampton, VA, USA
Cryopumping in Cryogenic Insulations for a Reusable Launch Vehicle
Johnson, Theodore F.; Weiser, Erik S.; Grimsley, Brian W.; Jensen, Brian J.; [2003]; In English; No Copyright; Avail: CASI; A03, Hardcopy

Testing at cryogenic temperatures was performed to verify the material characteristics and manufacturing processes of reusable propellant tank cryogenic insulations for a Reusable Launch Vehicle (RLV). The unique test apparatus and test methods developed for the investigation of cryopumping in cryogenic insulations are described. Panel level test specimens with various types of cryogenic insulations were subjected to a specific thermal profile where the temperature varied from -262 C to 21 C. Cryopumping occurred if the interior temperature of the specimen exhibited abnormal temperature fluctuations, such as a sudden decrease in temperature during the heating phase.
Author
Cryogenic Temperature; Cryopumping; Insulation; Reusable Launch Vehicles; Manufacturing

20040000905 NASA Stennis Space Center, Bay Saint Louis, MS, USA
Improved Testing Capability and Adaptability Through the Use of Wireless Sensors
Solano, Wanda M.; April 14, 2003; In English; 2003 Propulsion Measurement Sensor Development Workshop, 13-15 May 2003, Huntsville, AL, USA; Original contains color illustrations
Report No.(s): SE-2003-04-00020-SSC; No Copyright; Avail: CASI; A02, Hardcopy

From the first Saturn V rocket booster (S-II-T) testing in 1966 and the routine Space Shuttle Main Engine (SSME) testing beginning in 1975, to more recent test programs such as the X-33 Aerospike Engine, the Integrated Powerhead Development (IPD) program, and the Hybrid Sounding Rocket (HYSR), Stennis Space Center (SSC) continues to be a premier location for conducting large-scale testing. Central to each test program is the capability for sensor systems to deliver reliable measurements and high quality data, while also providing a means to monitor the test stand area to the highest degree of safety and sustainability. Sensor wiring is routed along piping and through cable trenches, making its way from the engine test area, through the test stand area and to the signal conditioning building before final transfer to the test control center. When sensor requirements lie outside the reach of the routine sensor cable routing, the use of wireless sensor networks becomes particularly attractive due to their versatility and ease of installation. As part of an on-going effort to enhance the testing capabilities of Stennis Space Center, the Test Technology and Development group has found numerous applications for its sensor-adaptable wireless sensor suite. While not intended for critical engine measurements or control loops, in-house hardware and software development of the sensor suite can provide improved testing capability for a range of applications including the safety monitoring of propellant storage barrels and as an experimental test-bed for embedded health monitoring paradigms.
Author
Engine Tests; Engine Monitoring Instruments; Instrument Packages

20040000874 Lockheed Martin Space Operations, Bay Saint Louis, MS, USA
NASA Stennis Space Center V and V Capabilities Overview
Ryan, Robert; October 22, 2001; In English; MAPPS-ASPRS Conference: Measuring the Earth-Digital Elevation Technologies and Applications, 31 Oct. - 2 Nov. 2001, Saint Petersburg, FL, USA
Contract(s)/Grant(s): NAS13-650
Report No.(s): SE-2001-10-00062-SSC; No Copyright; Avail: CASI; A03, Hardcopy

The objective are provide independent field measurement ground capability for satellite and aircraft mounted in-flight sensor validation. NASA Stennis is considering expanding its Verification and Validation (V&V) range to support LIDAR and digital camera characterization. Soliciting input for requirements and suggestion.
Derived from text
Digital Cameras; Optical Radar; Aircraft Instruments

20040000627 NASA Glenn Research Center, Cleveland, OH, USA
Update on the Development and Capabilities of Unique Structural Seal Test Rigs
DeMange, Jeffrey J.; Dunlap, Patrick H., Jr.; Steinetz, Bruce M.; Breen, Daniel P.; Robbie, Malcolm G.; 2002 NASA Seal/Secondary Air System Workshop; September 2003; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

High temperature structural seals are necessary in many aerospace and aeronautical applications to minimize any detrimental effects originating from undesired leakage. The NASA Glenn Research Center has been and continues to be a pioneer in the development and evaluation of these types of seals. The current focus for the development of structural seals is for the 3rd Generation Reusable Launch Vehicle (RLV), which is scheduled to replace the current space shuttle system by 2025. Specific areas of development under this program include seals for propulsion systems (such as the hypersonic air-breathing ISTAR engine concept based upon Rocket Based Combined Cycle technology) and control surface seals for spacecraft including the autonomous rescue X-38 Crew Return Vehicle and the X-37 Space Maneuver Vehicle.
Derived from text
X-38 Crew Return Vehicle; X-37 Vehicle; Sealing; Control Surfaces; Air Breathing Engines; Reusable Launch Vehicles

20040000626 NASA Glenn Research Center, Cleveland, OH, USA
Third Generation RLV Structural Seal Development Programs at NASA GRC
Dunlap, Patrick H., Jr.; Steinetz, Bruce M.; DeMange, Jeffrey J.; 2002 NASA Seal/Secondary Air System Workshop; September 2003; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

NASA is currently developing technologies for the 3rd Generation Reusable Launch Vehicle (RLV) that is being designed to enter service around the year 2025. In particular, NASA's Glenn Research Center (GRC) is working on advanced high temperature structural seal designs including propulsion system and control surface seals. Propulsion system seals are required along the edges of movable panels in advanced engines, while control surface seals seal the edges and hinge lines of movables flaps and elevons on the vehicle. The overall goal is to develop reusable, resilient seals capable of operating at temperatures up to 2000 F. High temperature seal preloading devices (e.g., springs) are also being evaluated as a means of improving seal resiliency. In order to evaluate existing and potential new seal designs, GRC has designed and is installing several new test rigs capable of simulating the types of conditions that the seals would endure during service including temperatures, pressures, and scrubbing. Two new rigs, the hot compression test rig and the hot scrub test rig, will be used to perform seal compression and scrub tests for many cycles at temperatures up to 3000 F. Another new test rig allows simultaneous flow and scrub tests to be performed on the seals at room temperature to evaluate how the flow blocking performance of the seals varies as they accumulate damage during scrubbing. This presentation will give an overview of these advanced seal development efforts.
Author
Reusable Launch Vehicles; Structural Design; Sealing; Control Surfaces

20030112990 NASA Marshall Space Flight Center, Huntsville, AL, USA
Using CFD as Rocket Injector Design Tool: Recent Progress at Marshall Space Flight Center
Tucker, Kevin; West, Jeff; Williams, Robert; Lin, Jeff; Rocker, Marvin; Canabal, Francisco; Robles, Bryan; Garcia, Robert; Chenoweth, James; [2003]; In English; 5th International Symposium on Liquid Space Propulsion Long Life Combustion Devices Technology, 27-30 Oct. 2003, Chatanooga, TN, USA; Copyright; Avail: CASI; A03, Hardcopy

The choice of tools used for injector design is in a transitional phase between exclusive reliance on the empirically based correlations and extensive use of computational fluid dynamics (CFD). The Next Generation Launch Technology (NGLT) Program goals emphasizing lower costs and increased reliability have produced a need to enable CFD as an injector design tool in a shorter time frame. This is the primary objective of the Staged Combustor Injector Technology Task currently under way at Marshall Space Flight Center (MSFC). The documentation of this effort begins with a very brief status of current injector design tools. MSFC's vision for use of CFD as a tool for combustion devices design is stated and discussed with emphasis on the injector. The concept of the Simulation Readiness Level (SRL), comprised of solution fidelity, robustness and accuracy, is introduced and discussed. This quantitative measurement is used to establish the gap between the current state of

demonstrated capability and that necessary for regular use in the design process. MSFC's view of the validation process is presented and issues associated with obtaining the necessary data are noted and discussed. Three current experimental efforts aimed at generating validation data are presented. The importance of uncertainty analysis to understand the data quality is also demonstrated. First, a brief status of current injector design tools is provided as context for the current effort. Next, the MSFC vision for using CFD as an injector design tool is stated. A generic CFD-based injector design methodology is also outlined and briefly discussed. Three areas where MSFC is using injector CFD analyses for program support will be discussed. These include the Integrated Powerhead Development (IPD) engine which uses hydrogen and oxygen propellants in a full flow staged combustion (FFSC) cycle and the TR-107 and the RS84 engine both of which use RP-1 and oxygen in an ORSC cycle. Finally, an attempt is made to objectively summarize what progress has been made at MSFC in enabling CFD as an injector design tool.
Author
Computational Fluid Dynamics; Rocket Engine Design; Research Facilities; Injectors

20030112784 NASA Glenn Research Center, Cleveland, OH, USA
Computations of Internal and External Axisymmetric Nozzle Aerodynamics at Transonic Speeds
Dalbello, Teryn; Georgiadis, Nicholas; Yoder, Dennis; Keith, Theo; November 2003; In English; 39th Combustion/27th Airbreathing Propulsion/21st Propulsion Systems Hazards/Third Modeling and Simulation Joint Subcommittee Meeting, 1-5 Dec. 2003, Colorado Springs, CO, USA; Original contains color and black and white illustrations
Contract(s)/Grant(s): WBS 22-708-73-22
Report No.(s): NASA/TM-2003-212731; E-14256; NAS 1.15:212731; Copyright; Avail: CASI; A03, Hardcopy

Computational Fluid Dynamics (CFD) analyses of axisymmetric circular-arc boattail nozzles have been completed in support of NASA's Next Generation Launch Technology Program to investigate the effects of high-speed nozzle geometries on the nozzle internal flow and the surrounding boattail regions. These computations span the very difficult transonic flight regime, with shock-induced separations and strong adverse pressure gradients. External afterbody and internal nozzle pressure distributions computed with the Wind code are compared with experimental data. A range of turbulence models were examined in Wind, including an Explicit Algebraic Stress model (EASM). Computations on two nozzle geometries have been completed at freestream Mach numbers ranging from 0.6 to 0.9, driven by nozzle pressure ratios (NPR) ranging from 2.9 to 5. Results obtained on converging-only geometry indicate reasonable agreement to experimental data, with the EASM and Shear Stress Transport (SST) turbulence models providing the best agreement. Calculations completed on a converging-diverging geometry involving large-scale internal flow separation did not converge to a true steady-state solution when run with variable timestepping (steady-state). Calculations obtained using constant timestepping (time-accurate) indicate less variations in flow properties compared with steady-state solutions. This failure to converge to a steady-state solution was found to be the result of difficulties in using variable time-stepping with large-scale separations present in the flow. Nevertheless, time-averaged boattail surface pressure coefficient and internal nozzle pressures show fairly good agreement with experimental data. The SST turbulence model demonstrates the best over-all agreement with experimental data.
Author
Convergent-Divergent Nozzles; Nozzle Flow; Nozzle Geometry; Boattails

20030112231 NASA Marshall Space Flight Center, Huntsville, AL, USA
Orbital Space Plane Cost Credibility
Creech, Steve; January 9, 2003; In English; 54th International Astronautical Congress, 29 Sep. - 3 Oct. 2003, Bremen, Germany; No Copyright; Avail: Other Sources; Abstract Only

NASA's largest new start development program is the Orbital Space Plane (OSP) Program. The program is currently in the formulation stage. One of the critical issues to be resolved, prior to initiating full-scale development, is establishing cost credibility of NASA s budget estimates for development, production, and operations of the OSP. This paper will discuss the processes, tools, and methodologies that NASA, along with its industry partners, are implementing to assure cost credibility for the OSP program. Results of benchmarking of current tools and the development of new cost estimating capabilities and approaches will be discussed.
Author
Cost Estimates; Aerospace Planes

20030111907 NASA Marshall Space Flight Center, Huntsville, AL, USA
NASA's Next Generation Launch Technology Program - Strategy and Plans
Hueter, Uwe; [2003]; In English; 54th International Astronautical Congress, 29 Sep. - 3 Oct. 2003, Bremen, Germany
Report No.(s): 1AC-03-V.5.01; No Copyright; Avail: CASI; A02, Hardcopy

The National Aeronautics and Space Administration established a new program office, Next Generation Launch Technology (NGLT) Program Office, last year to pursue technologies for future space launch systems. NGLT will fund research in key technology areas such as propulsion, launch vehicles, operations and system analyses. NGLT is part of NASA s Integrated Space Technology Plan. The NGLT Program is sponsored by NASA s Office of Aerospace Technology and is part of the Space Launch Initiative theme that includes both NGLT and Orbital Space Plane. NGLT will focus on technology development to increase safety and reliability and reduce overall costs associated with building, flying and maintaining the nation s next-generations of space launch vehicles. These investments will be guided by systems engineering and analysis with a focus on the needs of National customers.
Author
Aerospace Engineering; Aerospace Planes; Spacecraft Launching

20030111798 NASA Marshall Space Flight Center, Huntsville, AL, USA
Orbital Space Plane Program Status
Dumbacher, Daniel L.; [2003]; In English; 54th International Astronautical Congress, 29 Sep. - 3 Oct. 2003, Bremen, Germany; No Copyright; Avail: Other Sources; Abstract Only

The Orbital Space Plane Program is an integral part of NASA's Integrated Space Transportation Program (ISTP). The ISTP consists of three major programs: Space Shuttle, Orbital Space Plane, and Next Generation Launch Technology. The Orbital Space Plane (OSP) Program will develop a new Crew Transfer Vehicle (CTV) with multipurpose utility for the Agency. The CTV will complement and back up the Space Shuttle by taking crews to and from the International Space Station (ISS), as well as enable a transition path to future reusable launch vehicle systems. In the CTV development cycle, around 2010 it will be used as a Crew Return Vehicle (CRV). The OSP will be launched on an Evolved Expendable Launch Vehicle (EELV). NASA is in the process of establishing Level 1 Requirements and initiating concept studies. Ongoing flight demonstrators will continue, while new flight demonstrator projects will begin. The OSP Program contains two elements: (1) Technology and Demonstrations, and (2) Design, Development, and Production. The OSP Design, Development, and Production element will enter the Formulation Phase in FY03. Per NASA Procedures and Guidelines 7120.5B, the Formulation Phase will be utilized to establish the Program schedule and budget plans. Current budget planning is based on Phase A concept studies being conducted in FY03 and FY04, preliminary design activities conducted in FY04 and FY05, and a Preliminary Design Review in FY05. An OSP full-scale development decision will be made in FY05. At that point, a conclusion to proceed will result in the OSP Program transitioning from the Formulation Phase to the Development Phase.
Author
Aerospace Planes; NASA Space Programs; Launch Vehicles; Space Shuttles; Space Shuttle Orbiters; Reusable Launch Vehicles

20030111797 Lockheed Martin Space Mission Systems and Services, Littleton, CO, USA
Orbital Space Plane (OSP) Program
McKenzie, Patrick M.; September 19, 2003; In English; 54th International Astronautical Congress, 29 Sep. - 3 Oct. 2003, Bremen, Germany
Contract(s)/Grant(s): NAS8-01098; No Copyright; Avail: CASI; A02, Hardcopy

Lockheed Martin has been an active participant in NASA's Space Launch Initiative (SLI) programs over the past several years. SLI, part of NASA's Integrated Space Transportation Plan (ISTP), was restructured in November of 2002 to focus the overall theme of safer, more afford-able space transportation along two paths - the Orbital Space Plane Program and the Next Generation Launch Technology programs. The Orbital Space Plane Program has the goal of providing rescue capability from the International Space Station by 2008 and transfer capability for crew (and limited cargo) by 2012. The Next Generation Launch Technology program is combining research and development efforts from the 2nd Generation Reusable Launch Vehicle (2GRLV) program with cutting-edge, advanced space transportation programs (previously designated 3rd Generation) into one program aimed at enabling safe, reliable, cost-effective reusable launch systems by the middle of the next decade. Lockheed Martin is one of three prime contractors working to bring Orbital Space Plane system concepts to a system definition level of maturity by December of 2003. This paper and presentation will update the international community on the progress of the' OSP program, from an industry perspective, and provide insights into Lockheed Martin's role in enabling the vision of a safer, more affordable means of taking people to and from space.
Author
Space Transportation; International Space Station; Systems Analysis; Aerospace Planes

20030111778 NASA Marshall Space Flight Center, Huntsville, AL, USA

Orbital Space Plane Program Flight Demonstrators Status

Turner, Susan G.; [2003]; In English; 54th International Astronautical Congress, 29 Sep. - 3 Oct. 2003, Bremen, Germany; No Copyright; Avail: Other Sources; Abstract Only

Under the Orbital Space Plane Program, NASA is currently pursuing the maturation of technologies via three flight demonstrators - DART (Demonstration of Autonomous Rendezvous Technology), X-37, and PAD (Pad Abort Demonstrator). Flight demonstrators provide the opportunity to test key technologies in their actual working environment. These flight demonstrators are required at this stage to mature technologies needed to support full-scale development design of a future competitively selected Orbital Space.

Derived from text

Flight Tests; Flight Simulators; Autonomy; Aerospace Planes

20030108286 NASA Glenn Research Center, Cleveland, OH, USA

Durability of Membrane Electrode Assemblies (MEAs) in PEM Fuel Cells Operated on Pure Hydrogen and Oxygen

Stanic, Vesna; Braun, James; Hoberecht, Mark; [2003]; In English; International Energy Conversion Engineering Conference, 17-21 Aug. 2003, Portsmouth, VA, USA

Contract(s)/Grant(s): NAS3-02093; 22-721-22-09; No Copyright; Avail: CASI; A02, Hardcopy

Proton exchange membrane (PEM) fuel cells are energy sources that have the potential to replace alkaline fuel cells for space programs. Broad power ranges, high peak-to-nominal power capabilities, low maintenance costs, and the promise of increased life are the major advantages of PEM technology in comparison to alkaline technology. The probability of PEM fuel cells replacing alkaline fuel cells for space applications will increase if the promise of increased life is verified by achieving a minimum of 10,000 hours of operating life. Durability plays an important role in the process of evaluation and selection of MEAs for Teledyne s Phase I contract with the NASA Glenn Research Center entitled Proton Exchange Membrane Fuel cell (PEMFC) Power Plant Technology Development for 2nd Generation Reusable Launch Vehicles (RLVs). For this contract, MEAs that are typically used for H2/air operation were selected as potential candidates for H2/O2 PEM fuel cells because their catalysts have properties suitable for O2 operation. They were purchased from several well-established MEA manufacturers who are world leaders in the manufacturing of diverse products and have committed extensive resources in an attempt to develop and fully commercialize MEA technology. A total of twelve MEAs used in H2/air operation were initially identified from these manufacturers. Based on the manufacturers specifications, nine of these were selected for evaluation. Since 10,000 hours is almost equivalent to 14 months, it was not possible to perform continuous testing with each MEA selected during Phase I of the contract. Because of the lack of time, a screening test on each MEA was performed for 400 hours under accelerated test conditions. The major criterion for an MEA pass or fail of the screening test was the gas crossover rate. If the gas crossover rate was higher than the membrane intrinsic permeability after 400 hours of testing, it was considered that the MEA had failed the test. Three types of MEAs out of the nine total membranes failed the test. The evaluation results showed that fuel cell operating conditions (current, pressure, stoichiometric flow rates) were the parameters that influenced the durability of MEAs. In addition, the durability test results indicated that the type of membrane was also an important parameter for MEA durability. At accelerated test conditions, the MEAs with casted membranes failed during the 400 hour test. However, the MEAs prepared from the casted membrane with support as well as extruded membranes, both passed the 400h durability test at accelerated operating test conditions. As a result of the MEA accelerated durability tests, four MEAs were selected for further endurance testing. These tests are being carried out with four-cell stacks under nominal fuel cell operating conditions.

Author

Durability; Electrodes; Membranes; Protons; Manufacturing; Hydrogen Oxygen Fuel Cells

20030108283 NASA Glenn Research Center, Cleveland, OH, USA

Performance Evaluation of Electrochem's PEM Fuel Cell Power Plant for NASA's 2nd Generation Reusable Launch Vehicle

Kimble, Michael C.; Hoberecht, Mark; [2003]; In English; International Energy Conversion Engineering Conference, 17-21 Aug. 2003, Portsmouth, VA, USA

Contract(s)/Grant(s): NAS3-02092; 22-721-22-09; No Copyright; Avail: CASI; A02, Hardcopy

NASA's Next Generation Launch Technology (NGLT) program is being developed to meet national needs for civil and commercial space access with goals of reducing the launch costs, increasing the reliability, and reducing the maintenance and operating costs. To this end, NASA is considering an all- electric capability for NGLT vehicles requiring advanced electrical power generation technology at a nominal 20 kW level with peak power capabilities six times the nominal power. The proton exchange membrane (PEM) fuel cell has been identified as a viable candidate to supply this electrical power; however, several

technology aspects need to be assessed. Electrochem, Inc., under contract to NASA, has developed a breadboard power generator to address these technical issues with the goal of maximizing the system reliability while minimizing the cost and system complexity. This breadboard generator operates with dry hydrogen and oxygen gas using eductors to recirculate the gases eliminating gas humidification and blowers from the system. Except for a coolant pump, the system design incorporates passive components allowing the fuel cell to readily follow a duty cycle profile and that may operate at high 6:1 peak power levels for 30 second durations. Performance data of the fuel cell stack along with system performance is presented to highlight the benefits of the fuel cell stack design and system design for NGLT vehicles.
Author
Reusable Launch Vehicles; Protons; Membranes; Fuel Cell Power Plants; Performance Prediction; NASA Space Programs; Fuel Cells

20030107869 NASA Stennis Space Center, Bay Saint Louis, MS, USA
Engineering the Future of Full-Scale Propulsion Testing
Ryan, H. M.; Solano, W.; Holland, R.; Saint Cyr, W.; Rahman, S.; [2001]; In English; 39th AIAA Aerospace Sciences Meeting and Exhibit, 8-11 Jan. 2001, Reno, NV, USA
Report No.(s): AIAA Paper 2001-0746; SE-2001-01-00001-SSC; Copyright; Avail: CASI; A03, Hardcopy

Year 2000 has been an active one for rocket propulsion testing at the NASA John C. Stennis Space Center. This paper highlights several major test facilites for large-scale propulsion devices, and summarizes the varied nature of the recent test projects conducted at the Stennis Space Center (SSC) such as the X-33 Aerospike Engine, Ultra Low Cost Engine (ULCE) thrust chamber program, and the Hybrid Sounding Rocket (HYSR) program. Further, an overview of relevant engineering capabilities and technology challenges in conducting full-scale propulsion testing are outlined.
Author
Engine Tests; Full Scale Tests; Propellant Tests; Test Facilities

20030106071 Lockheed Martin Space Systems Co., Huntsville, AL, USA
Orbital Space Plane (OSP) Program at Lockheed Martin
Ford, Robert; September 19, 2003; In English; Space 2003, 24 Sep.2003, Long Beach, CA, USA
Contract(s)/Grant(s): NAS8-01098; No Copyright; Avail: CASI; A02, Hardcopy

Lockheed Martin has been an active participant in NASA's Space Launch Initiative (SLI) programs over the past several years. SLI, part of NASA's Integrated Space Transportation Plan (ISTP), was restructured in November 2002 to focus the overall theme of safer, more affordable space transportation along two paths the Orbital Space Plane (OSP) and the Next Generation Launch Technology programs. The Orbital Space Plane program has the goal of providing rescue capability from the International Space Station by 2008 or earlier and transfer capability for crew (and contingency cargo) by 2012. The Next Generation Launch Technology program is combining research and development efforts from the 2d Generation Reusable Launch Vehicle (2GRLV) program with cutting-edge, advanced space transportation programs (previously designated 31d Generation) into one program aimed at enabling safe, reliable, cost-effective reusable launch systems by the middle of the next decade. Lockheed Martin is one of three prime contractors working to bring Orbital Space Plane system concepts to a system design level of maturity by December 2003. This paper and presentation will update the aerospace community on the progress of the OSP program, from an industry perspective, and provide insights into Lockheed Martin's role in enabling the vision of a safer, more affordable means of taking people to and from space.
Author
Aerospace Planes; Space Transportation; Reusable Launch Vehicles; NASA Space Programs; Spacecraft Orbits; Spacecraft Configurations

20030093628 Vanderbilt Univ., TN, USA
Impedance-Based Structural Health Monitoring for Composite Laminates at Cryogenic Environments
Tseng, Kevin; The 2002 NASA Faculty Fellowship Program Research Reports; April 2003; In English
Contract(s)/Grant(s): NAG8-1859; No Copyright; Avail: CASI; C01, CD-ROM; A01, Hardcopy

One of the important ways of increasing the payload in a reusable launch vehicle (RLV) is to replace heavy metallic materials by lightweight composite laminates. Among various parts and systems of the RLV, this project focuses on tanks containing cryogenic fuel. Historically, aluminum alloys have been used as the materials to construct fuel tanks for launch vehicles. To replace aluminum alloys with composite laminates or honeycomb materials, engineers have to make sure that the composites are free of defects before, during, and after launch. In addition to robust design and manufacturing procedures, the

performance of the composite structures needs to be monitored constantly.In recent years, the impedance-based health monitoring technique has shown its promise in many applications. This technique makes use of the special properties of smart piezoelectric materials to identify the change of material properties due to the nucleation and progression of damage. The piezoceramic patch serves as a sensor and an actuator simultaneously. The piezoelectric patch is bonded onto an existing structure or embedded into a new structure and electrically excited at high frequencies. The signature (impedance or admittance) is extracted as a function of the exciting frequency and is compared with the baseline signature of the healthy state. The damage is quantified using root mean square deviation (RMSD) in the impedance signatures with respect to the baseline signature. A major advantage of this technique is that the procedure is nondestructive in nature and does not perturb the properties and performance of the materials and structures. This project aims at applying the impedance-based nondestructive testing technique to the damage identification of composite laminates at cryogenic temperature.
Author
Laminates; Reusable Launch Vehicles; Piezoelectric Actuators

20030093602 Rose-Hulman Inst. of Tech., Terre Haute, IN, USA
Fuzzy Logic Trajectory Design and Guidance for Terminal Area Energy Management
Burchett, Bradley; The 2002 NASA Faculty Fellowship Program Research Reports; April 2003; In English
Contract(s)/Grant(s): NAG8-1859; No Copyright; Avail: CASI; C01, CD-ROM; A01, Hardcopy

The second generation reusable launch vehicle will leverage many new technologies to make flight to low earth orbit safer and more cost effective. One important capability will be completely autonomous flight during reentry and landing, thus making it unnecessary to man the vehicle for cargo missions with stringent weight constraints. Implementation of sophisticated new guidance and control methods will enable the vehicle to return to earth under less than favorable conditions. The return to earth consists of three phases--Entry, Terminal Area Energy Management (TAEM), and Approach and Landing. The Space Shuttle is programmed to fly all three phases of flight automatically, and under normal circumstances the astronaut-pilot takes manual control only during the Approach and Landing phase. The automatic control algorithms used in the Shuttle for TAEM and Approach and Landing have been developed over the past 30 years. They are computationally efficient, and based on careful study of the spacecraft's flight dynamics, and heuristic reasoning. The gliding return trajectory is planned prior to the mission, and only minor adjustments are made during flight for perturbations in the vehicle energy state. With the advent of the X-33 and X-34 technology demonstration vehicles, several authors investigated implementing advanced control methods to provide autonomous real-time design of gliding return trajectories thus enhancing the ability of the vehicle to adjust to unusual energy states. The bulk of work published to date deals primarily with the approach and landing phase of flight where changes in heading angle are small, and range to the runway is monotonically decreasing. These benign flight conditions allow for model simplification and fairly straightforward optimization. This project focuses on the TAEM phase of flight where mathematically precise methods have produced limited results. Fuzzy Logic methods are used to make onboard autonomous gliding return trajectory design robust to a wider energy envelope, and the possibility of control surface failures, thus increasing the flexibility of unmanned gliding recovery and landing.
Derived from text
Fuzzy Systems; Logic Design; Trajectories; Terminal Area Energy Management; Automatic Control

20030093594 Embry-Riddle Aeronautical Univ., USA
System Identification of X-33 Neural Network
Aggarwal, Shiv; The 2002 NASA Faculty Fellowship Program Research Reports; April 2003; In English
Contract(s)/Grant(s): NAG8-1859; No Copyright; Avail: CASI; C01, CD-ROM; A01, Hardcopy

Modern flight control research has improved spacecraft survivability as its goal. To this end we need to have a failure detection system on board. In case the spacecraft is performing imperfectly, reconfiguration of control is needed. For that purpose we need to have parameter identification of spacecraft dynamics. Parameter identification of a system is called system identification. We treat the system as a black box which receives some inputs that lead to some outputs. The question is: what kind of parameters for a particular black box can correlate the observed inputs and outputs? Can these parameters help us to predict the outputs for a new given set of inputs? This is the basic problem of system identification. The X33 was supposed to have the onboard capability of evaluating the current performance and if needed to take the corrective measures to adapt to desired performance. The X33 is comprised of both rocket and aircraft vehicle design characteristics and requires, in general, analytical methods for evaluating its flight performance. Its flight consists of four phases: ascent, transition, entry and TAEM (Terminal Area Energy Management). It spends about 200 seconds in ascent phase, reaching an altitude of about 180,000 feet and a speed of about 10 to 15 Mach. During the transition phase which lasts only about 30 seconds, its altitude may increase to about 190,000 feet but its speed is reduced to about 9 Mach. At the beginning of this phase, the Main Engine

is Cut Off (MECO) and the control is reconfigured with the help of aerosurfaces (four elevons, two flaps and two rudders) and reaction control system (RCS). The entry phase brings down the altitude of X33 to about 90,000 feet and its speed to about Mach 3. It spends about 250 seconds in this phase. Main engine is still cut off and the vehicle is controlled by complex maneuvers of aerosurfaces. The last phase TAEM lasts for about 450 seconds and the altitude and speed, both are reduced to zero. The present attempt, as a start, focuses only on the entry phase. Since the main engine remains cut off in this phase, there is no thrust acting on the system. This considerably simplifies the equations of motion. We introduce another simplification by assuming the system to be linear after some non-linearities are removed analytically from our consideration. Under these assumptions, the problem could be solved by Classical Statistics by employing the least sum of squares approach. Instead we chose to use the Neural Network method. This method has many advantages. It is modern, more efficient, can be adapted to work even when the assumptions are diluted. In fact, Neural Networks try to model the human brain and are capable of pattern recognition.
Derived from text
Aircraft Design; Design Analysis; Flight Characteristics; Flight Control; Neural Nets; X-33 Reusable Launch Vehicle

20030093586 Missouri Univ., Rolla, MO, USA
Adaptive Critic Neural Network-Based Terminal Area Energy Management and Approach and Landing Guidance
Grantham, Katie; The 2002 NASA Faculty Fellowship Program Research Reports; April 2003; In English
Contract(s)/Grant(s): NAG8-1859; No Copyright; Avail: CASI; C01, CD-ROM; A02, Hardcopy

Reusable Launch Vehicles (RLVs) have different mission requirements than the Space Shuttle, which is used for benchmark guidance design. Therefore, alternative Terminal Area Energy Management (TAEM) and Approach and Landing (A/L) Guidance schemes can be examined in the interest of cost reduction. A neural network based solution for a finite horizon trajectory optimization problem is presented in this paper. In this approach the optimal trajectory of the vehicle is produced by adaptive critic based neural networks, which were trained off-line to maintain a gradual glideslope.
Derived from text
Aircraft Guidance; Aircraft Landing; Neural Nets; Reusable Launch Vehicles; Terminal Area Energy Management

20030093585 Alabama Univ., Huntsville, AL, USA
An Improved RLV Stability Analysis Via a Continuation Approach
Friedman, Mark; The 2002 NASA Faculty Fellowship Program Research Reports; April 2003; In English
Contract(s)/Grant(s): NAG8-1859; No Copyright; Avail: CASI; C01, CD-ROM; A01, Hardcopy

The continuation of Invariant Subspaces (CIS) algorithm produces a smooth orthogonal similarity transformation to block triangular form of A(s). A practical CIS algorithm and stability analysis of the X-33 are described.
Derived from text
Algorithms; Invariant Imbeddings; X-33 Reusable Launch Vehicle; Stability Tests; Continuity (Mathematics); Branching (Mathematics)

20030093576 Alabama Univ., Tuscaloosa, AL, USA, Alabama Univ., Huntsville, AL, USA
The 2002 NASA Faculty Fellowship Program Research Reports
Bland, J., Compiler; April 2003; In English; Original contains color and black and white illustrations
Contract(s)/Grant(s): NAG8-1859
Report No.(s): NASA/CR-2003-212397; M-1074; NAS 1.26:212397; No Copyright; Avail: CASI; C01, CD-ROM

Contents include the following: System Identification of X-33. Neural Network Advanced Ceramic Technology for Space Applications at NASA MSFC. Developing a MATLAB-Based Tool for Visualization and Transformation. Subsurface Stress Fields in Single Crystal (Anisotropic). Contacts Our Space Future: A Challenge to the Conceptual Artist Concept Art for Presentation and Education. Identification and Characterization of Extremophile Microorganisms. Significant to Astrobiology. Mathematical Investigation of Gamma Ray and Neutron. Absorption Grid Patterns for Homeland Defense-Related Fourier Imaging Systems. The Potential of Microwave Radiation for Processing Martian Soil. Fuzzy Logic Trajectory Design and Guidance for Terminal Area.
Derived from text
Active Control; Combustion Chambers; Cylindrical Bodies; Data Processing; Design to Cost

20030079974 NASA Langley Research Center, Hampton, VA, USA

Hypervelocity Impact Test Results for a Metallic Thermal Protection System

Karr, Katherine L.; Poteet, Carl C.; Blosser, Max L.; August 2003; In English; Original contains color and black and white illustrations

Contract(s)/Grant(s): 721-21-87-07

Report No.(s): NASA/TM-2003-212440; L-18192; NAS 1.15:212440; No Copyright; Avail: CASI; A05, Hardcopy

Hypervelocity impact tests have been performed on specimens representing metallic thermal protection systems (TPS) developed at NASA Langley Research Center for use on next-generation reusable launch vehicles (RLV). The majority of the specimens tested consists of a foil gauge exterior honeycomb panel, composed of either Inconel 617 or Ti-6Al-4V, backed with 2.0 in. of fibrous insulation and a final Ti-6Al-4V foil layer. Other tested specimens include titanium multi-wall sandwich coupons as well as TPS using a second honeycomb sandwich in place of the foil backing. Hypervelocity impact tests were performed at the NASA Marshall Space Flight Center Orbital Debris Simulation Facility. An improved test fixture was designed and fabricated to hold specimens firmly in place during impact. Projectile diameter, honeycomb sandwich material, honeycomb sandwich facesheet thickness, and honeycomb core cell size were examined to determine the influence of TPS configuration on the level of protection provided to the substructure (crew, cabin, fuel tank, etc.) against micrometeoroid or orbit debris impacts. Pictures and descriptions of the damage to each specimen are included.

Author

Hypervelocity Impact; Impact Tests; Thermal Protection; Fabrication; Intermetallics

20030077847 NASA Marshall Space Flight Center, Huntsville, AL, USA

Experimental Performance Evaluation of a Supersonic Turbine for Rocket Engine Applications

Snellgrove, Lauren M.; Griffin, Lisa W.; Sieja, James P.; Huber, Frank W.; July 08, 2003; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul. 2003, Huntsville, AL, USA

Report No.(s): AIAA Paper 2003-4918; Copyright; Avail: CASI; A03, Hardcopy

In order to mitigate the risk of rocket propulsion development, efficient, accurate, detailed fluid dynamics analysis and testing of the turbomachinery is necessary. To support this requirement, a task was developed at NASA Marshall Space Flight Center (MSFC) to improve turbine aerodynamic performance through the application of advanced design and analysis tools. These tools were applied to optimize a supersonic turbine design suitable for a reusable launch vehicle (RLV). The hot gas path and blading were redesigned-to obtain an increased efficiency. The goal of the demonstration was to increase the total-to-static efficiency of the turbine by eight points over the baseline design. A sub-scale, cold flow test article modeling the final optimized turbine was designed, manufactured, and tested in air at MSFC s Turbine Airflow Facility. Extensive on- and off-design point performance data, steady-state data, and unsteady blade loading data were collected during testing.

Author

Supersonic Turbines; Reusable Launch Vehicles; Turbine Blades; Aerodynamic Characteristics

20030075777 NASA Ames Research Center, Moffett Field, CA, USA

Accelerating the Design of Space Vehicles

Laufenberg, Larry, Editor; Infusion; August 08, 2003; In English

Contract(s)/Grant(s): UPN 704-00-00; No Copyright; Avail: CASI; A01, Hardcopy

One of NASA's key goals is to increase the safety and reduce the cost of space transportation. Thus, a key element of NASA's new Integrated Space Transportation Plan is to develop new propulsion, structures, and operations for future generations of reusable launch vehicles (RLVs). As part of this effort to develop the next RLV, the ClCT Program's Computing, Networking, and Information Systems (CNIS) Project is developing and demonstrating collaborative software technologies that use the collective power of the NASA Grid to accelerate spacecraft design. One of these technologies, called AeroDB, automates the execution and monitoring of computational fluid dynamics (CFD) parameter studies on the NASA Grid. About the NASA Grid The NASA Grid, or Information Power Grid,. is being developed to leverage the distributed resources of NASA's many computers. instruments, simulators, and data storage systems. The goal is to use these combined resources to sdve difficult NASA challenges, such as iimulating the entire flight of a space vehicle from ascent to descent.To realize the vision of the NASA Grid, the CNIS Project is developing the software framework and protocols for building domain-specific environments and interfaces, new Grid services based on emerging industry standards, and advanced networking and computing testbeds to support new Grid-based applications such as AeroDB.

Author

Reusable Launch Vehicles; Computational Fluid Dynamics

20030071272 Aerojet-General Corp., Sacramento, CA, USA

Non-Toxic Dual Thrust Reaction Control Engine Development for On-Orbit APS Applications

Robinson, Philip J.; Veith, Eric M.; [2003]; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 20-23 Jul. 2003, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

A non-toxic dual thrust proof-of-concept demonstration engine was successfully tested at the Aerojet Sacramento facility under a technology contract sponsored by the National Aeronautics and Space Administration's (NASA) Marshall Space Flight Center (MSFC). The goals of the NASA MSFC contract (NAS8-01109) were to develop and expand the technical maturity of a non-toxic, on-orbit auxiliary propulsion system (APS) thruster under the Next Generation Launch Technology (NGLT) program. The demonstration engine utilized the existing Kistler K-1 870 lbf LOX/Ethanol orbital maneuvering engine (O m) coupled with some special test equipment (STE) that enabled engine operation at 870 lbf in the primary mode and 25 lbf in the vernier mode. Ambient testing in primary mode varied mixture ratio (MR) from 1.28 to 1.71 and chamber pressure (P(c) from 110 to 181 psia, and evaluated electrical pulse widths (EPW) of 0.080, 0.100 and 0.250 seconds. Altitude testing in vernier mode explored igniter and thruster pulsing characteristics, long duration steady state operation (greater than 420 sec) and the impact of varying the percent fuel film cooling on vernier performance and chamber thermal response at low PC (4 psia). Data produced from the testing provided calibration of the performance and thermal models used in the design of the next version of the dual thrust Reaction Control Engine (RCE).

Author

Auxiliary Propulsion; Oxygen-Hydrocarbon Rocket Engines

20030071238 Ohio Aerospace Inst., Brook Park, OH, USA

Structural Seal Development

DeMange, Jeff; November 29, 2002; In English

Contract(s)/Grant(s): NCC3-879

Report No.(s): OAI-02-011; No Copyright; Avail: CASI; A03, Hardcopy

NASA currently has aggressive goals for the development of the next generations of Reusable Launch Vehicles (RLV). Many of the goals center around significantly reducing the time and cost of deployment (including payload) for these spacecraft, as well as substantially increasing the safety of the crew and hardware. A core component in the design of these next generation spacecraft is the propulsion system. Many concepts are currently being evaluated including Rocket Based Combined Cycle (RBCC) and Turbine Based Combined Cycle (TBCC) engine technologies, shown.

Author

Cost Reduction; Time Lag; Payloads; Rocket-Based Combined-Cycle Engines

20030068866 QSS Group, Inc., Cleveland, OH, USA

Oxidation-Reduction Resistance of Advanced Copper Alloys

Greenbauer-Seng, L., Technical Monitor; Thomas-Ogbuji, L.; Humphrey, D. L.; Setlock, J. A.; August 2003; In English; International Symposium on Corrosion Science in the 21st Century, 6-11 Jul. 2003, Manchester, UK

Contract(s)/Grant(s): NAS3-98008; WBS-22-721-26-06

Report No.(s): NASA/CR-2003-212549; E-14115; NAS 1.26:212549; No Copyright; Avail: CASI; A03, Hardcopy

Resistance to oxidation and blanching is a key issue for advanced copper alloys under development for NASA's next generation of reusable launch vehicles. Candidate alloys, including dispersion-strengthened Cu-Cr-Nb, solution-strengthened Cu-Ag-Zr, and ODS Cu-Al2O3, are being evaluated for oxidation resistance by static TGA exposures in low-p(O2) and cyclic oxidation in air, and by cyclic oxidation-reduction exposures (using air for oxidation and CO/CO2 or H2/Ar for reduction) to simulate expected service environments. The test protocol and results are presented.

Author

Copper Alloys; Oxidation-Reduction Reactions; Oxidation Resistance; Powder Metallurgy; Degradation

20030068243 Ohio Univ., Athens, OH, USA

Integrated G and C Implementation within IDOS: A Simulink Based Reusable Launch Vehicle Simulation

Fisher, Joseph E.; Bevacqua, Tim; Lawrence, Douglas A.; Zhu, J. Jim; Mahoney, Michael; January 15, 2003; In English

Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A03, Hardcopy

The implementation of multiple Integrated Guidance and Control (IG&C) algorithms per flight phase within a vehicle simulation poses a daunting task to coordinate algorithm interactions with the other G&C components and with vehicle subsystems. Currently being developed by Universal Space Lines LLC (USL) under contract from NASA, the Integrated

Development and Operations System (IDOS) contains a high fidelity Simulink vehicle simulation, which provides a means to test cutting edge G&C technologies. Combining the modularity of this vehicle simulation and Simulink s built-in primitive blocks provide a quick way to implement algorithms. To add discrete-event functionality to the unfinished IDOS simulation, Vehicle Event Manager (VEM) and Integrated Vehicle Health Monitoring (IVHM) subsystems were created to provide discrete-event and pseudo-health monitoring processing capabilities. Matlab's Stateflow is used to create the IVHM and Event Manager subsystems and to implement a supervisory logic controller referred to as the Auto-commander as part of the IG&C to coordinate the control system adaptation and reconfiguration and to select the control and guidance algorithms for a given flight phase. Manual creation of the Stateflow charts for all of these subsystems is a tedious and time-consuming process. The Stateflow Auto-builder was developed as a Matlab based software tool for the automatic generation of a Stateflow chart from information contained in a database. This paper describes the IG&C, VEM and IVHM implementations in IDOS. In addition, this paper describes the Stateflow Auto-builder.
Author
Algorithms; Controllers; Simulation

20030068239 Iowa State Univ. of Science and Technology, Ames, IA, USA
On-Board Entry Trajectory Planning Expanded to Sub-orbital Flight
Lu, Ping; Shen, Zuojun; January 15, 2003; In English; AIAA Guidance, Navigation, and Control Conference, 11-14 Aug. 2003, Austin, TX
Contract(s)/Grant(s): NAS8-01105; NAG8-1637; Copyright; Avail: CASI; A03, Hardcopy

A methodology for on-board planning of sub-orbital entry trajectories is developed. The algorithm is able to generate in a time frame consistent with on-board environment a three-degree-of-freedom (3DOF) feasible entry trajectory, given the boundary conditions and vehicle modeling. This trajectory is then tracked by feedback guidance laws which issue guidance commands. The current trajectory planning algorithm complements the recently developed method for on-board 3DOF entry trajectory generation for orbital missions, and provides full-envelope autonomous adaptive entry guidance capability. The algorithm is validated and verified by extensive high fidelity simulations using a sub-orbital reusable launch vehicle model and difficult mission scenarios including failures and aborts.
Author
Reusable Launch Vehicles; Trajectory Planning; Boundary Conditions; Algorithms

20030067947 Lockheed Martin Astronautics, Denver, CO, USA
Selection of Lockheed Martin's Preferred TSTO Configurations for the Space Launch Initiative
Hopkins, Joshua B.; [2002]; In English; 2002 World Space Congress (IAF/COSPAR/AIAA), 10-19 Oct. 2002, Houston, TX, USA
Contract(s)/Grant(s): NAS8-01098
Report No.(s): IAC-02-V.4.03; Copyright; Avail: CASI; A02, Hardcopy

Lockheed Martin is developing concepts for safe, affordable Two Stage to Orbit (TSTO) reusable launch vehicles as part of NASA s Space Launch Initiaiive. This paper discusses the options considered for the design of the TSTO, the impact of each of these options on the vehicle configuration, the criteria used for selection of preferred configurations, and the results of the selection process. More than twenty configurations were developed in detail in order to compare optioiis such as propellant choice, serial vs. parallel burn sequence, use of propellant crossfeed between stages, bimese or optimized stage designs, and high or low staging velocities. Each configuration was analyzed not only for performance and sizing, but also for cost and reliability. The study concluded that kerosene was the superior fuel for first stages, and that bimese vehicles were not attractive.
Author
Spacecraft Launching; Reusable Launch Vehicles; Spacecraft Configurations; Astronautics; Spacecraft Design; NASA Programs

20030067930 Rose-Hulman Inst. of Tech., Terre Haute, IN, USA
Genetic Algorithm Tuned Fuzzy Logic for Gliding Return Trajectories
Burchett, Bradley T.; [2003]; In English
Contract(s)/Grant(s): NAG8-1915; No Copyright; Avail: CASI; A03, Hardcopy

The problem of designing and flying a trajectory for successful recovery of a reusable launch vehicle is tackled using fuzzy logic control with genetic algorithm optimization. The plant is approximated by a simplified three degree of freedom

non-linear model. A baseline trajectory design and guidance algorithm consisting of several Mamdani type fuzzy controllers is tuned using a simple genetic algorithm. Preliminary results show that the performance of the overall system is shown to improve with genetic algorithm tuning.
Author
Trajectory Control; Reusable Launch Vehicles; Fuzzy Systems; Genetic Algorithms

20030067885 NASA Marshall Space Flight Center, Huntsville, AL, USA
NASA's Orbital Space Plane Risk Reduction Strategy
Dumbacher, Dan; [2003]; In English; AIAA/ICAS International Air and Space Symposium and Exposition: The Next 100 Years, 14-17 Jul. 2003, Dayton, OH, USA
Report No.(s): AIAA Paper 2003-3332; Copyright; Avail: CASI; A02, Hardcopy

This paper documents the transformation of NASA s Space Launch Initiative (SLI) Second Generation Reusable Launch Vehicle Program under the revised Integrated Space Transportation Plan, announced November 2002. Outlining the technology development approach followed by the original SLI, this paper gives insight into the current risk-reduction strategy that will enable confident development of the Nation s first orbital space plane (OSP). The OSP will perform an astronaut and contingency cargo transportation function, with an early crew rescue capability, thus enabling increased crew size and enhanced science operations aboard the International Space Station. The OSP design chosen for full-scale development will take advantage of the latest innovations American industry has to offer. The OSP Program identifies critical technologies that must be advanced to field a safe, reliable, affordable space transportation system for U.S. access to the Station and low-Earth orbit. OSP flight demonstrators will test crew safety features, validate autonomous operations, and mature thermal protection systems. Additional enabling technologies may be identified during the OSP design process as part of an overall risk-management strategy. The OSP Program uses a comprehensive and evolutionary systems acquisition approach, while applying appropriate lessons learned.
Author
Risk; NASA Space Programs; Aerospace Planes; Reusable Launch Vehicles; Autonomy; Orbital Maneuvering Vehicles

20030067819 NASA Marshall Space Flight Center, Huntsville, AL, USA
A Novel Approach for Modeling Chemical Reaction in Generalized Fluid System Simulation Program
Sozen, Mehmet; Majumdar, Alok; [2002]; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul. 2003, Huntsville, AL, USA; Copyright; Avail: CASI; A01, Hardcopy

The Generalized Fluid System Simulation Program (GFSSP) is a computer code developed at NASA Marshall Space Flight Center for analyzing steady state and transient flow rates, pressures, temperatures, and concentrations in a complex flow network. The code, which performs system level simulation, can handle compressible and incompressible flows as well as phase change and mixture thermodynamics. Thermodynamic and thermophysical property programs, GASP, WASP and GASPAK provide the necessary data for fluids such as helium, methane, neon, nitrogen, carbon monoxide, oxygen, argon, carbon dioxide, fluorine, hydrogen, water, a hydrogen, isobutane, butane, deuterium, ethane, ethylene, hydrogen sulfide, krypton, propane, xenon, several refrigerants, nitrogen trifluoride and ammonia. The program which was developed out of need for an easy to use system level simulation tool for complex flow networks, has been used for the following purposes to name a few: Space Shuttle Main Engine (SSME) High Pressure Oxidizer Turbopump Secondary Flow Circuits, Axial Thrust Balance of the Fastrac Engine Turbopump, Pressurized Propellant Feed System for the Propulsion Test Article at Stennis Space Center, X-34 Main Propulsion System, X-33 Reaction Control System and Thermal Protection System, and International Space Station Environmental Control and Life Support System design. There has been an increasing demand for implementing a combustion simulation capability into GFSSP in order to increase its system level simulation capability of a liquid rocket propulsion system starting from the propellant tanks up to the thruster nozzle for spacecraft as well as launch vehicles. The present work was undertaken for addressing this need. The chemical equilibrium equations derived from the second law of thermodynamics and the energy conservation equation derived from the first law of thermodynamics are solved simultaneously by a Newton-Raphson method. The numerical scheme was implemented as a User Subroutine in GFSSP.
Author
Computer Programs; Computerized Simulation; Chemical Reactions; Liquid Propellant Rocket Engines; Flow Visualization

20030067733 NASA Marshall Space Flight Center, Huntsville, AL, USA

Evaluation and Characterization Study of Dual Pulse Laser-Induced Spark (DPLIS) for Rocket Engine Ignition System Application

Osborne, Robin; Wehrmeyer, Joseph; Trinh, Huu; Early, James; July 07, 2003; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul. 2003, Huntsville, AL, USA; Original contains black and white illustrations; Copyright; Avail: CASI; A02, Hardcopy

This paper addresses the progress of technology development of a laser ignition system at NASA Marshall Space Flight Center (MSFC). Laser ignition has been used at MSFC in recent test series to successfully ignite RP1/GOX propellants in a subscale rocket chamber, and other past studies by NASA GRC have demonstrated the use of laser ignition for rocket engines. Despite the progress made in the study of this ignition method, the logistics of depositing laser sparks inside a rocket chamber have prohibited its use. However, recent advances in laser designs, the use of fiber optics, and studies of multi-pulse laser formats3 have renewed the interest of rocket designers in this state-of the-art technology which offers the potential elimination of torch igniter systems and their associated mechanical parts, as well as toxic hypergolic ignition systems. In support of this interest to develop an alternative ignition system that meets the risk-reduction demands of Next Generation Launch Technology (NGLT), characterization studies of a dual pulse laser format for laser-induced spark ignition are underway at MSFC. Results obtained at MSFC indicate that a dual pulse format can produce plasmas that absorb the laser energy as efficiently as a single pulse format, yet provide a longer plasma lifetime. In an experiments with lean H2/air propellants, the dual pulse laser format, containing the same total energy of a single laser pulse, produced a spark that was superior in its ability to provide sustained ignition of fuel-lean H2/air propellants. The results from these experiments are being used to optimize a dual pulse laser format for future subscale rocket chamber tests. Besides the ignition enhancement, the dual pulse technique provides a practical way to distribute and deliver laser light to the combustion chamber, an important consideration given the limitation of peak power that can be delivered through optical fibers. With this knowledge, scientists and engineers at Los Alamos National Laboratory and CFD Research Corporation have designed and fabricated a miniaturized, first-generation optical prototype of a laser ignition system that could be the basis for a laser ignition system for rocket applications. This prototype will be tested at MSFC in future subscale rocket ignition tests.

Author

Computational Fluid Dynamics; Ignition Systems; Pulsed Lasers; Rocket Engines; Laser Outputs; Spark Ignition; Fabrication

20030067586 NASA Marshall Space Flight Center, Huntsville, AL, USA

X-33 LH2 Tank Failure Investigation Findings

Niedermeyer, Melinda; [2003]; In English; International Conference on Composites Engineering, 20-25 Jul. 2003, New Orleans, LA, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation provides information on the composite sandwich-honeycomb structure of the liquid hydrogen tank of the X-33 reusable launch vehicle, and describes why the the first pressure test to determine the tank's structural integrity failed. The presentation includes images of the tank before and after the failed test, including photomicrographs. It then reaches conclusions on the nature of the microcracks which caused the liquid hydrogen leakage.

CASI

X-33 Reusable Launch Vehicle; Structural Failure; Liquid Hydrogen; Microcracks; Leakage

20030067582 Northrop Grumman Corp., Redondo, CA, USA, NASA Marshall Space Flight Center, Huntsville, AL, USA

Design and Testing of Non-Toxic RCS Thrusters for Second Generation Reusable Launch Vehicle

Calvignac, Jacky; Dang, Lisa; Tramel, Terri; Paseur, Lila; Champion, Robert, Technical Monitor; July 22, 2003; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul. 2003, Huntsville, AL, USA

Contract(s)/Grant(s): NAS8-01110; NAG8-0110

Report No.(s): AIAA Paper 2003-4922; Copyright; Avail: CASI; A03, Hardcopy

Under NASA sponsorship, Northrop Grumman Space Technology (NGST) designed, built and tested two non-toxic, reaction control engines, one using liquid oxygen (LOX) and liquid hydrogen (LH2) and the other using liquid oxygen and ethanol. This paper presents the design and testing of the LOX/LH2 thruster. The two key enabling technologies are the coaxial liquid-on-liquid pintle injector and the fuelcooling duct. The workhorse thruster was hotfire tested at the NASA Marshall Space Flight Center Test Stand 500 in March and April of 2002. All tests were performed at sea-level conditions. During the test program, 7 configurations were tested, including 2 combustion chambers, 3 LOX injector pintle tips, and 4 LHp injector settings. The operating conditions surveyed were 70 to 100% thrust levels, mixture ratios from 3.27 to 4.29, and LH2 duct cooling from 18.0 to 25.5% fuel flow. The copper heat sink chamber was used for 16 burns, each burn lasting from 0.4 to 10 seconds, totaling 51.4 seconds, followed by Haynes chamber testing ranging from 0.9 to 120 seconds, totaling 300.9 seconds.

The performance of the engine reached 95% C* efficiency. The temperature on the Haynes chamber remained well below established material limits, with the exception of one localized hot spot. These results demonstrate that both the coaxial liquid-on-liquid pintle injector design and fuel duct concepts are viable for the intended application. The thruster headend design maintained cryogenic injection temperatures while firing, which validates the selected injector design approach for minimal heat soak-back. Also, off -nominal operation without adversely impacting the thermal response of the engine showed the robustness of the duct design, a key design feature for this application. By injecting fuel into the duct, the throat temperatures are manageable, yet the split of fuel through the cooling duct does not compromise the overall combstion efficiency, which indicates that, provided proper design refinement, such a concept could be applied to a high-performance version of the thruster.
Author
Aerospace Engineering; Reusable Launch Vehicles; Engine Tests; Combustion Chambers; Combustion Efficiency; Spacecraft Performance; Thrust

20030067336 NASA Dryden Flight Research Center, Edwards, CA, USA
Advanced Range Safety System for High Energy Vehicles
Claxton, Jeffrey S.; Linton, Donald F.; September 2002; In English; Eleventh International Conference on Space Planes and Hypersonic Systems and Technologies, 29 Sep. - 4 Oct. 2002, Orleans, France; Original contains black and white illustrations
Contract(s)/Grant(s): WU 710-35-74-00-FF-00-RSL
Report No.(s): H-2508; No Copyright; Avail: CASI; A02, Hardcopy

The advanced range safety system project is a collaboration between the National Aeronautics and Space Administration and the USA Air Force to develop systems that would reduce costs and schedule for safety approval for new classes of unmanned high-energy vehicles. The mission-planning feature for this system would yield flight profiles that satisfy the mission requirements for the user while providing an increased quality of risk assessment, enhancing public safety. By improving the speed and accuracy of predicting risks to the public, mission planners would be able to expand flight envelopes significantly. Once in place, this system is expected to offer the flexibility of handling real-time risk management for the high-energy capabilities of hypersonic vehicles including autonomous return-from-orbit vehicles and extended flight profiles over land. Users of this system would include mission planners of Space Launch Initiative vehicles, space planes, and other high-energy vehicles. The real-time features of the system could make extended flight of a malfunctioning vehicle possible, in lieu of an immediate terminate decision. With this improved capability, the user would have more time for anomaly resolution and potential recovery of a malfunctioning vehicle.
Author
Hypersonic Vehicles; Launch Vehicles; Mission Planning; Range Safety; Aerospace Planes

20030066934 NASA Dryden Flight Research Center, Edwards, CA, USA
Flight Demonstration of X-33 Vehicle Health Management System Components on the F/A-18 Systems Research Aircraft
Schweikhard, Keith A.; Richards, W. Lance; Theisen, John; Mouyos, William; Garbos, Raymond; December 2001; In English; 19th Digital Avionics Systems Conference, 7-11 Oct. 2000, Philadelphia, PA, USA; Original contains black and white illustrations
Contract(s)/Grant(s): WU 529-35-34-E8-RR
Report No.(s): NASA/TM-2001-209037; H-2435; NAS 1.15:209037; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 reusable launch vehicle demonstrator has identified the need to implement a vehicle health monitoring system that can acquire data that monitors system health and performance. Sanders, a Lockheed Martin Company, has designed and developed a COTS-based open architecture system that implements a number of technologies that have not been previously used in a flight environment. NASA Dryden Flight Research Center and Sanders teamed to demonstrate that the distributed remote health nodes, fiber optic distributed strain sensor, and fiber distributed data interface communications components of the X-33 vehicle health management (VHM) system could be successfully integrated and flown on a NASA F-18 aircraft. This paper briefly describes components of X-33 VHM architecture flown at Dryden and summarizes the integration and flight demonstration of these X-33 VHM components. Finally, it presents early results from the integration and flight efforts.
Author
X-33 Reusable Launch Vehicle; Systems Health Monitoring; Management Systems; Systems Integration; Flight Tests

20030066328 Lockheed Martin Michoud Space Systems, New Orleans, LA, USA
Composite Development and Applications for RLV Tankage
Wright, Richard J.; Achary, David C.; McBain, Michael C.; [2003]; In English; AIAA/ICAS International Air and Space Symposium and Exposition, 14-17 Jul. 2003, Dayton, OH, USA
Contract(s)/Grant(s): NCC8-191; No Copyright; Avail: CASI; A03, Hardcopy

The development of polymer composite cryogenic tanks is a critical step in creating the next generation of launch vehicles. Future launch vehicles need to minimize the gross liftoff weight (GLOW), which is possible due to the 28%-41% reduction in weight that composite materials can provide over current aluminum technology. The development of composite cryogenic tanks, feedlines, and unpressurized structures are key enabling technologies for performance and cost enhancements for Reusable Launch Vehicles (RLVs). The technology development of composite tanks has provided direct and applicable data for feedlines, unpressurized structures, material compatibility, and cryogenic fluid containment for highly loaded complex structures and interfaces. All three types of structure have similar material systems, processing parameters, scaling issues, analysis methodologies, NDE development, damage tolerance, and repair scenarios. Composite cryogenic tankage is the most complex of the 3 areas and provides the largest breakthrough in technology. A building block approach has been employed to bring this family of difficult technologies to maturity. This approach has built up composite materials, processes, design, analysis and test methods technology through a series of composite test programs beginning with the NASP program to meet aggressive performance goals for reusable launch vehicles. In this paper, the development and application of advanced composites for RLV use is described.
Author

Reusable Launch Vehicles; Composite Materials; Tanks (Containers); Cryogenics

20030066309 NASA Marshall Space Flight Center, Huntsville, AL, USA
Flight Demonstrations of Orbital Space Plane (OSP) Technologies
Turner, Susan; [2003]; In English; AIAA/ICAS International Air and Space Symposium, 14-18 Jul. 2003, Dayton, OH, USA
Report No.(s): AIAA Paper 2003-2710; No Copyright; Avail: CASI; A02, Hardcopy

The Orbital Space Plane (OSP) Program embodies NASA s priority to transport Space Station crews safely, reliably, and affordably, while it empowers the Nation s greater strategies for scientific exploration and space leadership. As early in the development cycle as possible, the OSP will provide crew rescue capability, offering an emergency ride home from the Space Station, while accommodating astronauts who are deconditioned due to long- duration missions, or those that may be ill or injured. As the OSP Program develops a fully integrated system, it will use existing technologies and employ computer modeling and simulation. Select flight demonstrator projects will provide valuable data on launch, orbital, reentry, and landing conditions to validate thermal protection systems, autonomous operations, and other advancements, especially those related to crew safety and survival.
Author

Aerospace Vehicles; X-37 Vehicle; Space Transportation System

20030066222 NASA Marshall Space Flight Center, Huntsville, AL, USA
Design and Analysis of Turbines for Space Applications
Griffin, Lisa W.; Dorney, Daniel J.; Huber, Frank W.; June 13, 2003; In English; 33rd AIAA Fluid Dynamics Conference, 23-26 Jun. 2003, Orlando, FL, USA; Copyright; Avail: CASI; A03, Hardcopy

In order to mitigate the risk of rocket propulsion development, efficient, accurate, detailed fluid dynamics analysis of the turbomachinery is necessary. This analysis is used for component development, design parametrics, performance prediction, and environment definition. To support this requirement, a task was developed at NASAh4arshall Space Flight Center (MSFC) to improve turbine aerodynamic performance through the application of advanced design and analysis tools. There are four major objectives of this task: 1) to develop, enhance, and integrate advanced turbine aerodynamic design and analysis tools; 2) to develop the methodology for application of the analytical techniques; 3) to demonstrate the benefits of the advanced turbine design procedure through its application to a relevant turbine design point; and 4) to verify the optimized design and analysis with testing. The turbine chosen on which to demonstrate the procedure was a supersonic design suitable for a reusable launch vehicle (RLV). The hot gas path and blading were redesigned to obtain an increased efficiency. The redesign of the turbine was conducted with a consideration of system requirements, realizing that a highly efficient turbine that, for example, significantly increases engine weight, is of limited benefit. Both preliminary and detailed designs were considered.

To generate an improved design, one-dimensional (1D) design and analysis tools, computational fluid dynamics (CFD), response surface methodology (RSM), and neural nets (NN) were used.
Author
Turbomachinery; Design Analysis; Computational Fluid Dynamics; Performance Prediction; Dynamic Response; Aerodynamic Characteristics

20030066115 NASA Marshall Space Flight Center, Huntsville, AL, USA
Earth-to-Orbit Rocket Propulsion
Beaurain, Andre; Souchier, Alain; Moravie, Michel; Sackheim, Robert L.; Cikanek, Harry A., III; March 31, 2003; In English; International Air and Space Symposium and Exposition: The Next 100 Years, 14-18 Jul. 2003, Dayton, OH, USA; Copyright; Avail: CASI; A03, Hardcopy

The Earth-to-orbit (ETO) phase of access to space is and always will be the first and most critical phase of all space missions. This first phase of all space missions has unique characteristics that have driven space launcher propulsion requirements for more than half a century. For example, the need to overcome the force of the Earth s gravity in combination with high levels of atmospheric drag to achieve the initial orbital velocity; i.e., Earth parking orbit or =9 km/s, will always require high thrust- to-weight (TN) propulsion systems. These are necessary with a T/W ratio greater than one during the ascent phase. The only type of propulsion system that can achieve these high T/W ratios are those that convert thermal energy to kinetic energy. There are only two basic sources of onboard thermal energy: chemical combustion-based systems or nuclear thermal-based systems (fission, fusion, or antimatter). The likelihood of advanced open-cycle, nuclear thermal propulsion being developed for flight readiness or becoming environmentally acceptable during the next century is extremely low. This realization establishes that chemical propulsion for ET0 launchers will be the technology of choice for at least the next century, just as it has been for the last half century of rocket flight into space. The world s space transportation propulsion requirements have evolved through several phases over the history of the space program, as has been necessitated by missions and systems development, technological capabilities available, and the growth and evolution of the utilization of space for economic, security, and science benefit. Current projections for the continuing evolution of requirements and concepts may show how future space transportation system needs could be addressed. The evolution and projections will be described in detail in this manuscript.
Author
Earth Orbits; Reusable Rocket Engines; Reusable Spacecraft; Reusable Launch Vehicles

20030066112 NASA Marshall Space Flight Center, Huntsville, AL, USA
Multi-Disciplinary Analysis for Future Launch Systems Using NASA's Advanced Engineering Environment (AEE)
Monell, Donald; Mathias, Donovan; Reuther, James; Garn, Michelle; [2003]; In English; 16th AIAA Computational Fluid Dynamics Conference, 23-26 Jun. 2003, Orlando, FL, USA
Report No.(s): AIAA Paper 2003-3428; No Copyright; Avail: CASI; A03, Hardcopy

A new engineering environment constructed for the purposes of analyzing and designing Reusable Launch Vehicles (RLVs) is presented. The new environment has been developed to allow NASA to perform independent analysis and design of emerging RLV architectures and technologies. The new Advanced Engineering Environment (AEE) is both collaborative and distributed. It facilitates integration of the analyses by both vehicle performance disciplines and life-cycle disciplines. Current performance disciplines supported include: weights and sizing, aerodynamics, trajectories, propulsion, structural loads, and CAD-based geometries. Current life-cycle disciplines supported include: DDT&E cost, production costs, operations costs, flight rates, safety and reliability, and system economics. Involving six NASA centers (ARC, LaRC, MSFC, KSC, GRC and JSC), AEE has been tailored to serve as a web-accessed agency-wide source for all of NASA's future launch vehicle systems engineering functions. Thus, it is configured to facilitate (a) data management, (b) automated tool/process integration and execution, and (c) data visualization and presentation. The core components of the integrated framework are a customized PTC Windchill product data management server, a set of RLV analysis and design tools integrated using Phoenix Integration's Model Center, and an XML-based data capture and transfer protocol. The AEE system has seen production use during the Initial Architecture and Technology Review for the NASA 2nd Generation RLV program, and it continues to undergo development and enhancements in support of its current main customer, the NASA Next Generation Launch Technology (NGLT) program.
Author
Reusable Launch Vehicles; Design Analysis; Systems Engineering; Systems Analysis

20030066011 NASA Glenn Research Center, Cleveland, OH, USA

Tantalum Addition to Zirconium Diboride for Improved Oxidation Resistance

Levine, Stanley R.; Opila, Eliizabeth J.; July 2003; In English; 27th Annual Conference on Composites, Materials and Structures, 27-31 Jan. 2003, Cape Canaveral, FL, USA; Original contains color and black and white illustrations

Contract(s)/Grant(s): WBS 22-706-85-08

Report No.(s): NASA/TM-2003-212483; E-14025; NAS 1.15:212483; Copyright; Avail: CASI; A03, Hardcopy

Ultrahigh temperature ceramics have performed unreliably due to material flaws and attachment design. These deficiencies are brought to the fore by the low fracture toughness and thermal shock resistance of UHTCs. If these deficiencies are overcome, we are still faced with poor oxidation resistance as a limitation on UHTC applicability to reusable launch vehicles. We have been addressing the deficiencies of UHTCs with our focus on composite constructions and functional grading to address the mechanical issues, and on composition modification to address the oxidation issue. The approaches and progress toward the latter are reported.

Author

Ceramics; Refractory Materials; Tantalum; Zirconium; Borides; Oxidation Resistance; Mechanical Properties

20030065957 NASA Marshall Space Flight Center, Huntsville, AL, USA

Design and Testing of Non-Toxic RCS Thrusters for Second Generation Reusable Launch Vehicle

Calvignac, Jacky; Tramel, Terri; [2003]; In English; Joint Propulsion Conference, Jul. 2003, Huntsville, AL, USA

Contract(s)/Grant(s): NAS8-01110; No Copyright; Avail: Other Sources; Abstract Only

The current NASA Space Shuttle auxiliary propulsion system utilizes nitrogen tetroxide (NTO) and monomethylhydrazine (MMH), hypergolic propellants. This use of these propellants has resulted in high levels of maintenance and precautions that contribute to costly launch operations. By employing alternate propellant combinations, those less toxic to humans, the hazards and time required between missions can be significantly reduced. Use of alternate propellants can thereby increase the efficiency and lower the cost in launch operations. In support of NASA's Space Launch Initiative (SLI), TRW proposed a three-phase project structured to significantly increase the technology readiness of a high-performance reaction control subsystem (RCS) thruster using non-toxic propellant for an operationally efficient and reusable auxiliary propulsion system (APS). The project enables the development of an integrated primary/vernier thruster capable of providing dual-thrust levels of both 1000-lbf class thrust and 25-lbf thrust. The intent of the project is to reduce the risk associated with the development of an improved RCS flight design that meets the primary NASA objectives of improved safety and reliability while reducing systems operations and maintenance costs. TRW proposed two non-toxic auxiliary propulsion engine designs, one using liquid oxygen and liquid hydrogen and the other using liquid oxygen and liquid ethanol, as candidates to meet the goals of reliability and affordability at the RCS level. Both of these propellant combinations offer the advantage of a safe environment for maintenance, while at the same time providing adequate to excellent performance for a conventional liquid propulsion systems. The key enabling technology incorporated in both TRW thrusters is the coaxial liquid on liquid pintle injector. This paper will concentrate on only the design and testing of one of the thrusters, the liquid oxygen (LOX) and liquid hydrogen (LH2) thruster. The LOX/LH2 thruster design includes a LOX-centered pintle injector, consisting of two rows of slots that create a radial spoke spray pattern in the combustion chamber. The main fuel injector creates a continuous sheet of LH2 originating upstream of the LOX pintle injector. The two propellants impinge at the pintle slots, where the resulting momentum ratio and spray pattern determines the combustion efficiency and thermal effects on the hardware. Another enabling technology used in the design of this thruster is fuel film cooling through a duct, lining the inner wall of the combustion chamber barrel section. The duct is also acts as a secondary fuel injection point. The variation in the amount of LH2 used for the duct allows for adjustments in the cooling capacity for the thruster. The Non-Toxic LOX-LH2 RCS Workhorse Thruster was tested at the NASA Marshall Space Flight Center's Test Stand 500. Hot-fire tests were conducted between March 08, 2002 and April 05, 2002. All testing during the program base period were performed at sea-level conditions. During the test program, 7 configurations were tested, including 2 combustion chambers, 3 LOX injector pintle tips, and 4 LH2 injector stroke settings. The operating conditions that were surveyed varied thrust levels, mixture ratio and LH2 duct cooling flow. The copper heat sink chamber was used for 16 burns, each burn lasting from 0.4 to 10 seconds, totaling 51.4 seconds, followed by Haynes chamber testing ranging from 0.9 to 120 seconds, totaling 300.9 seconds. The total accumulated burn time for the test program is 352.3 seconds. C* efficiency was calculated and found to be within expectable limits for most operating conditions. The temperature on the Haynes combustion chamber remained below established material limits, with the exception of one localized hot spot. The test results demonstrate that both the coaxial liquid-on-liquid pintle injector design and fuel duct concepts are viable for the intended application. The thruster head-e design maintained cryogenic injection temperatures while firing, which validates the concept for minimal heat soak back. By injecting fuel into the duct, the throat temperatures were manageable, yet the split of fuel through the cooling duct does not compromise the overall combustion efficiency, which

indicates that, provided proper design refinement, such a concept can be applied to a high-performance version of the thruster. These hot fire tests demonstrate the robustness of the duct design concept and good capability to withstand off-nominal operating conditions without adversely impacting the thermal response of the engine, a key design feature for a cryogenic thruster.
Author
Liquid Rocket Propellants; Thrustors; Ethyl Alcohol; Engine Tests; Aerospace Safety; Hydrogen Fuels; Liquid Hydrogen; Liquid Oxygen

20030065923 NASA Marshall Space Flight Center, Huntsville, AL, USA, NASA Johnson Space Center, Houston, TX, USA
NASA's New Orbital Space Plane: A Bridge to the Future
Davis, Stephan R.; Engler, Leah M.; Fisher, Mark F.; Dumbacher, Dan L.; Boswell, Barry E.; [2003]; In English; AIAA 10th Anniverary of Flight, 14-17 Jul. 2003, Dayton, OH, USA; No Copyright; Avail: CASI; A03, Hardcopy

NASA is developing a new spacecraft system called the Orbital Space Plane (OSP). The OSP will be launched on an expendable launch vehicle and serve to augment the shuttle in support of the International Space Station by transporting astronauts to and from the International Space Station and by providing a crew rescue system.
Author
Aerospace Planes; Development; Project Management

20030065841 NASA Marshall Space Flight Center, Huntsville, AL, USA
Rocket Engine Health Management: Early Definition of Critical Flight Measurements
Christenson, Rick L.; Nelson, Michael A.; Butas, John P.; July 2003; In English; 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 20-23 Jul. 2003, Huntsville, AL, USA; Original contains black and white illustrations
Report No.(s): AIAA Paper 2003-5251; No Copyright; Avail: CASI; A02, Hardcopy

The NASA led Space Launch Initiative (SLI) program has established key requirements related to safety, reliability, launch availability and operations cost to be met by the next generation of reusable launch vehicles. Key to meeting these requirements will be an integrated vehicle health management (M) system that includes sensors, harnesses, software, memory, and processors. Such a system must be integrated across all the vehicle subsystems and meet component, subsystem, and system requirements relative to fault detection, fault isolation, and false alarm rate. The purpose of this activity is to evolve techniques for defining critical flight engine system measurements-early within the definition of an engine health management system (EHMS). Two approaches, performance-based and failure mode-based, are integrated to provide a proposed set of measurements to be collected. This integrated approach is applied to MSFC s MC-1 engine. Early identification of measurements supports early identification of candidate sensor systems whose design and impacts to the engine components must be considered in engine design.
Author
Reusable Launch Vehicles; Rocket Engines; Failure Modes; Fault Detection; False Alarms; Reliability

20030065656 NASA Marshall Space Flight Center, Huntsville, AL, USA
Analysis of Parallel Burn Without Crossfeed TSTO RLV Architectures and Comparison to Parallel Burn With Crossfeed and Series Burn Architectures
Smith, Garrett; Phillips, Alan; November 4, 2002; In English; 39th Joint Propulsion Conference, 20-23 Jul. 2003, Huntsville, AL, USA
Contract(s)/Grant(s): NAS8-02057; Copyright; Avail: Other Sources; Abstract Only

There are currently three dominant TSTO class architectures. These are Series Burn (SB), Parallel Burn with crossfeed (PBw/cf), and Parallel Burn without crossfeed (PBncf). The goal of this study was to determine what factors uniquely affect PBncf architectures, how each of these factors interact, and to determine from a performance perspective whether a PBncf vehicle could be competitive with a PBw/cf or SB vehicle using equivalent technology and assumptions. In all cases, performance was evaluated on a relative basis for a fixed payload and mission by comparing gross and dry vehicle masses of a closed vehicle. Propellant combinations studied were LOX: LH2 propelled orbiter and booster (HH) and LOX: Kerosene booster with LOX: LH2 orbiter (KH). The study conclusions were: 1) a PBncf orbiter should be throttled as deeply as possible after launch until the staging point. 2) a detailed structural model is essential to accurate architecture analysis and evaluation. 3) a PBncf TSTO architecture is feasible for systems that stage at mach 7. 3a) HH architectures can achieve a mass growth relative to PBw/cf of < 20%. 3b) KH architectures can achieve a mass growth relative to Series Burn of < 20%. 4) center of gravity (CG) control will be a major issue for a PBncf vehicle, due to the low orbiter specific thrust to weight ratio and to the

position of the orbiter required to align the nozzle heights at liftoff. 5) thrust to weight ratios of 1.3 at liftoff and between 1.0 and 0.9 when staging at mach 7 appear to be close to ideal for PBncf vehicles. 6) performance for all vehicles studied is better when staged at mach 7 instead of mach 5. The study showed that a Series Burn architecture has the lowest gross mass for HH cases, and has the lowest dry mass for KH cases. The potential disadvantages of SB are the required use of an air-start for the orbiter engines and potential CG control issues. A Parallel Burn with crossfeed architecture solves both these problems, but the mechanics of a large bipropellant crossfeed system pose significant technical difficulties. Parallel Burn without crossfeed vehicles start both booster and orbiter engines on the ground and thus avoid both the risk of orbiter air-start and the complexity of a crossfeed system. The drawback is that the orbiter must use 20% to 35% of its propellant before reaching the staging point. This induces a weight penalty in the orbiter in order to carry additional propellant, which causes a further weight penalty in the booster to achieve the same staging point. One way to reduce the orbiter propellant consumption during the first stage is to throttle down the orbiter engines as much as possible. Another possibility is to use smaller or fewer engines. Throttling the orbiter engines soon after liftoff minimizes CG control problems due to a low orbiter liftoff thrust, but may result in an unnecessarily high orbiter thrust after staging. Reducing the number or size of engines size may cause CG control problems and drift at launch. The study suggested possible methods to maximize performance of PBncf vehicle architectures in order to meet mission design requirements.
Author
Combustion; Propulsion System Configurations; Propulsion System Performance; Reusable Launch Vehicles

20030065617 Lockheed Martin Corp., Denver, CO, USA
Lockheed Martin Response to the OSP Challenge
Sullivan, Robert T.; Munkres, Randy; Megna, Thomas D.; Beckham, Joanne; June 25, 2003; In English; International Air and Space Symposium and Exposition, 14-17 Jul. 2003, Dayton, OH, USA
Contract(s)/Grant(s): NAS8-01098
Report No.(s): AIAA Paper 2003-2708; No Copyright; Avail: CASI; A02, Hardcopy

The Lockheed Martin Orbital Space Plane System provides crew transfer and rescue for the International Space Station more safely and affordably than current human space transportation systems. Through planned upgrades and spiral development, it is also capable of satisfying the Nation’s evolving space transportation requirements and enabling the national vision for human space flight. The OSP System, formulated through rigorous requirements definition and decomposition, consists of spacecraft and launch vehicle flight elements, ground processing facilities and existing transportation, launch complex, range, mission control, weather, navigation, communication and tracking infrastructure. The concept of operations, including procurement, mission planning, launch preparation, launch and mission operations and vehicle maintenance, repair and turnaround, is structured to maximize flexibility and mission availability and minimize program life cycle cost. The approach to human rating and crew safety utilizes simplicity, performance margin, redundancy, abort modes and escape modes to mitigate credible hazards that cannot be designed out of the system.
Author
Aerospace Planes; Launch Vehicles; Ground Support Systems; Spacecraft Launching; Navigation; Spacecraft Docking

20030064145 NASA Marshall Space Flight Center, Huntsville, AL, USA
X-37 C-SiC CMC Control Surface Components Development: Status of the NASA/Boeing/USAF Orbital Vehicle and Related Efforts
Valentine, Peter G.; Rivers, H. Kevin; Chen, Victor L.; 2003; In English; 27th Annual Conference on Composites, Materials, and Structures, 27-30 Jan. 2003, Cape Canaveral, FL, USA; Copyright; Avail: Other Sources; Abstract Only

Carbon/Silicon-Carbide (C-SiC) ceramic matrix composite (CMC) flaperon and ruddervator control surface components are being developed for the X-37 Orbital Vehicle (OV). The results of the prior NASA LaRC led work, aimed at developing C-SiC flaperon and ruddervator components for the X-37, will be reviewed. The status of several on-going and/or planned NASA, USAF, and Boeing programs that will support the development of control surface components for the X-37 OV will also be reviewed. The overall design and development philosophy being employed to assemble a team(s) to develop both (a) C-SiC hot structure control surface components for the X-37 OV, and (b) carbon-carbon (C-C) hot structure components (a risk-reduction backup option for the OV), will be presented.
Author
Ceramic Matrix Composites; X-37 Vehicle; Silicon Carbides; Control Surfaces

20030064094 NASA Marshall Space Flight Center, Huntsville, AL, USA
Permeability After Impact Testing of Composite Laminates
Nettles, Alan T.; [2003]; In English; SAMPE 2003 Symposium and Exhibition, 11-15 May 2003, Long Beach, CA, USA; No Copyright; Avail: CASI; A03, Hardcopy

Since composite laminates are beginning to be identified for use in reusable launch vehicle propulsion systems, an understanding of their permeance is needed. A foreign object impact event can cause a localized area of permeability (leakage) in a polymer matrix composite and it is the aim of this study to assess a method of quantifying permeability-after-impact results. A simple test apparatus is presented and variables that could affect the measured values of permeability-after-impact were assessed. Once it was determined that valid numbers were being measured, a fiber/resin system was impacted at various impact levels and the resulting permeability measured, first with a leak check solution (qualitative) then using the new apparatus (quantitative). The results showed that as the impact level increased, so did the measured leakage. As the pressure to the specimen was increased, the leak rate was seen to increase in a non-linear fashion for almost all of the specimens tested.
Author
Laminates; Permeability; Impact Tests; Leakage; Polymer Matrix Composites

20030064052 NASA Marshall Space Flight Center, Huntsville, AL, USA
Ceramic Matrix Composite Cooled Nozzle Material Development Program
Lawrence, Tim; Eckel, Andy; Porter, John; Pichon, T.; Patterson, B.; Paquette, T.; [2002]; In English; JANNAF 13th NDES Meeting, 27-29 Aug. 2002, San Antonio, TX, USA
Contract(s)/Grant(s): NAS8-00141; NAS8-99112; NAS8-00140; NAS8-00124; No Copyright; Avail: Other Sources; Abstract Only

The X-33 program initiated a risk reduction technology project to develop an actively cooled cermic matrix composite (CMC) nozzle ramp for the linear aerospike engine. The objective was to reduce the weight and increase the operating temperature capabilities of the nozzle ramp. A complement to this original project was subsequently supported by NASA's Second Generation Reusable Launch Vehicle Program to develop a high risk high payoff cooled composite nozzle ramp. This project focused on less mature technologies and concepts having the potential to achieve a significant weight reduction beyond those systems which were being considered in the original X-33 project. The aerospike engine was not selected under the initial Space Launch Initiative (SLI) Program research announcement. However, in recognition of the tremendous application opportunities of such technology to other areas and systems in the rocket industry, the effort was continued and is in the process of being transferred to the 3rd Generation Reusable Launch Vehicle (RLV) Program where it will be combined with the current cooled CMC panel project. The objective of the refocused project is to advance the material, design, and analysis work on the ramp concepts to a point that would allow the selection of the most promising candidate(s) for continued development. The concept(s) carried on in the 3rd Generation RLV Program will be modified to address the goals of this program. Originally, four contracts with different design concepts were initiated. Each contractor has performed design and analysis of their concept and submitted a subscale component for testing in the Cell 22 test rig at Glenn Research Center. This paper will discuss the results to date of each design concept and the potential applications to future rocket nozzle systems. The engineering technology challenges for each concept were determined and addressed during this phase of the effort. These challenges and the success in addressing them will also be discussed. High temperature, noneroding nozzle materials have the potential to significantly improve nozzle performance. The work performed under this effort is at the forefront of CMC material development and will give the rocket nozzle community a good view into the status of cooled CMC materials.
Author
Ceramic Matrix Composites; X-33 Reusable Launch Vehicle; Rocket Nozzles; Ramps (Structures)

20030064017 Boeing Phantom Works, Huntington Beach, CA, USA
Boeing 2nd Generation Reusable Launch Vehicle Architecture
Bienhoff, Dallas; [2003]; In English; Space Technology and Applications International Forum (STAIF 2003), 2-5 Feb. 2003, Albuquerque, NM, USA
Contract(s)/Grant(s): NAS8-01099; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents viewgraphs on the Space Launch Initiative (SLI) space shuttle safety upgrades.
CASI
Reusable Launch Vehicles; Spacecraft Design; Space Shuttles; Space Transportation System

20030063938 NASA Marshall Space Flight Center, Huntsville, AL, USA
Selection and Evaluation of An Alloy for Nozzle Application
Pandey, Awadh; Shah, Sandeep; Shadoan, Mike; May 22, 2003; In English; Aeromat Conference 2003, 9-12 Jun. 2003, Dayton, OH, USA; Copyright; Avail: CASI; A03, Hardcopy

The goal was to select and characterize a material with superior weldability and hydrogen resistance for nozzle application under Second Generation Reusable Launch Vehicle (2GRLV) program funded by NASA_MSFC. Currently standard A-286 is used for SSME nozzle.Howewer, material with superior weldability and hydrogen resistance will be desired for (2GRLV).
Derived from text
Product Development; Alloys; Hydrogen; Weldability

20030063899 Lockheed Martin Engineering and Sciences Co., Hampton, VA, USA
Characterization of Thin Film Polymers Through Dynamic Mechanical Analysis and Permeation
Herring, Helen; June 2003; In English
Contract(s)/Grant(s): NAS1-00135; 762-30-51-03
Report No.(s): NASA/CR-2003-212422; NAS 1.26:212422; No Copyright; Avail: CASI; A03, Hardcopy

Thin polymer films are being considered, as candidate materials to augment the permeation resistance of cryogenic hydrogen fuel tanks such as would be required for future reusable launch vehicles. To evaluate performance of candidate films after environmental exposure, an experimental study was performed to measure the thermal/mechanical and permeation performance of six, commercial-grade materials. Dynamic storage modulus, as measured by Dynamic Mechanical Analysis, was found over a range of temperatures. Permeability, as measured by helium gas diffusion, was found at room temperature. Test data was correlated with respect to film type and pre-test exposure to moisture, elevated temperature, and cryogenic temperature. Results indicated that the six films were comparable in performance and their resistance to environmental degradation.
Author
Thin Films; Permeability; Polymeric Films; Cryogenic Fluid Storage; Hydrogen Fuels

20030063104 Clark-Atlanta Univ., GA, USA
Characterization of Polyimide Foams for Ultra-Lightweight Space Structures
Meador, Michael, Technical Monitor; Hillman, Keithan; Veazie, David R.; HBCUs/OMUs Research Conference Agenda and Abstracts; February 2003; In English; No Copyright; Abstract Only; Available from CASI only as part of the entire parent document

Ultra-lightweight materials have played a significant role in nearly every area of human activity ranging from magnetic tapes and artificial organs to atmospheric balloons and space inflatables. The application range of ultra-lightweight materials in past decades has expanded dramatically due to their unsurpassed efficiency in terms of low weight and high compliance properties. A new generation of ultra-lightweight materials involving advanced polymeric materials, such as TEEK (TM) polyimide foams, is beginning to emerge to produce novel performance from ultra-lightweight systems for space applications. As a result, they require that special conditions be fulfilled to ensure adequate structural performance, shape retention, and thermal stability. It is therefore important and essential to develop methodologies for predicting the complex properties of ultra-lightweight foams. To support NASA programs such as the Reusable Launch Vehicle (RLV), Clark Atlanta University, along with SORDAL, Inc., has initiated projects for commercial process development of polyimide foams for the proposed cryogenic tank integrated structure (see figure 1). Fabrication and characterization of high temperature, advanced aerospace-grade polyimide foams and filled foam sandwich composites for specified lifetimes in NASA space applications, as well as quantifying the lifetime of components, are immensely attractive goals. In order to improve the development, durability, safety, and life cycle performance of ultra-lightweight polymeric foams, test methods for the properties are constant concerns in terms of timeliness, reliability, and cost. A major challenge is to identify the mechanisms of failures (i.e., core failure, interfacial debonding, and crack development) that are reflected in the measured properties. The long-term goal of the this research is to develop the tools and capabilities necessary to successfully engineer ultra-lightweight polymeric foams. The desire is to reduce density at the material and structural levels, while at the same time maintaining or increasing mechanical and other properties.
Author
Foams; Polyimides; Technology Utilization; Costs; Fabrication

20030062263 Boeing Co., Huntington Beach, CA, USA

Thermal-Mechanical Cyclic Test of a Composite Cryogenic Tank for Reusable Launch Vehicles

Messinger, Ross; Pulley, John; [2003]; In English; 44th AIAA/ASME/ASCE/AHS/ASC Structure, Structural Dynamics, & Material Conference, 7-10 Apr. 2003, Norfolk, VA, USA

Contract(s)/Grant(s): NCC8-39; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation provides an overview of thermal-mechanical cyclic tests conducted on a composite cryogenic tank designed for reusable launch vehicles. Topics covered include: a structural analysis of the composite cryogenic tank, a description of Marshall Space Flight Center's Cryogenic Structure Test Facility, cyclic test plans and accomplishments, burst test and analysis and post-testing evaluation.

CASI

Cryogenic Fluid Storage; Storage Tanks; Composite Materials; Destructive Tests; Cryogenic Fluids; Thermal Cycling Tests; Mechanical Properties

20030062173 NASA Marshall Space Flight Center, Huntsville, AL, USA

Vision for CFD-Based Combustion Instability Predictions

Rocker, Marvin; West, Jeffrey S.; April 15, 2003; In English; NASA Spring Workshop on Fluids, 22-24 Apr. 2003, Birmingham, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph representation presents an overview of the proposed development for computational fluid dynamics (CFD) based combustion instability predictions. These predictions are discussed in regard to Next Generation Launch Technologies (NGLT). Topics covered include: relevance of combustion instability to NGLT, evolution of combustion instability prediction techniques, and development of CFD-based models to support combustion instability predictions to support NGLT and beyond.

CASI

Combustion Stability; Computational Fluid Dynamics; Prediction Analysis Techniques; Launch Vehicles; Spacecraft Propulsion; Mathematical Models

20030062123 NASA Langley Research Center, Hampton, VA, USA

Comparison of Methods for Determining Boundary Layer Edge Conditions for Transition Correlations

Liechty, Derek S.; Berry, Scott A.; Hollis, Brian R.; Horvath, Thomas J.; [2003]; In English; 33rd AIAA Fluid Dynamics Conference and Exhibit, 23-26 Jun. 2003, Orlando, FL, USA; Original contains color illustrations

Report No.(s): AIAA Paper 2003-3590; No Copyright; Avail: CASI; A03, Hardcopy

Data previously obtained for the X-33 in the NASA Langley Research Center 20-Inch Mach 6 Air Tunnel have been reanalyzed to compare methods for determining boundary layer edge conditions for use in transition correlations. The experimental results were previously obtained utilizing the phosphor thermography technique to monitor the status of the boundary layer downstream of discrete roughness elements via global heat transfer images of the X-33 windward surface. A boundary layer transition correlation was previously developed for this data set using boundary layer edge conditions calculated using an inviscid/integral boundary layer approach. An algorithm was written in the present study to extract boundary layer edge quantities from higher fidelity viscous computational fluid dynamic solutions to develop transition correlations that account for viscous effects on vehicles of arbitrary complexity. The boundary layer transition correlation developed for the X-33 from the viscous solutions are compared to the previous boundary layer transition correlations. It is shown that the boundary layer edge conditions calculated using an inviscid/integral boundary layer approach are significantly different than those extracted from viscous computational fluid dynamic solutions. The present results demonstrate the differences obtained in correlating transition data using different computational methods.

Author

Boundary Layer Transition; Inviscid Flow; Computational Fluid Dynamics; Surface Roughness; Viscous Flow

20030062069 NASA Marshall Space Flight Center, Huntsville, AL, USA

Systems Engineering Approach to Technology Integration for NASA's 2nd Generation Reusable Launch Vehicle

Thomas, Dale; Smith, Charles; Thomas, Leann; Kittredge, Sheryl; [2002]; In English; Space Technology Applications International Forum, 2-6 Feb. 2003, Albuquerque, NM, USA; Copyright; Avail: Other Sources; Abstract Only

The overall goal of the 2nd Generation RLV Program is to substantially reduce technical and business risks associated with developing a new class of reusable launch vehicles. NASA's specific goals are to improve the safety of a 2nd-generation system by 2 orders of magnitude - equivalent to a crew risk of 1-in-10,000 missions - and decrease the cost tenfold, to

approximately $1,000 per pound of payload launched. Architecture definition is being conducted in parallel with the maturating of key technologies specifically identified to improve safety and reliability, while reducing operational costs. An architecture broadly includes an Earth-to-orbit reusable launch vehicle, on-orbit transfer vehicles and upper stages, mission planning, ground and flight operations, and support infrastructure, both on the ground and in orbit. The systems engineering approach ensures that the technologies developed - such as lightweight structures, long-life rocket engines, reliable crew escape, and robust thermal protection systems - will synergistically integrate into the optimum vehicle. To best direct technology development decisions, analytical models are employed to accurately predict the benefits of each technology toward potential space transportation architectures as well as the risks associated with each technology. Rigorous systems analysis provides the foundation for assessing progress toward safety and cost goals. The systems engineering review process factors in comprehensive budget estimates, detailed project schedules, and business and performance plans, against the goals of safety, reliability, and cost, in addition to overall technical feasibility. This approach forms the basis for investment decisions in the 2nd Generation RLV Program's risk-reduction activities. Through this process, NASA will continually refine its specialized needs and identify where Defense and commercial requirements overlap those of civil missions.
Author
Reusable Launch Vehicles; Risk; Safety; Reliability; Flight Operations; Ground Operational Support System; Mission Planning

20030061202 NASA Marshall Space Flight Center, Huntsville, AL, USA
Overview of MSFC's Applied Fluid Dynamics Analysis Group Activities
Garcia, Roberto; Griffin, Lisa; Williams, Robert; [2002]; In English; MSFC Fall Fluids Workshop, 19-21 Nov. 2002, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph report presents an overview of activities and accomplishments of NASA's Marshall Space Flight Center's Applied Fluid Dynamics Analysis Group. Expertise in this group focuses on high-fidelity fluids design and analysis with application to space shuttle propulsion and next generation launch technologies. Topics covered include: computational fluid dynamics research and goals, turbomachinery research and activities, nozzle research and activities, combustion devices, engine systems, MDA development and CFD process improvements.
CASI
Computational Fluid Dynamics; NASA Programs; Research Facilities; Turbomachinery; Systems Engineering; Design Analysis; Spacecraft Propulsion

20030061159 NASA Marshall Space Flight Center, Huntsville, AL, USA
Systems Engineering Approach to Technology Integration for NASA's 2nd Generation Reusable Launch Vehicle
Thomas, Dale; Smith, Charles; Thomas, Leann; Kittredge, Sheryl; [2002]; In English; Space Technology Applications International Forum, 2-6 Feb. 2003, Albuquerque, NM, USA; No Copyright; Avail: CASI; A03, Hardcopy

The overall goal of the 2nd Generation RLV Program is to substantially reduce technical and business risks associated with developing a new class of reusable launch vehicles. NASA's specific goals are to improve the safety of a 2nd generation system by 2 orders of magnitude - equivalent to a crew risk of 1-in-10,000 missions - and decrease the cost tenfold, to approximately $1,000 per pound of payload launched. Architecture definition is being conducted in parallel with the maturating of key technologies specifically identified to improve safety and reliability, while reducing operational costs. An architecture broadly includes an Earth-to-orbit reusable launch vehicle, on-orbit transfer vehicles and upper stages, mission planning, ground and flight operations, and support infrastructure, both on the ground and in orbit. The systems engineering approach ensures that the technologies developed - such as lightweight structures, long-life rocket engines, reliable crew escape, and robust thermal protection systems - will synergistically integrate into the optimum vehicle. To best direct technology development decisions, analytical models are employed to accurately predict the benefits of each technology toward potential space transportation architectures as well as the risks associated with each technology. Rigorous systems analysis provides the foundation for assessing progress toward safety and cost goals. The systems engineering review process factors in comprehensive budget estimates, detailed project schedules, and business and performance plans, against the goals of safety, reliability, and cost, in addition to overall technical feasibility. This approach forms the basis for investment decisions in the 2nd Generation RLV Program's risk-reduction activities. Through this process, NASA will continually refine its specialized needs and identify where Defense and commercial requirements overlap those of civil missions.
Author
Reusable Launch Vehicles; Systems Engineering; Technology Assessment; Space Transportation; Safety Factors; Ground Operational Support System

20030060643 NASA Marshall Space Flight Center, Huntsville, AL, USA

Base-Bleed Effect on X-33 Aerospike Plume Induced Base-Heating Environment During Power-Pack Out

Wang, Tee-See; Droege, Alan; D'Agostino, Mark; Lee, Young-Ching; Williams, Robert; [2003]; In English; 36th AIAA Thermophysics Conference, 23-26 Jun. 2003, Orlando, FL, USA; No Copyright; Avail: CASI; A03, Hardcopy

A computational heat transfer methodology was developed to study the dual-engine linear aerospike plume induced base-heating environment during one power-pack out, in ascent flight. One power-pack out results in reduction of power levels for both engines. That, in turn, reduces the amount of base-bleed and changes the distribution of base-bleed on the two pillows. Hence, the concern of increased base-heating during power-pack out. The thermo-flowfield of the entire vehicle was computed. The computational methodology for the convective heating is based on a three-dimensional, finite-volume, viscous, chemically reacting, and pressure-based computational fluid dynamics formulation. The computational methodology for the radiative heating is based on a three-dimensional, finite-volume, and spectral-line-based weighted-sum-of-gray-gases absorption computational radiation heat transfer formulation. A separate radiation model was used for diagnostic purposes. The computational methodology was systematically benchmarked. In this study, near-base radiative heat fluxes were computed and they compared well with those measured from an installed linear aerospike engine tests. The base-heating environment of 18 trajectory points selected from three power-pack out ascent scenarios was computed and is presented here. The power-pack out condition has the most impact on convective base-heating when it happens early in flight. The some of its impact comes from the asymmetric and reduced base-bleed.

Author

Aerospike Engines; X-33 Reusable Launch Vehicle; Base Heating; Plumes

20030060421 NASA Marshall Space Flight Center, Huntsville, AL, USA

CFD-Based Design Optimization for Single Element Rocket Injector

Vaidyanathan, Rajkumar; Tucker, Kevin; Papila, Nilay; Shyy, Wei; [2003]; In English; 41st Aerospace Sciences Meeting and Exhibit, 6-9 Jan. 2003, Reno, NV, USA; Copyright; Avail: CASI; A03, Hardcopy

To develop future Reusable Launch Vehicle concepts, we have conducted design optimization for a single element rocket injector, with overall goals of improving reliability and performance while reducing cost. Computational solutions based on the Navier-Stokes equations, finite rate chemistry, and the k-E turbulence closure are generated with design of experiment techniques, and the response surface method is employed as the optimization tool. The design considerations are guided by four design objectives motivated by the consideration in both performance and life, namely, the maximum temperature on the oxidizer post tip, the maximum temperature on the injector face, the adiabatic wall temperature, and the length of the combustion zone. Four design variables are selected, namely, H2 flow angle, H2 and O2 flow areas with fixed flow rates, and O2 post tip thickness. In addition to establishing optimum designs by varying emphasis on the individual objectives, better insight into the interplay between design variables and their impact on the design objectives is gained. The investigation indicates that improvement in performance or life comes at the cost of the other. Best compromise is obtained when improvements in both performance and life are given equal importance.

Author

Computational Fluid Dynamics; Injectors; Reusable Launch Vehicles; Design Optimization; Rocket Engines; Space Shuttle Main Engine

20030059030 NASA Langley Research Center, Hampton, VA, USA

X-33 Computational Aeroheating/Aerodynamic Predictions and Comparisons With Experimental Data

Hollis, Brian R.; Thompson, Richard A.; Berry, Scott A.; Horvath, Thomas J.; Murphy, Kelly J.; Nowak, Robert J.; Alter, Stephen J.; January 05, 2003; In English; Original contains color and black and white illustrations

Contract(s)/Grant(s): WU 762-30-51-30

Report No.(s): NASA/TP-2003-212160; L-18254; NAS 1.60:212160; No Copyright; Avail: CASI; A05, Hardcopy

This report details a computational fluid dynamics study conducted in support of the phase II development of the X-33 vehicle. Aerodynamic and aeroheating predictions were generated for the X-33 vehicle at both flight and wind-tunnel test conditions using two finite-volume, Navier-Stokes solvers. Aerodynamic computations were performed at Mach 6 and Mach 10 wind-tunnel conditions for angles of attack from 10 to 50 with body-flap deflections of 0 to 20. Additional aerodynamic computations were performed over a parametric range of free-stream conditions at Mach numbers of 4 to 10 and angles of attack from 10 to 50. Laminar and turbulent wind-tunnel aeroheating computations were performed at Mach 6 for angles of attack of 20 to 40 with body-flap deflections of 0 to 20. Aeroheating computations were performed at four flight conditions with Mach numbers of 6.6 to 8.9 and angles of attack of 10 to 40. Surface heating and pressure distributions, surface

streamlines, flow field information, and aerodynamic coefficients from these computations are presented, and comparisons are made with wind-tunnel data.
Author
Aerodynamic Heating; Computational Fluid Dynamics; Flow Distribution; Navier-Stokes Equation; Pressure Distribution

20030059016 NASA Langley Research Center, Hampton, VA, USA
Advanced Durability and Damage Tolerance Design and Analysis Methods for Composite Structures: Lessons Learned from NASA Technology Development Programs
Harris, Charles E.; Starnes, James H., Jr.; Shuart, Mark J.; June 2003; In English
Contract(s)/Grant(s): 706-85-12-01
Report No.(s): NASA/TM-2003-212420; L-18297; NAS 1.15:212420; No Copyright; Avail: CASI; A03, Hardcopy

Aerospace vehicles are designed to be durable and damage tolerant. Durability is largely an economic life-cycle design consideration whereas damage tolerance directly addresses the structural airworthiness (safety) of the vehicle. However, both durability and damage tolerance design methodologies must address the deleterious effects of changes in material properties and the initiation and growth of microstructural damage that may occur during the service lifetime of the vehicle. Durability and damage tolerance design and certification requirements are addressed for commercial transport aircraft and NASA manned spacecraft systems. The state-of-the-art in advanced design and analysis methods is illustrated by discussing the results of several recently completed NASA technology development programs. These programs include the NASA Advanced Subsonic Technology Program demonstrating technologies for large transport aircraft and the X-33 hypersonic test vehicle demonstrating technologies for a single-stage-to-orbit space launch vehicle.
Author
Aircraft Reliability; Damage; Design Analysis; Microstructure; Life (Durability)

20030059007 NASA Langley Research Center, Hampton, VA, USA
X-33 Rev-F Turbulent Aeroheating Results From Test 6817 in NASA Langley 20-Inch Mach 6 Air Tunnel and Comparisons With Computations
Hollis, Brian R.; Horvath, Thomas J.; Berry, Scott A.; June 2003; In English
Contract(s)/Grant(s): WU 721-28-00-05
Report No.(s): NASA/TM-2003-211962; L-18293; NAS 1.15:211962; No Copyright; Avail: CASI; A04, Hardcopy

Measurements and predictions of the X-33 turbulent aeroheating environment have been performed at Mach 6, perfect-gas air conditions. The purpose of this investigation was to compare measured turbulent aeroheating levels on smooth models, models with discrete trips, and models with arrays of bowed panels (which simulate bowed thermal protections system tiles) with each other and with predictions from two Navier-Stokes codes, LAURA and GASP. The wind tunnel testing was conducted at free stream Reynolds numbers based on length of 1.8 x 10(exp 6) to 6.1 x 10(exp 6) on 0.0132 scale X-33 models at a = 40-deg. Turbulent flow was produced by the discrete trips and by the bowed panels at ill but the lowest Reynolds number, but turbulent flow on the smooth model was produced only at the highest Reynolds number. Turbulent aeroheating levels on each of the three model types were measured using global phosphor thermography and were found to agree to within .he estimated uncertainty (plus or minus 15%) of the experiment. Computations were performed at the wind tunnel free stream conditions using both codes. Turbulent aeroheating levels predicted using the LAURA code were generally 5%-10% lower than those from GASP, although both sets of predictions fell within the experimental accuracy of the wind tunnel data.
Author
X-33 Reusable Launch Vehicle; Aerodynamic Heating; Hypersonic Speed; Turbulent Flow; Scale Models; Wind Tunnel Tests; Computational Fluid Dynamics

20030055618 NASA Glenn Research Center, Cleveland, OH, USA
Affordable Flight Demonstration of the GTX Air-Breathing SSTO Vehicle Concept
Krivanek, Thomas M.; Roche, Joseph M.; Riehl, John P.; Kosnareo, Daniel N.; April 2003; In English; Combustion, Airbreathing Propulsion, Propulsion Systems Hazards, and Modelling and Simulation Subcommittees Joint Meeting, 8-12 Apr. 2002, Destin, FL, USA; Original contains color illustrations
Contract(s)/Grant(s): WBS 22-708-90-63
Report No.(s): NASA/TM-2003-212315; E-13316; NAS 1.15:212315; APS-III-22; No Copyright; Avail: CASI; A03, Hardcopy

The rocket based combined cycle (RBCC) powered single-stage-to-orbit (SSTO) reusable launch vehicle has the potential

to significantly reduce the total cost per pound for orbital payload missions. To validate overall system performance, a flight demonstration must be performed. This paper presents an overview of the first phase of a flight demonstration program for the GTX SSTO vehicle concept. Phase 1 will validate the propulsion performance of the vehicle configuration over the supersonic and hypersonic air- breathing portions of the trajectory. The focus and goal of Phase 1 is to demonstrate the integration and performance of the propulsion system flowpath with the vehicle aerodynamics over the air-breathing trajectory. This demonstrator vehicle will have dual mode ramjetkcramjets, which include the inlet, combustor, and nozzle with geometrically scaled aerodynamic surface outer mold lines (OML) defining the forebody, boundary layer diverter, wings, and tail. The primary objective of this study is to demon- strate propulsion system performance and operability including the ram to scram transition, as well as to validate vehicle aerodynamics and propulsion airframe integration. To minimize overall risk and develop ment cost the effort will incorporate proven materials, use existing turbomachinery in the propellant delivery systems, launch from an existing unmanned remote launch facility, and use basic vehicle recovery techniques to minimize control and landing requirements. A second phase would demonstrate propulsion performance across all critical portions of a space launch trajectory (lift off through transition to all-rocket) integrated with flight-like vehicle systems.
Author
Rocket-Based Combined-Cycle Engines; Single Stage to Orbit Vehicles; Air Breathing Engines; Flight Tests; Reusable Launch Vehicles; Aircraft Design; Hypersonic Aircraft

20030054429 NASA Kennedy Space Center, Cocoa Beach, FL, USA
STS-102 Photo-op/Suit-up/Depart O&C/Launch Discovery On Orbit/Landing/Crew Egress
March 22, 2001; In English; No Copyright; Avail: CASI; V04, Videotape-VHS; B04, Videotape-Beta

The spacecrews of STS-102 and the Expedition 1 and 2 crews of the International Space Station (ISS) are seen in this video, which presents an overview of their activities. The crew consists of Commander Jim Wetherbee, Pilot James Kelly, and Mission Specialists Andrew Thomas, and Paul Richards. The sections of the video include: Photo-op, Suit-up, Depart O&C, Ingress, Launch with Playbacks, On-orbit, Landing with Playbacks, and Crew Egress & Departs. The prelaunch activities are explained by two narrators, and the crew members are assisted in the White Room just before boarding the Space Shuttle Discovery. Isolated views of the shuttle's launch include: VAB, PAD-B, DLTR-3, UCS-23 Tracker, PATRICK IGOR, UCS-10 Tracker, Grandstand, Tower-1, OTV-160, OTV-170, OTV-171, and On-board Camera. The video shows two extravehicular activities (EVAs) to perform work on the ISS, one by astronauts Helms and Voss from Expedition 2, and another by Richards and Thomas. The attachment of the Leonardo Multipurpose Logistics Module, a temporary resupply module, is shown in a series of still images. The on-orbit footage also includes a view of the Nile River, and a crew exhange ceremony between Expedition 1 (Commander Yuri Gidzenko, Flight Engineer Sergei Krikalev) and Expedition 2 (Commander Yury Usachev, Flight Engineers James Voss, Susan Helms). Isolated views of the landing at Kennedy Space Center include: North Runway Camera, VAB, Tower-1, Mid-field, Midfield IR, Tower-2, and UCS-12 IR. The Crew Transfer Vehicle (CTV) for unloading the astronauts is shown, administrators greet the crew upon landing, and Commander Wetherbee gives a briefing.
CASI
Discovery (Orbiter); International Space Station; Spacecrews; Liftoff (Launching); Horizontal Spacecraft Landing; Spacecrew Transfer

20030053149 Vanderbilt Univ., Nashville, TN, USA
System Risk Assessment and Allocation in Conceptual Design
Mahadevan, Sankaran; Smith, Natasha L.; Zang, Thomas A., Technical Monitor; May 2003; In English
Contract(s)/Grant(s): NCC1-01031; RTOP 706-31-61-01
Report No.(s): NASA/CR-2003-212162; NAS 1.26:21262; No Copyright; Avail: CASI; A06, Hardcopy

As aerospace systems continue to evolve in addressing newer challenges in air and space transportation, there exists a heightened priority for significant improvement in system performance, cost effectiveness, reliability, and safety. Tools, which synthesize multidisciplinary integration, probabilistic analysis, and optimization, are needed to facilitate design decisions allowing trade-offs between cost and reliability. This study investigates tools for probabilistic analysis and probabilistic optimization in the multidisciplinary design of aerospace systems. A probabilistic optimization methodology is demonstrated for the low-fidelity design of a reusable launch vehicle at two levels, a global geometry design and a local tank design. Probabilistic analysis is performed on a high fidelity analysis of a Navy missile system. Furthermore, decoupling strategies are introduced to reduce the computational effort required for multidisciplinary systems with feedback coupling.
Author
Multidisciplinary Design Optimization; Aerospace Systems; Probability Theory; Reliability Analysis; Design Analysis; Risk; Systems Engineering; Models

20030046938

Analysis of space concepts enabled by new transportation (ASCENT) study

Webber, Derek; AIP Conference Proceedings; January 14, 2002; ISSN 0094-243X; Volume 608, no. 1; In English; SPACE TECHNOLOGY and APPLICATIONS INTERNATIONAL FORUM- STAIF 2002, 3-6 Feb 2002, Albuquerque, New Mexico, USA; Copyright

The ASCENT Study is evaluating all those markets (both commercial and government missions) that may be served by a future Second Generation Reusable Launch Vehicle. The markets are being analyzed, and twenty-year forecasts are being generated, in order to assist NASA in defining the attributes of the Second Generation RLV that will eventually replace the Space Shuttle. The forecasts will make possible an evaluation of alternative architectures for the Second Generation RLV, and also help assess the extent to which new commercial markets, enabled by the proposed pricing and reliability benefits of the Second Generation RLV, can contribute to the funding of the vehicle. This paper provides a status of the project at the time of the conference, and describes the way in which the markets are being treated and data is being obtained. The sectors range from the relatively well understood telecommunications markets to the newer evolving market opportunities like public space travel. [copyright] 2002 American Institute of Physics.

Author (AIP)

Aerospace Industry; Commerce; Cost Analysis; Cost Effectiveness; Cost Estimates; Market Research; Reliability; Reusable Launch Vehicles; Space Transportation; Spacecraft; Telecommunication; Transportation

20030046831

Advanced spacecraft designs in support of human missions to earth's neighborhood

Fletcher, David; AIP Conference Proceedings; January 14, 2002; ISSN 0094-243X; Volume 608, no. 1; In English; SPACE TECHNOLOGY and APPLICATIONS INTERNATIONAL FORUM- STAIF 2002, 3-6 Feb 2002, Albuquerque, New Mexico, USA; Copyright

NASA's strategic planning for technology investment draws on engineering studies of potential future missions. A number of hypothetical mission architectures have been studied. A recent study completed by the NASA/JSC Advanced Design Team addresses one such possible architecture strategy for missions to the moon. This conceptual study presents an overview of each of the spacecraft elements that would enable such missions. These elements include an orbiting lunar outpost at lunar L1 called the Gateway, a crew transfer vehicle (CTV) which ferries a crew of four from the ISS to the Gateway, a lunar lander which ferries the crew from the Gateway to the lunar surface, and a one-way lunar habitat lander capable of supporting the crew for 30 days. Other supporting elements of this architecture discussed below include the CTV kickstage, a solar-electric propulsion (SEP) stage, and a logistics lander capable of re-supplying the 30-day habitat lander and bringing other payloads totaling 10. 3 mt in support of surface mission activities. Launch vehicle infrastructure to low-earth orbit includes the Space Shuttle, which brings up the CTV and crew, and the Delta-IV Heavy expendable launch vehicle which launches the landers, kickstage, and SEP. [copyright] 2002 American Institute of Physics.

Author (AIP)

Earth Orbits; Electric Propulsion; Lunar Surface; Management Planning; Spacecraft

20030046819

Estimate of avoidance maneuver rate for HASTOL tether boost facility

Forward, Robert L.; AIP Conference Proceedings; January 14, 2002; ISSN 0094-243X; Volume 608, no. 1; In English; SPACE TECHNOLOGY and APPLICATIONS INTERNATIONAL FORUM- STAIF 2002, 3-6 Feb 2002, Albuquerque, New Mexico, USA; Copyright

The Hypersonic Airplane Space Tether Orbital Launch (HASTOL) Architecture uses a hypersonic airplane (or reusable launch vehicle) to carry a payload from the surface of the Earth to 150 km altitude and a speed of Mach 17. The hypersonic airplane makes a rendezvous with the grapple at the tip of a long, rotating, orbiting space tether boost facility, which picks up the payload from the airplane. Release of the payload at the proper point in the tether rotation boosts the payload into a higher orbit, typically into a Geosynchronous Transfer Orbit (GTO), with lower orbits and Earth escape other options. The HASTOL Tether Boost Facility will have a length of 636 km. Its center of mass will be in a 604 km by 890 km equatorial orbit. It is estimated that by the time of the start of operations of the HASTOL Tether Boost facility in the year 2020, there will be 500 operational spacecraft using the same volume of space as the HASTOL facility. These operational spacecraft would likely be made inoperative by an impact with one of the lines in the multiline HASTOL Hoytether[trademark] and should be avoided. There will also be non-operational spacecraft and large pieces of orbital debris with effective size greater than five meters in diameter that could cut a number of lines in the HASTOL Hoytether[trademark], and should also be avoided. It is estimated, using two different methods and combining them, that the HASTOL facility will need to make avoidance

maneuvers about once every four days if the 500 operational spacecraft and large pieces of orbital debris greater than 5 m in diameter, were each protected by a 2 km diameter miss distance protection sphere. If by 2020, the ability to know the positions of operational spacecraft and large pieces of orbital debris improved to allow a 600 m diameter miss distance protection sphere around each object, then the number of HASTOL facility maneuvers needed drops to one every two weeks. [copyright] 2002 American Institute of Physics.
Author (AIP)
Artificial Satellites; Earth Orbits; Hypersonic Vehicles; Mach Number; Reusable Launch Vehicles; Space Debris; Spacecraft Launching

20030032198 NASA Glenn Research Center, Cleveland, OH, USA
Advanced Control Surface Seal Development at NASA GRC for 3rd Generation RLV
Dunlap, Patrick H., Jr.; Steinetz, Bruce M.; DeMange, Jeffrey J.; Breen, Daniel P.; November 12, 2002; In English; RLV/SOV Airframe Technology Review, 19-22 Nov. 2002, Hampton, VA, USA; Original contains black and white illustrations; Copyright; Avail: CASI; A03, Hardcopy

NASA's Glenn Research Center (GRC) is developing advanced control surface seal technologies for the 3rd Generation RLV program. GRC has designed several new test rigs to simulate the temperatures, pressures, and scrubbing conditions that seals would have to endure during service. A hot compression test rig and hot scrub test rig are being developed to perform tests at temperatures up to 3000 F. Another new test rig allows simultaneous seal flow and scrub tests at room temperature to evaluate changes in seal performance with scrubbing. High temperature ceramic preloading devices are being developed and evaluated as ways to improve seal resiliency.
Author
Reusable Launch Vehicles; Control Surfaces; Design Analysis; Temperature; Simulation; Pressure; Seals (Stoppers)

20030022661 NASA Glenn Research Center, Cleveland, OH, USA
High Altitude Launch for a Practical SSTO
Landis, Geoffrey A.; Denis, Vincent; Lyons, Valerie, Technical Monitor; February 2002; In English; Conference on Next Generation Space Transportation, Space Technology and Applications Forum, 2-6 Feb. 2002, Albuquerque, NM, USA; Copyright; Avail: CASI; A02, Hardcopy

Existing engineering materials allow the construction of towers to heights of many kilometers. Orbital launch from a high altitude has significant advantages over sea-level launch due to the reduced atmospheric pressure, resulting in lower atmospheric drag on the vehicle and allowing higher rocket engine performance. High-altitude launch sites are particularly advantageous for single-stage to orbit (SSTO) vehicles, where the payload is typically 2% of the initial launch mass. An earlier paper enumerated some of the advantages of high altitude launch of SSTO vehicles. In this paper, we calculate launch trajectories for a candidate SSTO vehicle, and calculate the advantage of launch at launch altitudes 5 to 25 kilometer altitudes above sea level. The performance increase can be directly translated into increased payload capability to orbit, ranging from 5 to 20% increase in the mass to orbit. For a candidate vehicle with an initial payload fraction of 2% of gross lift-off weight, this corresponds to 31% increase in payload (for 5-km launch altitude) to 122% additional payload (for 25-km launch altitude).
Author
High Altitude; Towers; Spacecraft Launching; Materials Selection; Trajectory Analysis; Single Stage to Orbit Vehicles

20030020634 NASA Glenn Research Center, Cleveland, OH, USA
Towers for Earth Launch
Landis, Geoffrey A.; Lyons, Valerie J., Technical Monitor; September 2002; In English; 1st Workshop on Space Elevators, 12-13 Aug. 2002, Seattle, WA., USA
Contract(s)/Grant(s): RTOP 755-A4-02; No Copyright; Avail: CASI; A03, Hardcopy

This report lists some characteristics of a hypothetical 15 kilometer tower for launching spacecraft, the advantages of launching from high altitude, and some equations pertaining to launch from a 15 kilometer tower.
CASI
High Altitude; Spacecraft Launching; Towers

20030016552 NASA Marshall Space Flight Center, Huntsville, AL USA
Analysis of X-33 Linear Aerospike Plume Induced Base-Heating Environment During Power-Pack Out
Wang, Ten-See; Williams, Robert; Droege, Alan; Dagnostino, Mark; Lee, Young-Ching; Douglas, Stan; Turner, James E., Technical Monitor; [2001]; In English; NASA/MSFC Fluids Workshop, 4-5 Apr. 2001, Huntsville, AL, USA; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

The objectives of this viewgraph presentation are to predict the following: (1) dual-engine base-heating at 57% PL at sea level, and (2) dual-engine base-heating during PPO at three ascent abort trajectories. A systematically anchored computational fluid dynamics and heat transfer three-dimensional transfer simulation is being used to study the effect of reduced power levels on base-heating environments during sea level testing and during PPO. Preliminary results show the following: (1) convective heating is higher for the 57% PL than for 100% PL on most of the pillows and flex seals during sea level testing; and (2) convective heating on pillows and flex seals on the 'off' engine side is higher than that on the 'on' engine side.
Derived from text
Computational Fluid Dynamics; Convective Heat Transfer; X-33 Reusable Launch Vehicle; Computerized Simulation; Three Dimensional Models

20030015407 NASA Ames Research Center, Moffett Field, CA USA
Progress in Unsteady Turbopump Flow Simulations Using Overset Grid Systems
Kiris, Cetin C.; Chan, William; Kwak, Dochan; [2002]; In English; NASA MSFC 2002 Workshop, 19-21 Nov. 2002, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation provides information on unsteady flow simulations for the Second Generation RLV (Reusable Launch Vehicle) baseline turbopump. Three impeller rotations were simulated by using a 34.3 million grid points model. MPI/OpenMP hybrid parallelism and MLP shared memory parallelism has been implemented and benchmarked in INS3D, an incompressible Navier-Stokes solver. For RLV turbopump simulations a speed up of more than 30 times has been obtained. Moving boundary capability is obtained by using the DCF module. Scripting capability from CAD geometry to solution is developed. Unsteady flow simulations for advanced consortium impeller/diffuser by using a 39 million grid points model are currently underway. 1.2 impeller rotations are completed. The fluid/structure coupling is initiated.
Derived from text
Turbine Pumps; Unsteady Flow; Computational Grids; Impellers; Simulation; Rotation

20030014810 NASA Marshall Space Flight Center, Huntsville, AL USA
Advanced Guidance and Control Methods for Reusable Launch Vehicles: Test Results
Hanson, John M.; Jones, Robert E.; Krupp, Don R.; Fogle, Frank R., Technical Monitor; [2002]; In English; AIAA Guidance, Navigation and Control Conference, 5-8 Aug. 2002, Monterey, CA, USA; Copyright; Avail: CASI; A03, Hardcopy; Distribution as joint owner in the copyright

There are a number of approaches to advanced guidance and control (AG&C) that have the potential for achieving the goals of significantly increasing reusable launch vehicle (RLV) safety/reliability and reducing the cost. In this paper, we examine some of these methods and compare the results. We briefly introduce the various methods under test, list the test cases used to demonstrate that the desired results are achieved, show an automated test scoring method that greatly reduces the evaluation effort required, and display results of the tests. Results are shown for the algorithms that have entered testing so far.
Author
Reusable Launch Vehicles; Flight Control; Flight Tests; Entry Guidance (Sts); Control Systems Design; Methodology

20030014743 NASA Ames Research Center, Moffett Field, CA USA
Calculation of Supersonic Combustion Using Implicit Schemes
Yoon, Seokkwan; Kwak, Dochan, Technical Monitor; [2003]; In English; AIAA 16th Computational Fluid Dynamics Conference, 23-26 Jun. 2003, Orlando, FL, USA; No Copyright; Avail: CASI; A01, Hardcopy

One of the technology goals of NASA for advanced space transportation is to develop highly efficient propulsion systems to reduce the cost of payload for space missions. Developments of rockets for the second generation Reusable Launch Vehicle (RLV) in the past several years have been focused on low-cost versions of conventional engines. However, recent changes in the Integrated Space Transportation Program to build a crew transportation vehicle to extend the life of the Space Shuttle fleet might suggest that air-breathing rockets could reemerge as a possible propulsion system for the third generation RLV to replace the Space Shuttle after 2015. The weight of the oxygen tank exceeds thirty percent of the total weight of the Space Shuttle at launch while the payload is only one percent of the total weight. The air-breathing rocket propulsion systems, which consume oxygen in the air, offer clear advantages by making vehicles lighter and more efficient. Experience in the National Aerospace Plane Program in the late 1980s indicates that scramjet engines can achieve high specific impulse for low hypersonic vehicle speeds. Whether taking a form of Rocket Based Combined Cycle (RBCC) or Turbine Based Combined Cycle (TBCC), the scramjet is an essential mode of operation for air-breathing rockets. It is well known that fuel-air mixing

and rapid combustion are of crucial importance for the success of scramjet engines since the spreading rate of the supersonic mixing layer decreases as the Mach number increases. A factored form of the Gauss-Seidel relaxation method has been widely used in hypersonic flow research since its first application to non-equilibrium flows. However, difficulties in stability and convergence have been encountered when there is strong interaction between fluid motion and chemical reaction, such as multiple fuel injection problems. The present paper reports the results from investigation of the effect of modifications to the original algorithm on the performance for multiple injectors.
Author
Reusable Launch Vehicles; Supersonic Combustion Ramjet Engines; Rocket-Based Combined-Cycle Engines; Hypersonic Vehicles

20030014601 NASA Ames Research Center, Moffett Field, CA USA
Thermal Protection System Evaluation Using Arc-jet Flows: Flight Simulation or Research Tool?
Stewart, David A.; Venkatapathy, Ethiras, Technical Monitor; Nov. 20, 2002; In English; 105th Annual Meeting and Exposition of the American Ceramic Society, 27-30 Apr. 203, Nashville, TN, USA; No Copyright; Avail: Other Sources; Abstract Only

The arc-jet has been used to evaluate thermal protection systems (TPS) and materials for the past forty years. Systems that have been studied in this environmerd include ablators, active, and passive TPS concepts designed for vehicles entering planetary and Earth atmospheres. The question of whether arc-jet flow can simulate a flight environment or is it a research tool that provides an aero-thermodynamic heating environment to obtain critical material properties will be addressed. Stagnation point tests in arc-jets are commonly used to obtain material properties such as mass loss rates, thermal chemical stability data, optical properties, and surface catalytic efficiency. These properties are required in computational fluid dynamic codes to accurately predict the performance of a TPS during flight. Special facilities have been developed at NASA Ames Research Center to approximate the flow environment over the mid-fuselage and body flap regions of proposed space-planes type vehicles. This paper compares flow environments generated in flight over a vehicle with those created over an arc-jet test articles in terms of scale, chemistry, and fluid dynamic properties. Flight experiments are essential in order to validate the material properties obtained from arc-jet tests and used to predict flight performance of any TPS being considered for use on a vehicle entering the Earth atmosphere at hypersonic speed.
Author
Thermal Protection; Flight Simulation; Arc Jet Engines; Jet Flow; Computational Fluid Dynamics; Aerothermodynamics

20030013452 NASA Marshall Space Flight Center, Huntsville, AL USA
NASA 2nd Generation RLV Program Introduction, Status and Future Plans
Dumbacher, Dan L.; Smith, Dennis E., Technical Monitor; [2002]; In English; 38th Joint Propulsion Conference, 7-10 Jul. 2002, Indianapolis, IN, USA
Report No.(s): AIAA Paper 2002-4570; Copyright; Avail: CASI; A02, Hardcopy; Distribution under U.S. Government purpose rights

The Space Launch Initiative (SLI), managed by the Second Generation Reusable Launch Vehicle (2ndGen RLV) Program, was established to examine the possibility of revolutionizing space launch capabilities, define conceptual architectures, and concurrently identify the advanced technologies required to support a next-generation system. Initial Program funds have been allocated to design, evaluate, and formulate realistic plans leading to a 2nd Gen RLV full-scale development (FSD) decision by 2006. Program goals are to reduce both risk and cost for accessing the limitless opportunities afforded outside Earth's atmosphere fo civil, defense, and commercial enterprises. A 2nd Gen RLV architecture includes a reusable Earth-to-orbit launch vehicle, an on-orbit transport and return vehicle, ground and flight operations, mission planning, and both on-orbit and on-the-ground support infrastructures All segments of the architecture must advance in step with development of the RLV if a next-generation system is to be fully operational early next decade. However, experience shows that propulsion is the single largest contributor to unreliability during ascent, requires the largest expenditure of time for maintenance, and takes a long time to develop; therefore, propulsion is the key to meeting safety, reliability, and cost goals. For these reasons, propulsion is SLI's top technology investment area.
Author
Reusable Launch Vehicles; Spacecraft Launching; NASA Space Programs; Spacecraft Design; Technology Utilization

20030007706 NASA Langley Research Center, Hampton, VA USA

X-33 Turbulent Aeroheating Measurements and Predictions

Hollis, Brian R.; Berry, Scott A.; Horvath, Thomas J.; [2002]; In English; AIAA Atmospheric Flight Mechanics Conference and Exhibit, 5-8 Aug. 2002, Monterey, CA, USA; Original contains color illustrations

Report No.(s): AIAA Paper 2002-4700; Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

Measurements and predictions of the X-33 turbulent aeroheating environment have been performed for Mach 6, perfect-gas air conditions. The purpose of this investigation was to compare turbulent aeroheating predictions from two Navier-Stokes codes, LAURA and GASP, with each other and with experimental data in which turbulent flow was produced through either natural transition or forced transition using roughness elements. The wind tunnel testing was conducted at free stream Reynolds numbers of 0.72 x 10(exp 7)/m to 2.4 x 10(exp 7)/m (2.2 x 10(exp 6)/ft to 7.3 x 10(exp 6)/ft) on 0.254 m (10.0-in.) X-33 models at alpha = 40 deg with smooth surfaces, smooth surfaces with discrete trips, and surfaces with simulated bowed thermal protection system panels. Turbulent flow was produced by the discrete trips and bowed panels for all but the lowest Reynolds number, while turbulent flow on the smooth model was produced only at the highest Reynolds number. Turbulent aeroheating levels on each of the three model types were measured using global phosphor thermography and agreed to within the experimental accuracy (+/= 15%) of the test technique. Computations were performed at the wind tunnel free stream conditions using both codes. Turbulent aeroheating levels predicted using the LAURA code were generally 5%-10% lower than those from GASP, although both sets of predictions fell within the experimental accuracy of the wind tunnel data.

Author

Aerodynamic Heating; Wind Tunnel Tests; Navier-Stokes Equation; Turbulent Flow; Reynolds Number; Thermography; Computer Programs; Computational Fluid Dynamics

20030006793

High Altitude Launch for a Practical SSTO

Landis, Geoffrey A.; Denis, Vincent; AIP Conference Proceedings; January 28, 2003; ISSN 0094-243X; Volume 654, no. 1; In English; SPACE TECHNOLOGY and APPLICATIONS INT.FORUM-STAIF 2003: Conf.on Thermophysics in Microgravity; Commercial/Civil Next Generation Space Transportation; Human Space Exploration, 2-5 February 2003, Albuquerque, New Mexico, USA; Copyright

Existing engineering materials allow the constuction of towers to heights of many kilometers. Orbital launch from a high altitude has significant advantages over sea-level launch due to the reduced atmospheric pressure, resulting in lower atmospheric drag on the vehicle and allowing higher rocket engine performance. High-altitude launch sites are particularly advantageous for single-stage to orbit (SSTO) vehicles, where the payload is typically 2% of the initial launch mass. An earlier paper enumerated some of the advantages of high altitude launch of SSTO vehicles. In this paper, we calculate launch trajectories for a candidate SSTO vehicle, and calculate the advantage of launch at launch altitudes 5 to 25 kilometer altitudes above sea level. The performance increase can be directly translated into increased payload capability to orbit, ranging from 5 to 20% increase in the mass to orbit. For a candidate vehicle with an initial payload fraction of 2% of gross lift-off weight, this corresponds to 31% increase in payload (for 5-km launch altitude) to 122% additional payload (for 25-km launch altitude).
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Author (AIP)

Cost Estimates; Density; Fractals; Ground Support Systems; High Altitude; Single Stage to Orbit Vehicles; Spacecraft; Structural Engineering; Towers

20030006266 NASA Marshall Space Flight Center, Huntsville, AL USA

Propellant Densification for Shuttle: The SSME Perspective

Greene, William D.; Boxx, Dayna L.; Tiller, Bruce K., Technical Monitor; May 20, 2002; In English; 38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 7-10 Jul. 2002, Indianapolis, IN, USA

Report No.(s): AIAA Paper 2002-3602; Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

The subject of cryogenic propellant densification as a potential upgrade to the Space Shuttle is a subject that has been raised on several occasions over the last decade. Due to advancements in densification technology made as a part of and in parallel to the X-33 project, the subject was raised and studied once again in May 2001. Across the Space Shuttle program people from many disciplines converged to discuss issues and perform trade studies to determine whether densified propellants was worth pursuing. This paper discusses one of these areas, specifically the Space Shuttle Main Engine (SSME). The effects

of propellant densification on steady state performance are presented along with discussions of potential transient performance issues. Engine component redesign and retrofit issues are discussed as well the high level requirements to modify the ground test stands to accommodate propellant densification hardware and tanks. And finally, the matter of programmatic concerns enters the subject at hand as part of a discussion of SSME recertification requirements. In the end, potential benefits to SSME performance can be demonstrated and, subject to the densification scheme chosen, there does not appear to insurmountable technical obstacles.
Author
Space Shuttle Main Engine; Cryogenic Rocket Propellants

20030006264 NASA Langley Research Center, Hampton, VA USA
Preliminary Thermal-Mechanical Sizing of Metallic TPS: Process Development and Sensitivity Studies
Poteet, Carl C.; Abu-Khajeel, Hasan; Hsu, Su-Yuen; [2002]; In English; 40th AIAA Aerospace Sciences Meeting and Exhibit, 14-17 Jan. 2002, Reno, NV, USA
Report No.(s): AIAA Paper 2002-0505; Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

The purpose of this research was to perform sensitivity studies and develop a process to perform thermal and structural analysis and sizing of the latest Metallic Thermal Protection System (TPS) developed at NASA LaRC (Langley Research Center). Metallic TPS is a key technology for reducing the cost of reusable launch vehicles (RLV), offering the combination of increased durability and competitive weights when compared to other systems. Accurate sizing of metallic TPS requires combined thermal and structural analysis. Initial sensitivity studies were conducted using transient one-dimensional finite element thermal analysis to determine the influence of various TPS and analysis parameters on TPS weight. The thermal analysis model was then used in combination with static deflection and failure mode analysis of the sandwich panel outer surface of the TPS to obtain minimum weight TPS configurations at three vehicle stations on the windward centerline of a representative RLV. The coupled nature of the analysis requires an iterative analysis process, which will be described herein. Findings from the sensitivity analysis are reported, along with TPS designs at the three RLV vehicle stations considered.
Author
Sensitivity Analysis; Thermodynamics; Thermal Analysis; Structural Analysis; Thermal Protection

20030005865 NASA Langley Research Center, Hampton, VA USA
Investigation of Fundamental Modeling and Thermal Performance Issues for a Metallic Thermal Protection System Design
Blosser, Max L.; [2002]; In English; 40th Aerospace Sciences Meeting and Exhibit, 14-17 Jan. 2002, Reno, NV, USA
Report No.(s): AIAA Paper 2002-0503; No Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

A study was performed to develop an understanding of the key factors that govern the performance of metallic thermal protection systems for reusable launch vehicles. A current advanced metallic thermal protection system (TPS) concept was systematically analyzed to discover the most important factors governing the thermal performance of metallic TPS. A large number of relevant factors that influence the thermal analysis and thermal performance of metallic TPS were identified and quantified. Detailed finite element models were developed for predicting the thermal performance of design variations of the advanced metallic TPS concept mounted on a simple, unstiffened structure. The computational models were also used, in an automated iterative procedure, for sizing the metallic TPS to maintain the structure below a specified temperature limit. A statistical sensitivity analysis method, based on orthogonal matrix techniques used in robust design, was used to quantify and rank the relative importance of the various modeling and design factors considered in this study. Results of the study indicate that radiation, even in small gaps between panels, can reduce significantly the thermal performance of metallic TPS, so that gaps should be eliminated by design if possible. Thermal performance was also shown to be sensitive to several analytical assumptions that should be chosen carefully. One of the factors that was found to have the greatest effect on thermal performance is the heat capacity of the underlying structure. Therefore the structure and TPS should be designed concurrently.
Author
Mathematical Models; Matrices (Mathematics); Performance Prediction; Sensitivity Analysis; Statistical Analysis; Temperature Effects; Thermal Analysis; Thermal Protection

20030005683 Alabama Univ., Huntsville, AL USA

Improved Re-Configurable Sliding Mode Controller for Reusable Launch Vehicle of Second Generation Addressing Aerodynamic Surface Failures and Thrust Deficiencies

Shtessel, Yuri B.; 2002; In English

Contract(s)/Grant(s): NAG8-1787; No Copyright; Avail: CASI; A03, Hardcopy

In this report we present a time-varying sliding mode control (TV-SMC) technique for reusable launch vehicle (RLV) attitude control in ascent and entry flight phases. In ascent flight the guidance commands Euler roll, pitch and yaw angles, and in entry flight it commands the aerodynamic angles of bank, attack and sideslip. The controller employs a body rate inner loop and the attitude outer loop, which are separated in time-scale by the singular perturbation principle. The novelty of the TVSMC is that both the sliding surface and the boundary layer dynamics can be varied in real time using the PD-eigenvalue assignment technique. This salient feature is used to cope with control command saturation and integrator windup in the presence of severe disturbance or control effector failure, which enhances the robustness and fault tolerance of the controller. The TV-SMC is developed and tuned up for the X-33 sub-orbital technology demonstration vehicle in launch and re-entry modes. A variety of nominal, dispersion and failure scenarios have tested via high fidelity 6DOF simulations using MAVERIC/SLIM simulation software.

Author

Aerodynamics; Controllers; Reusable Launch Vehicles; Sliding; Thrust; Flight Control

20030003786 NASA Glenn Research Center, Cleveland, OH USA

Experimental Evaluation of a Subscale Gaseous Hydrogen/Gaseous Oxygen Coaxial Rocket Injector

Smith, Timothy D.; Klem, Mark D.; Breisacher, Kevin J.; Farhangi, Shahram; Sutton, Robert; November 2002; In English; Original contains color illustrations

Contract(s)/Grant(s): RTOP 708-73-10

Report No.(s): NASA/TM-2002-211982; E-13652; NAS 1.15:211982; Copyright; Avail: CASI; A03, Hardcopy; Distribution as joint owner in the copyright

The next generation reusable launch vehicle may utilize a Full-Flow Stage Combustion (FFSC) rocket engine cycle. One of the key technologies required is the development of an injector that uses gaseous oxygen and gaseous hydrogen as propellants. Gas-gas propellant injection provides an engine with increased stability margin over a range of throttle set points. This paper summarizes an injector design and testing effort that evaluated a coaxial rocket injector for use with gaseous oxygen and gaseous hydrogen propellants. A total of 19 hot-fire tests were conducted up to a chamber pressure of 1030 psia, over a range of 3.3 to 6.7 for injector element mixture ratio. Post-test condition of the hardware was also used to assess injector face cooling. Results show that high combustion performance levels could be achieved with gas-gas propellants and there were no problems with excessive face heating for the conditions tested.

Author

Injectors; Test Facilities; Coaxial Flow; Hydrogen; Oxygen; Gaseous Rocket Propellants; Rocket Engine Design

20030002800 NASA Marshall Space Flight Center, Huntsville, AL USA, Boeing Co., Huntington Beach, CA USA

Launch Vehicle Sizing Benefits Utilizing Main Propulsion System Crossfeed and Project Status

Chandler, Frank; Scheiern, M.; Champion, R.; Mazurkivich, P.; Lyles, Garry, Technical Monitor; Jul. 09, 2002; In English; 38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 7-10 Jul. 2002, Indianapolis, IN, USA

Contract(s)/Grant(s): NAS8-01099

Report No.(s): AIAA Paper 2002-3900; Copyright; Avail: CASI; A02, Hardcopy; Distribution as joint owner in the copyright

To meet the goals for a next generation Reusable Launch Vehicle (RLV), a unique propulsion feed system concept was identified using crossfeed between the booster and orbiter stages that could reduce the Two-Stage-to-Orbit (TSTO) vehicle weight and Design, Development, Test and Evaluation (DDT&E) costs by approximately 25%, while increasing safety and reliability. The Main Propulsion System (MPS) crossfeed water demonstration test program addresses all activities required to reduce the risks for the MPS crossfeed system from a Technology Readiness Level (TRL) of 2 to 4 by the completion of testing and analysis by June 2003. During the initial period, that ended in March 2002, a subscale water flow test article was defined. Procurement of a subscale crossfeed check valve was initiated and the specifications for the various components were developed. The fluid transient and pressurization analytical models were developed separately and successfully integrated. The test matrix for the water flow test was developed to correlate the integrated model. A computational fluid dynamics (CFD) model of the crossfeed check valve was developed to assess flow disturbances and internal flow dynamics. Based on the results, the passive crossfeed system concept was very feasible and offered a safe system to be used in an RLV architecture. A water flow test article was designed to accommodate a wide range of flows simulating a number of different types of

propellant systems. During the follow-on period, the crossfeed system model will be further refined, the test article will be completed, the water flow test will be performed, and finally the crossfeed system model will be correlated with the test data. This validated computer model will be used to predict the full-scale vehicle crossfeed system performance.
Author
Computational Fluid Dynamics; Feed Systems; Mathematical Models; Reusable Launch Vehicles; Computerized Simulation; Technology Assessment; Propulsion System Performance

20030002474 NASA Marshall Space Flight Center, Huntsville, AL USA
Space Ops 2002: Bringing Space Operations into the 21st Century. Track 3: Operations, Mission Planning and Control. 2nd Generation Reusable Launch Vehicle-Concepts for Flight Operations
Hagopian, Jeff; [2002]; In English; No Copyright; Avail: CASI; A02, Hardcopy

With the successful implementation of the International Space Station (ISS), the National Aeronautics and Space Administration (NASA) enters a new era of opportunity for scientific research. The ISS provides a working laboratory in space, with tremendous capabilities for scientific research. Utilization of these capabilities requires a launch system capable of routinely transporting crew and logistics to/from the ISS, as well as supporting ISS assembly and maintenance tasks. The Space Shuttle serves as NASA's launch system for performing these functions. The Space Shuttle also serves as NASA's launch system for supporting other science and servicing missions that require a human presence in space. The Space Shuttle provides proof that reusable launch vehicles are technically and physically implementable. However, a couple of problems faced by NASA are the prohibitive cost of operating and maintaining the Space Shuttle and its relative inability to support high launch rates. The 2nd Generation Reusable Launch Vehicle (2nd Gen RLV) is NASA's solution to this problem. The 2nd Gen RLV will provide a robust launch system with increased safety, improved reliability and performance, and less cost. The improved performance and reduced costs of the 2nd Gen RLV will free up resources currently spent on launch services. These resource savings can then be applied to scientific research, which in turn can be supported by the higher launch rate capability of the 2nd Gen RLV. The result is a win - win situation for science and NASA. While meeting NASA's needs, the 2nd Gen RLV also provides the USA aerospace industry with a commercially viable launch capability. One of the keys to achieving the goals of the 2nd Gen RLV is to develop and implement new technologies and processes in the area of flight operations. NASA's experience in operating the Space Shuttle and the ISS has brought to light several areas where automation can be used to augment or eliminate functions performed by crew and ground controllers. This experience has also identified the need for new approaches to staffing and training for both crew and ground controllers. This paper provides a brief overview of the mission capabilities provided by the 2nd Gen RLV, a description of NASA's approach to developing the 2nd Gen RLV, a discussion of operations concepts, and a list of challenges to implementing those concepts.
Author
Flight Operations; Reusable Launch Vehicles; Space Shuttles; International Space Station; NASA Programs

20030002215 Ohio Univ., USA
On-Board Generation of Three-Dimensional Constrained Entry Trajectories
Shen, Zuojun; Lu, Ping; Jackson, Scott, Technical Monitor; Jan. 09, 2002; In English; 2002 AIAA GN&C Conference, 5-8 Aug. 2002, Monterey, CA, USA
Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A02, Hardcopy

A methodology for very fast design of 3DOF entry trajectories subject to all common inequality and equality constraints is developed. The approach make novel use of the well known quasi-equilibrium glide phenomenon in lifting entry as a center piece for conveniently enforcing the inequality constraints which are otherwise difficulty to handle. The algorithm is able to generate a complete feasible 3DOF entry trajectory, given the entry conditions, values of constraint parameters, and final conditions in about 2 seconds on a PC. Numerical simulations with the X-33 vehicle model for various entry missions to land at Kennedy Space Center will be presented.
Author
Trajectories; Algorithms; Three Dimensional Models; Atmospheric Entry

20030001848 NASA Glenn Research Center, Cleveland, OH USA
NASA Overview
Rusick, Jeff; Fifth Annual Workshop on the Application of Probabilistic Methods for Gas Turbine Engines; October 2002; In English; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation provides information on computer programs for the application of probabilistics in aerospace

design. The presentation defines risk, and lists probabilistic tools which can take risk into account. It then lists current and future examples of NASA aerospace technology program support which utilize probabilistic methods. Current examples include turbofan engines, combustor liners, life estimation, the analysis of rotors and blades, and shuttle ground operations. The 2nd Generation Reusable Launch Vehicle is one of the future areas of support. The presentation also suggests other future uses of probabilistic tools in aerospace design, and optimal methods for their implementation.
CASI
Computer Programs; Probability Theory; Risk; Aircraft Design; Spacecraft Design; Engine Design

20030001139 NASA Marshall Space Flight Center, Huntsville, AL USA
Space Launch Initiative: New Capabilities ... New Horizons
Dumbacher, Daniel L.; Nov. 15, 2002; In English; 4th European Conference on Hot Structures and Thermal Protection Systems for Space Vehicles, 25-29 Nov. 2002, Palermo, Italy; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents NASA's Space Launch Initiative (SLI) with new capabilities and new horizons. The topics include: 1) Integrated Space Transportation Plan; 2) SLI: The Work of an Nation; 3) SLI Goals and Status; 4) Composites and Materials; and 5) SLI & DoD/USAF Collaboration. This paper is presented in viewgraph form.
CASI
Spacecraft Launching; NASA Space Programs; Space Transportation System; Aerospace Engineering; Reusable Launch Vehicles

20030001115 NASA Ames Research Center, Moffett Field, CA USA
Automated CFD Parameter Studies on Distributed Parallel Computers
Rogers, Stuart E.; Aftosmis, Michael; Pandya, Shishir; Tejnil, Edward; Ahmad, Jasim; Kwak, Dochan, Technical Monitor; [2002]; In English; 16th AIAA Computational Fluid Dynamics Conference, 23-26 Jun. 2003, Orlando, FL, USA; No Copyright; Avail: CASI; A01, Hardcopy

The objective of the current work is to build a prototype software system which will automated the process of running CFD jobs on Information Power Grid (IPG) resources. This system should remove the need for user monitoring and intervention of every single CFD job. It should enable the use of many different computers to populate a massive run matrix in the shortest time possible. Such a software system has been developed, and is known as the AeroDB script system. The approach taken for the development of AeroDB was to build several discrete modules. These include a database, a job-launcher module, a run-manager module to monitor each individual job, and a web-based user portal for monitoring of the progress of the parameter study. The details of the design of AeroDB are presented in the following section. The following section provides the results of a parameter study which was performed using AeroDB for the analysis of a reusable launch vehicle (RLV). The paper concludes with a section on the lessons learned in this effort, and ideas for future work in this area.
Derived from text
Computational Fluid Dynamics; Parallel Computers; Automatic Control; Information Management; Distributed Processing

20030000980 NASA Marshall Space Flight Center, Huntsville, AL USA
Manufacturing Process Simulation of Large-Scale Cryotanks
Babai, Majid; Phillips, Steven; Griffin, Brian; Munafo, Paul M., Technical Monitor; [2002]; In English; AMPET, 16-18 Sep. 2002, Huntsville, AL, USA; No Copyright; Avail: Other Sources; Abstract Only

NASA's Space Launch Initiative (SLI) is an effort to research and develop the technologies needed to build a second-generation reusable launch vehicle. It is required that this new launch vehicle be 100 times safer and 10 times cheaper to operate than current launch vehicles. Part of the SLI includes the development of reusable composite and metallic cryotanks. The size of these reusable tanks is far greater than anything ever developed and exceeds the design limits of current manufacturing tools. Several design and manufacturing approaches have been formulated, but many factors must be weighed during the selection process. Among these factors are tooling reachability, cycle times, feasibility, and facility impacts. The manufacturing process simulation capabilities available at NASA's Marshall Space Flight Center have played a key role in down selecting between the various manufacturing approaches. By creating 3-D manufacturing process simulations, the varying approaches can be analyzed in a virtual world before any hardware or infrastructure is built. This analysis can detect and eliminate costly flaws in the various manufacturing approaches. The simulations check for collisions between devices, verify that design limits on joints are not exceeded, and provide cycle times which aid in the development of an optimized process flow. In addition, new ideas and concerns are often raised after seeing the visual representation of a manufacturing

process flow. The output of the manufacturing process simulations allows for cost and safety comparisons to be performed between the various manufacturing approaches. This output helps determine which manufacturing process options reach the safety and cost goals of the SLI.
Author
Reusable Launch Vehicles; Manufacturing; Feasibility; Cryogenic Fluid Storage; Fuel Tanks; Simulation

20030000832 NASA Langley Research Center, Hampton, VA USA
Managing MDO Software Development Projects
Townsend, J. C.; Salas, A. O.; [2002]; In English; 9th AIAA/ISSMO Symposium on Multidisciplinary Analysis and Optimization, 4-6 Sep. 2002, Atlanta, GA, USA
Report No.(s): AIAA Paper 2002-5442; Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

Over the past decade, the NASA Langley Research Center developed a series of ‘grand challenge’ applications demonstrating the use of parallel and distributed computation and multidisciplinary design optimization. All but the last of these applications were focused on the high-speed civil transport vehicle; the final application focused on reusable launch vehicles. Teams of discipline experts developed these multidisciplinary applications by integrating legacy engineering analysis codes. As teams became larger and the application development became more complex with increasing levels of fidelity and numbers of disciplines, the need for applying software engineering practices became evident. This paper briefly introduces the application projects and then describes the approaches taken in project management and software engineering for each project; lessons learned are highlighted.
Author
Multidisciplinary Design Optimization; Distributed Processing; Computer Networks; Software Engineering; Computer Aided Design; Applications Programs (Computers); Project Management; Parallel Computers

20030000831 NASA Langley Research Center, Hampton, VA USA
Discrete Roughness Effects on Shuttle Orbiter at Mach 6
Berry, Scott A.; Hamilton, H. Harris, II; [2002]; In English; 32nd AIAA Fluid Dynamics Conference, 24-27 Jun. 2002, Saint Louis, MO, USA; Original contains color illustrations
Report No.(s): AIAA Paper 2002-2744; Copyright; Avail: CASI; A03, Hardcopy; Distribution under U.S. Government purpose rights

Discrete roughness boundary layer transition results on a Shuttle Orbiter model in the NASA Langley Research Center 20-Inch Mach 6 Air Tunnel have been reanalyzed with new boundary layer calculations to provide consistency for comparison to other published results. The experimental results were previously obtained utilizing the phosphor thermography system to monitor the status of the boundary layer via global heat transfer images of the Orbiter windward surface. The size and location of discrete roughness elements were systematically varied along the centerline of the 0.0075-scale model at an angle of attack of 40 deg and the boundary layer response recorded. Various correlative approaches were attempted, with the roughness transition correlations based on edge properties providing the most reliable results. When a consistent computational method is used to compute edge conditions, transition datasets for different configurations at several angles of attack have been shown to collapse to a well-behaved correlation.
Author
Hypersonic Speed; Scale Models; Space Shuttle Orbiters; Reusable Launch Vehicles; Wind Tunnel Tests

20030000749 NASA Marshall Space Flight Center, Huntsville, AL USA
Thermal/Fluid Analysis of a Composite Heat Exchanger for Use on the RLV Rocket Engine
Nguyen, Dalton; Twelfth Thermal and Fluids Analysis Workshop; July 2002; In English; Original contains color illustrations; No Copyright; Avail: CASI; A02, Hardcopy

As part of efforts to design a regeneratively cooled composite nozzle ramp for use on the reusable vehicle (RLV) rocket engine, an C-SiC composites heat exchanger concept was proposed for thermal performance evaluation. To test the feasibility of the concept, sample heat exchanger panels were made to fit the Glenn Research Center’s cell 22 for testing. Operation of the heat exchanger was demonstrated in a combustion environment with high heat fluxes similar to the RLV Aerospike Ramp. Test measurements were reviewed and found to be valuable for the on going fluid and thermal analysis of the actual RLV composite ramp. Since the cooling fluid for the heat exchanger is water while the RLV Ramp cooling fluid is LH2, fluid and thermal models were constructed to correlate to the specific test set-up. The knowledge gained from this work will be helpful

for analyzing the thermal response of the actual RLV Composite Ramp. The coolant thermal properties for the models are taken from test data. The heat exchanger's cooling performance was analyzed using the Generalized Fluid System Simulation Program (GFSSP). Temperatures of the heat exchanger's structure were predicted in finite element models using Patran and Sinda. Results from the analytical models and the tests show that RSC's heat exchanger satisfied the combustion environments in a series of 16 tests.
Author
Thermal Analysis; Heat Exchangers; Reusable Launch Vehicles; Rocket Engines

20030000734 NASA Ames Research Center, Moffett Field, CA USA
Time-Dependent Simulations of Turbopump Flows
Kiris, Cetin; Kwak, Dochan; Chan, William; Williams, Robert; Twelfth Thermal and Fluids Analysis Workshop; July 2002; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

Unsteady flow simulations for RLV (Reusable Launch Vehicles) 2nd Generation baseline turbopump for one and half impeller rotations have been completed by using a 34.3 Million grid points model. MLP (Multi-Level Parallelism) shared memory parallelism has been implemented in INS3D, and benchmarked. Code optimization for cash based platforms will be completed by the end of September 2001. Moving boundary capability is obtained by using DCF module. Scripting capability from CAD (computer aided design) geometry to solution has been developed. Data compression is applied to reduce data size in post processing. Fluid/Structure coupling has been initiated.
Author
Turbine Pumps; Unsteady Flow; Computerized Simulation; Computational Grids; Computational Fluid Dynamics

20030000733 NASA Marshall Space Flight Center, Huntsville, AL USA
Stage Separation CFD Tool Development and Evaluation
Droege, Alan; Gomez, Reynaldo; Wang, Ten-See; Twelfth Thermal and Fluids Analysis Workshop; July 2002; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation evaluates CFD (Computational Fluid Dynamics) tools for solving stage separation problems. The demonstration and validation of the tools is for a second generation RLV (Reusable Launch Vehicle) stage separation. The flow solvers are: Cart3D; Overflow/Overflow-D; Unic.
CASI
Computational Fluid Dynamics; Stage Separation; Computer Programs

20030000479 NASA Marshall Space Flight Center, Huntsville, AL USA
Space Launch Initiative: New Capabilities - New Horizons
Dumbacher, Daniel; Smith, Dennis E., Technical Monitor; Sep. 10, 2002; In English; 10th Japan-US Conference on Composites Materials, 16-18 Sep. 2002, Stanford, CA, USA; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents NASA's Space Launch Initiative (SLI) with new capabilities and new horizons. The topics include: 1) Integrated Space Transportation Plan; 2) SLI: The Work of a Nation; 3) SLI Goals and Status; 4) Composites and Materials; and 5) SLI and DOD/USAF Collaboration. This paper is in viewgraph form.
CASI
Spacecraft Launching; Space Transportation System; NASA Programs; Reusable Launch Vehicles; Technology Utilization

20030000444 NASA Langley Research Center, Hampton, VA USA
Technology Roadmap for Dual-Mode Scramjet Propulsion to Support Space-Access Vision Vehicle Development
Cockrell, Charles E., Jr.; Auslender, Aaron H.; Guy, R. Wayne; McClinton, Charles R.; Welch, Sharon S.; [2002]; In English; 11th AIAA/AAAF International Space Planes and Hypersonic Systems, 24 Sep. - 4 Oct. 2002, Orleans, France
Report No.(s): AIAA Paper 2002-5188; No Copyright; Avail: CASI; A03, Hardcopy

Third-generation reusable launch vehicle (RLV) systems are envisioned that utilize airbreathing and combined-cycle propulsion to take advantage of potential performance benefits over conventional rocket propulsion and address goals of reducing the cost and enhancing the safety of systems to reach earth orbit. The dual-mode scramjet (DMSJ) forms the core of combined-cycle or combination-cycle propulsion systems for single-stage-to-orbit (SSTO) vehicles and provides most of the orbital ascent energy. These concepts are also relevant to two-stage-to-orbit (TSTO) systems with an airbreathing first or second stage. Foundation technology investments in scramjet propulsion are driven by the goal to develop efficient Mach 3-15

concepts with sufficient performance and operability to meet operational system goals. A brief historical review of NASA scramjet development is presented along with a summary of current technology efforts and a proposed roadmap. The technology addresses hydrogen-fueled combustor development, hypervelocity scramjets, multi-speed flowpath performance and operability, propulsion-airframe integration, and analysis and diagnostic tools.
Author
Reusable Launch Vehicles; Supersonic Combustion Ramjet Engines; Ascent Propulsion Systems; Systems Analysis; Propulsive Efficiency; Research and Development; Propulsion System Performance

20020094306 NASA Marshall Space Flight Center, Huntsville, AL USA
An Automated Method to Compute Orbital Re-entry Trajectories with Heating Constraints
Zimmerman, Curtis; Dukeman, Greg; Hanson, John; Fogle, Frank R., Technical Monitor; [2002]; In English; AIAA GNC Conference, 5-8 Aug. 2002, Monterey, CA, USA; No Copyright; Avail: CASI; A01, Hardcopy

Determining how to properly manipulate the controls of a re-entering re-usable launch vehicle (RLV) so that it is able to safely return to Earth and land involves the solution of a two-point boundary value problem (TPBVP). This problem, which can be quite difficult, is traditionally solved on the ground prior to flight. If necessary, a nearly unlimited amount of time is available to find the 'best' solution using a variety of trajectory design and optimization tools. The role of entry guidance during flight is to follow the pre- determined reference solution while correcting for any errors encountered along the way. This guidance method is both highly reliable and very efficient in terms of onboard computer resources. There is a growing interest in a style of entry guidance that places the responsibility of solving the TPBVP in the actual entry guidance flight software. Here there is very limited computer time. The powerful, but finicky, mathematical tools used by trajectory designers on the ground cannot in general be converted to do the job. Non-convergence or slow convergence can result in disaster. The challenges of designing such an algorithm are numerous and difficult. Yet the payoff (in the form of decreased operational costs and increased safety) can be substantiaL This paper presents an algorithm that incorporates features of both types of guidance strategies. It takes an initial RLV orbital re-entry state and finds a trajectory that will safely transport the vehicle to Earth. During actual flight, the computed trajectory is used as the reference to be flown by a more traditional guidance method.
Author
Reentry Trajectories; Trajectory Optimization; Reusable Launch Vehicles

20020092099 NASA Marshall Space Flight Center, Huntsville, AL USA
Life Cycle Systems Engineering Approach to NASA's 2nd Generation Reusable Launch Vehicle
Thomas, Dale; Smith, Charles; Safie, Fayssal; Kittredge, Sheryl; [2002]; In English; Workshop on Life Cycle Systems Engineering, 6-7 Nov. 2002, Redstone Arsenal, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

The overall goal of the 2nd Generation RLV Program is to substantially reduce technical and business risks associated with developing a new class of reusable launch vehicles. NASA's specific goals are to improve the safety of a 2nd- generation system by 2 orders of magnitude - equivalent to a crew risk of 1 -in- 10,000 missions - and decrease the cost tenfold, to approximately $1,000 per pound of payload launched. Architecture definition is being conducted in parallel with the maturating of key technologies specifically identified to improve safety and reliability, while reducing operational costs. An architecture broadly includes an Earth-to-orbit reusable launch vehicle, on-orbit transfer vehicles and upper stages, mission planning, ground and flight operations, and support infrastructure, both on the ground and in orbit. The systems engineering approach ensures that the technologies developed - such as lightweight structures, long-life rocket engines, reliable crew escape, and robust thermal protection systems - will synergistically integrate into the optimum vehicle. Given a candidate architecture that possesses credible physical processes and realistic technology assumptions, the next set of analyses address the system's functionality across the spread of operational scenarios characterized by the design reference missions. The safety/reliability and cost/economics associated with operating the system will also be modeled and analyzed to answer the questions 'How safe is it?' and 'How much will it cost to acquire and operate?' The systems engineering review process factors in comprehensive budget estimates, detailed project schedules, and business and performance plans, against the goals of safety, reliability, and cost, in addition to overall technical feasibility. This approach forms the basis for investment decisions in the 2nd Generation RLV Program's risk-reduction activities. Through this process, NASA will continually refine its specialized needs and identify where Defense and commercial requirements overlap those of civil missions.
Author
NASA Programs; Reusable Launch Vehicles; Systems Engineering; Life Cycle Costs; Technology Utilization

20020092015 Ohio Univ., OH USA

Direct Fault Tolerant RLV Altitude Control: A Singular Perturbation Approach

Zhu, J. J.; Lawrence, D. A.; Fisher, J.; Shtessel, Y. B.; Hodel, A. S.; Lu, P.; Jackson, Scott, Technical Monitor; Jan. 09, 2002; In English; 2002 AIAA GN and C Conference, 5-8 Aug. 2002, Monterey, CA, USA

Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A03, Hardcopy

In this paper, we present a direct fault tolerant control (DFTC) technique, where by 'direct' we mean that no explicit fault identification is used. The technique will be presented for the attitude controller (autopilot) for a reusable launch vehicle (RLV), although in principle it can be applied to many other applications. Any partial or complete failure of control actuators and effectors will be inferred from saturation of one or more commanded control signals generated by the controller. The saturation causes a reduction in the effective gain, or bandwidth of the feedback loop, which can be modeled as an increase in singular perturbation in the loop. In order to maintain stability, the bandwidth of the nominal (reduced-order) system will be reduced proportionally according to the singular perturbation theory. The presented DFTC technique automatically handles momentary saturations and integrator windup caused by excessive disturbances, guidance command or dispersions under normal vehicle conditions. For multi-input, multi-output (MIMO) systems with redundant control effectors, such as the RLV attitude control system, an algorithm is presented for determining the direction of bandwidth cutback using the method of minimum-time optimal control with constrained control in order to maintain the best performance that is possible with the reduced control authority. Other bandwidth cutback logic, such as one that preserves the commanded direction of the bandwidth or favors a preferred direction when the commanded direction cannot be achieved, is also discussed. In this extended abstract, a simplistic example is proved to demonstrate the idea. In the final paper, test results on the high fidelity 6-DOF X-33 model with severe dispersions will be presented.

Author

Actuators; Altitude Control; Attitude Control; Control Equipment; Failure; Fault Tolerance

20020092014 Ohio Univ., OH USA

Autocommander: A Supervisory Controller for Integrated Guidance and Control for the 2nd Generation Reusable Launch Vehicle

Fisher, J. E.; Lawrence, D. A.; Zhu, J. J.; Jackson, Scott, Technical Monitor; Jan. 09, 2002; In English; 2002 AIAA GN and C Conference, 5-8 Aug. 2002, Monterey, CA, USA

Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents a hierarchical architecture for integrated guidance and control that achieves risk and cost reduction for NASA's 2d generation reusable launch vehicle (RLV). Guidance, attitude control, and control allocation subsystems that heretofore operated independently will now work cooperatively under the coordination of a top-level autocommander. In addition to delivering improved performance from a flight mechanics perspective, the autocommander is intended to provide an autonomous supervisory control capability for traditional mission management under nominal conditions, G&C reconfiguration in response to effector saturation, and abort mode decision-making upon vehicle malfunction. This high-level functionality is to be implemented through the development of a relational database that is populated with the broad range of vehicle and mission specific data and translated into a discrete event system model for analysis, simulation, and onboard implementation. A Stateflow Autocoder software tool that translates the database into the Stateflow component of a Matlab/Simulink simulation is also presented.

Author

Attitude Control; Autocoders; Automatic Control; Mathematical Models; Systems Analysis; Architecture (Computers)

20020092011 Ohio Univ., Athens, OH USA

Reusable Launch Vehicle Attitude Control Using a Time-Varying Sliding Mode Control Technique

Shtessel, Yuri B.; Zhu, J. Jim; Daniels, Dan; Jackson, Scott, Technical Monitor; Jan. 09, 2002; In English; 2002 AIAA GN and C Conference, 5-8 Aug. 2002, Monterey, CA, USA

Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A03, Hardcopy

In this paper we present a time-varying sliding mode control (TVSMC) technique for reusable launch vehicle (RLV) attitude control in ascent and entry flight phases. In ascent flight the guidance commands Euler roll, pitch and yaw angles, and in entry flight it commands the aerodynamic angles of bank, attack and sideslip. The controller employs a body rate inner loop and the attitude outer loop, which are separated in time-scale by the singular perturbation principle. The novelty of the TVSMC is that both the sliding surface and the boundary layer dynamics can be varied in real time using the PD-eigenvalue assignment technique. This salient feature is used to cope with control command saturation and integrator windup in the presence of severe disturbance or control effector failure, which enhances the robustness and fault tolerance of the controller.

The TV-SMC ascent and descent designs are currently being tested with high fidelity, 6-DOF dispersion simulations. The test results will be presented in the final version of this paper.
Author
Attitude Control; Control Equipment; Fault Tolerance; Reusable Launch Vehicles; Robustness (Mathematics)

20020092010 NASA Marshall Space Flight Center, Huntsville, AL USA
A Quantitative Reliability, Maintainability and Supportability Approach for NASA's Second Generation Reusable Launch Vehicle
Safie, Fayssal M.; Daniel, Charles; Kalia, Prince; Smith, Charles A., Technical Monitor; [2002]; In English; Workshop on Life Cycle System Engineering, 6-7 Nov. 2002, Redstone Arsenal, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

The USA National Aeronautics and Space Administration (NASA) is in the midst of a 10-year Second Generation Reusable Launch Vehicle (RLV) program to improve its space transportation capabilities for both cargo and crewed missions. The objectives of the program are to: significantly increase safety and reliability, reduce the cost of accessing low-earth orbit, attempt to leverage commercial launch capabilities, and provide a growth path for manned space exploration. The safety, reliability and life cycle cost of the next generation vehicles are major concerns, and NASA aims to achieve orders of magnitude improvement in these areas. To get these significant improvements, requires a rigorous process that addresses Reliability, Maintainability and Supportability (RMS) and safety through all the phases of the life cycle of the program. This paper discusses the RMS process being implemented for the Second Generation RLV program.
Author
Reliability; Reusable Launch Vehicles; Safety; Maintainability; Life Cycle Costs; Life (Durability)

20020091983 NASA Glenn Research Center, Cleveland, OH USA
Overview of GRC's Advanced Sensor and Instrumentation Development
Mercer, Carolyn; 2001 NASA Seal/Secondary Air System Workshop; October 2002; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

Glenn Research Center develops advanced diagnostic techniques to measure surface and flow properties in research facilities. We support a variety of aerospace propulsion applications: Shuttle, X-33, X-43, ISS, and research engine components: inlets, compressors, combustors, nozzles. We are developing a suite of instrumentation specifically for 3rd Generation Reusable Launch Vehicle testing.
Derived from text
Sensors; Reusable Launch Vehicles; Measuring Instruments; Flow Characteristics; General Overviews; Test Facilities

20020091980 NASA Glenn Research Center, Cleveland, OH USA
Development and Capabilities of Unique Structural Seal Test Rigs
DeMange, Jeffrey J.; Dunlap, Patrick H., Jr.; Steinetz, Bruce M.; Breen, Daniel P.; Robbie, Malcolm G.; 2001 NASA Seal/Secondary Air System Workshop; October 2002; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

High temperature structural seals are necessary in many aerospace and aeronautical applications to minimize any detrimental effects originating from undesired leakage. The NASA Glenn Research Center has been and continues to be a pioneer in the development and evaluation of these types of seals. The current focus for the development of structural seals is for the 3rd Generation Reusable Launch Vehicle (RLV), which is scheduled to replace the current space shuttle system around 2025. Specific areas of development under this program include seals for propulsion systems (such as the hypersonic air-breathing ISTAR engine concept based upon Rocket Based Combined Cycle technology) and control surface seals for spacecraft including the autonomous rescue X-38 Crew Return Vehicle and the X-37 Space Maneuver Vehicle.
Derived from text
Seals (Stoppers); Engine Design; Air Breathing Engines; Spacecraft Propulsion; Aerospace Engineering; Reusable Launch Vehicles

20020091979 NASA Glenn Research Center, Cleveland, OH USA
Third Generation RLV Structural Seal Development Programs at NASA GRC
Dunlap, Patrick H., Jr.; Steinetz, Bruce M.; DeMange, Jeffrey J.; 2001 NASA Seal/Secondary Air System Workshop; October 2002; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

NASA GRC's work on high temperature structural seal development began in the late 1980's and early 1990's under the

NASP (National Aero-Space Plane) project. Bruce Steinetz led the in-house propulsion system seal development program and oversaw industry efforts for propulsion system and airframe seal development for this vehicle. a propulsion system seal location in the NASP engine is shown. The seals were located along the edge of a movable panel in the engine to seal the gap between the panel and adjacent engine sidewalls. More recently, we worked with Rocketdyne on high temperature seals for the linear aerospike engine ramps. In applications such as the former X-33 program, multiple aerospike engine modules would be installed side by side on the vehicle. Seals are required in between adjacent engine modules along the edges and base of the engines. The seals have to withstand the extreme temperatures produced byt he thrusters at the top of the ramps while accommodating large deflections between adjacent ramps. We came up with several promising seal concepts for this application and shared them with Rocketdyne.
Derived from text
National Aerospace Plane Program; X-33 Reusable Launch Vehicle; Seals (Stoppers); Systems Engineering; NASA Programs

20020091968 NASA Glenn Research Center, Cleveland, OH USA
NASA's Advanced Space Transportation Program: RTA Project Summary
Bartolotta, Paul; McNelis, Nancy; 2001 NASA Seal/Secondary Air System Workshop; October 2002; Volume 1; In English; Original contains color illustrations; No Copyright; Avail: CASI; A02, Hardcopy

This paper presents a general overview of NASA's Revolutionary Turbine Accelerator (RTA) Project. The contents include: 1) Why Turbine Accelerators; 2) Turbine Base Combination Cycle (TBCC) Interrelationships; 3) RTA Bridges the Gap Between Mach 3 and Mach 5; 4) TBCC and RTA; and 6) RTA Project Areas of Emphasis. A short summary of the TBCC and RTA project is also given. This paper is in viewgraph form.
CASI
Space Transportation; NASA Programs; Turbine Engines; Accelerators; Spacecraft Propulsion; Technology Utilization

20020091933 NASA Marshall Space Flight Center, Huntsville, AL USA, Science Applications International Corp., Huntsville, AL USA
Integrated Technology Assessment Center (ITAC) Update
Taylor, J. L.; Neely, M. A.; Curran, F. M.; Christensen, E. R.; Escher, D.; Lovell, N.; Morris, Charles, Technical Monitor; Jul. 10, 2002; In English; 38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 7-10 Jul. 2002, Indianapolis, IN, USA
Contract(s)/Grant(s): NASA Order H-32738-D
Report No.(s): AIAA Paper 2002-3550; Copyright; Avail: CASI; A02, Hardcopy; Distribution as joint owner in the copyright

The Integrated Technology Assessment Center (ITAC) has developed a flexible systems analysis framework to identify long-term technology needs, quantify payoffs for technology investments, and assess the progress of ASTP-sponsored technology programs in the hypersonics area. For this, ITAC has assembled an experienced team representing a broad sector of the aerospace community and developed a systematic assessment process complete with supporting tools. Concepts for transportation systems are selected based on relevance to the ASTP and integrated concept models (ICM) of these concepts are developed. Key technologies of interest are identified and projections are made of their characteristics with respect to their impacts on key aspects of the specific concepts of interest. Both the models and technology projections are then fed into the ITAC's probabilistic systems analysis framework in ModelCenter. This framework permits rapid sensitivity analysis, single point design assessment, and a full probabilistic assessment of each concept with respect to both embedded and enhancing technologies. Probabilistic outputs are weighed against metrics of interest to ASTP using a multivariate decision making process to provide inputs for technology prioritization within the ASTP. ITAC program is currently finishing the assessment of a two-stage-to-orbit (TSTO), rocket-based combined cycle (RBCC) concept and a TSTO turbine-based combined cycle (TBCC) concept developed by the team with inputs from NASA. A baseline all rocket TSTO concept is also being developed for comparison. Boeing has recently submitted a performance model for their Flexible Aerospace System Solution for Tomorrow (FASST) concept and the ISAT program will provide inputs for a single-stage-to-orbit (SSTO) TBCC based concept in the near-term. Both of these latter concepts will be analyzed within the ITAC framework over the summer. This paper provides a status update of the ITAC program.
Author
Technology Assessment; Systems Analysis; Systems Integration; Aerospace Systems; Hypersonics; Spacecraft Propulsion

20020091884 Ohio Univ., OH USA
On-Line Computation of a Local Attainable Moment Set for Reusable Launch Vehicles
Hodel, A. S.; Shtessel, Yuri B.; Jackson, Scott, Technical Monitor; Jan. 09, 2002; In English; 2002 AIAA GN and C Conference, 5-8 Aug. 2002, Monterey, CA, USA
Contract(s)/Grant(s): NAS8-01105; No Copyright; Avail: CASI; A01, Hardcopy

Traditional attitude control design of reusable launch vehicles involves independent design of autopilot and control allocation modules [SBB99], [SK97], [SHJ00], [PS00], [Hod00]. Unfortunately, this results in the potential for overly aggressive commands in the autopilot resulting in a loss of performance due to actuator saturation, particularly if the autopilot may suffer from integrator wind-up [HH01].
Author
Reusable Launch Vehicles; On-Line Systems; Modules; Attitude Control; Automatic Pilots

20020091874 NASA Marshall Space Flight Center, Huntsville, AL USA
Operations Analysis of the 2nd Generation Reusable Launch Vehicle
Noneman, Steven R.; Smith, C. A., Technical Monitor; [2002]; In English; Space Ops 2002, 9-12 Oct. 2002, Houston, TX, USA; No Copyright; Avail: CASI; A03, Hardcopy

The Space Launch Initiative (SLI) program is developing a second-generation reusable launch vehicle. The program goals include lowering the risk of loss of crew to 1 in 10,000 and reducing annual operations cost to one third of the cost of the Space Shuttle. The SLI missions include NASA, military and commercial satellite launches and crew and cargo launches to the space station. The SLI operations analyses provide an assessment of the operational support and infrastructure needed to operate candidate system architectures. Measures of the operability are estimated (i.e. system dependability, responsiveness, and efficiency). Operations analysis is used to determine the impact of specific technologies on operations. A conceptual path to reducing annual operations costs by two thirds is based on key design characteristics, such as reusability, and improved processes lowering labor costs. New operations risks can be expected to emerge. They can be mitigated with effective risk management with careful identification, assignment, tracking, and closure. SLI design characteristics such as nearly full reusability, high reliability, advanced automation, and lowered maintenance and servicing coupled with improved processes are contributors to operability and large operating cost reductions.
Author
Reusable Launch Vehicles; Spacecraft Launching; Design Analysis; Maintenance

20020090799 Boeing Phantom Works, Seattle, WA USA
Advanced High Temperature Structural Seals
Newquist, Charles W.; Verzemnieks, Juris; Keller, Peter C.; Rorabaugh, Michael; Shorey, Mark; October 2002; In English; Original contains color illustrations
Contract(s)/Grant(s): F33615-95-D-3203; RTOP 706-85-33; AF Proj. 0014
Report No.(s): NASA/CR-2002-211973; E-13617; NAS 1.26:211973; No Copyright; Avail: CASI; A06, Hardcopy

This program addresses the development of high temperature structural seals for control surfaces for a new generation of small reusable launch vehicles. Successful development will contribute significantly to the mission goal of reducing launch cost for small, 200 to 300 pound payloads. Development of high temperature seals is mission enabling. For instance, ineffective control surface seals can result in high temperature (3100 F) flows in the elevon area exceeding structural material limits. Longer sealing life will allow use for many missions before replacement, contributing to the reduction of hardware, operation and launch costs.
Author
Reusable Launch Vehicles; Seals (Stoppers); Research and Development; Elevons; Life (Durability); Product Development; Design Analysis; Thermal Analysis

20020085363 NASA Marshall Space Flight Center, Huntsville, AL USA
Main Engine Prototype Development for 2nd Generation RLV RS-83
Vilja, John; Fisher, Mark; Lyles, Garry M., Technical Monitor; [2002]; In English; 1st AIAA/IAF Symposium on Future Reusable Launch Vehicles, 12 Apr. 2002, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This presentation reports on the NASA project to develop a prototype for RS-83 engine designed for use on reusable launch vehicles (RLV). Topics covered include: program objectives, overview schedule, organizational chart, integrated systems engineering processes, requirement analysis, catastrophic engine loss, maintainability analysis tools, and prototype design analysis.
CASI
Design Analysis; Engine Design; Reusable Launch Vehicles; Prototypes; Research and Development

20020085342 NASA Marshall Space Flight Center, Huntsville, AL USA
On-Orbit Propulsion System Project Overview
Champion, Robert H.; Lyles, Garry M., Technical Monitor; [2002]; In English; 1st AIAA/IAF Symposium on Future Reusable Launch Vehicles, 11-12 Apr. 2002, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This conference presentation reports on the progress on NASA's On-Orbit Propulsion System Project which aims to support the development of second generation reusable launch vehicles (RLV) through advanced research and development and risk reduction activities. Topics covered include: project goals, project accomplishments, risk reduction activities, thruster design and development initiatives, and Aerojet LOX/Ethanol engine development and testing.
CASI
Reusable Launch Vehicles; Spacecraft Propulsion; Research and Development; Risk

20020082904 NASA Glenn Research Center, Cleveland, OH USA
GTX Reference Vehicle Structural Verification Methods and Weight Summary
Hunter, J. E.; McCurdy, D. R.; Dunn, P. W.; September 2002; In English; Combustion, Airbreathing Propulsion, Propulsion Systems Hazards and Modelling and Simulation Joint Meeting, 8-12 Apr. 2002, Destin, FL, USA
Contract(s)/Grant(s): RTOP 708-90-63
Report No.(s): NASA/TM-2002-211884; E-13200-1; NAS 1.15:211884; No Copyright; Avail: CASI; A03, Hardcopy

The design of a single-stage-to-orbit air breathing propulsion system requires the simultaneous development of a reference launch vehicle in order to achieve the optimal mission performance. Accordingly, for the GTX study a 300-lb payload reference vehicle was preliminarily sized to a gross liftoff weight (GLOW) of 238,000 lb. A finite element model of the integrated vehicle/propulsion system was subjected to the trajectory environment and subsequently optimized for structural efficiency. This study involved the development of aerodynamic loads mapped to finite element models of the integrated system in order to assess vehicle margins of safety. Commercially available analysis codes were used in the process along with some internally developed spreadsheets and FORTRAN codes specific to the GTX geometry for mapping of thermal and pressure loads. A mass fraction of 0.20 for the integrated system dry weight has been the driver for a vehicle design consisting of state-of-the-art composite materials in order to meet the rigid weight requirements. This paper summarizes the methodology used for preliminary analyses and presents the current status of the weight optimization for the structural components of the integrated system.
Author
Hypersonic Aircraft; Reusable Launch Vehicles; Structural Design; Design Analysis; Air Breathing Engines

20020080893 Ohio Aerospace Inst., Brook Park, OH USA
Design and Analysis of UHTC Leading Edge Attachment
Thomas, David J.; Nemeth, Noel N., Technical Monitor; July 2002; In English; Original contains color illustrations
Contract(s)/Grant(s): NCC3-756; RTOP 706-85-31
Report No.(s): NASA/CR-2002-211505; E-13271; NAS 1.26:211505; No Copyright; Avail: CASI; A04, Hardcopy

NASA Glenn Research Center was contacted to provide technical support to NASA Ames Research Center in the design and analysis of an ultra high temperature ceramic (UHTC) leading edge. UHTC materials are being considered for reusable launch vehicles because their high temperature capability may allow for un-cooled sharp leading edge designs. While ceramic materials have the design benefit of allowing subcomponents to run hot, they also provide a design challenge in that they invariably must be in contact with cooler subcomponents elsewhere in the structure. NASA Glenn Research Center proposed a modification to an existing attachment design. Thermal and structural analyses of the leading edge assembly were carried out using ABAQUS finite element software. Final results showed that the proposed modifications aided in thermally isolating hot and cold subcomponents and reducing bearing stresses at the attachment location.
Author
Ceramics; Design Analysis; High Temperature; Mathematical Models; Sharp Leading Edges

20020077952 NASA Langley Research Center, Hampton, VA USA
A Procedure for Structural Weight Estimation of Single Stage to Orbit Launch Vehicles (Interim User's Manual)
Martinovic, Zoran N.; Cerro, Jeffrey A.; September 2002; In English
Contract(s)/Grant(s): RTOP 706-88-21-03
Report No.(s): NASA/TM-2002-211931; L-18169; NAS 1.15:211931; No Copyright; Avail: CASI; A04, Hardcopy

This is an interim user's manual for current procedures used in the Vehicle Analysis Branch at NASA Langley Research

Center, Hampton, Virginia, for launch vehicle structural subsystem weight estimation based on finite element modeling and structural analysis. The process is intended to complement traditional methods of conceptual and early preliminary structural design such as the application of empirical weight estimation or application of classical engineering design equations and criteria on one dimensional 'line' models. Functions of two commercially available software codes are coupled together. Vehicle modeling and analysis are done using SDRC/I-DEAS, and structural sizing is performed with the Collier Research Corp. HyperSizer program.
Author
Structural Weight; Single Stage to Orbit Vehicles; Mathematical Models; User Manuals (Computer Programs); Launch Vehicles

20020074785 NASA Glenn Research Center, Cleveland, OH USA
Characterization of C/SiC Ceramic Matrix Composites (CMCs) with Novel Interface Fiber Coatings
Petko, Jeanne F.; Kiser, J. Douglas; Gray, Hugh R., Technical Monitor; [2002]; In English; National Space and Missile Materials Symposium (NSMMS), 24-28 Jun. 2002, Colorado Springs, CO, USA; Original contains color illustrations
Contract(s)/Grant(s): RTOP 708-73-25; No Copyright; Avail: CASI; A03, Hardcopy

Ceramic Matrix Composites (CMCs) are attractive candidate aerospace materials due to their high specific strength, low density and high temperature capabilities. The National Aeronautics and Space Administration (NASA) is pursuing the use of CMC components in advanced Reusable Launch Vehicle (RLV) propulsion applications. Carbon fiber-reinforced silicon carbide (C/SiC) is the primary material of interest for a variety of RLV propulsion applications. These composites consist of high-strength carbon fibers and a high modulus, oxidation resistant matrix. For RLV propulsion applications, environmental durability will be critical. Two types of carbon fibers were processed with both standard (pyrolytic carbon) and novel (multilayer and pseudoporous) types of interface coatings as part of a study investigating various combinations of constituents. The benefit of protecting the composites with a surface sealant was also investigated. The strengths, durability in oxidizing environments, and microstructures of these developmental composite materials are presented. The novel interface coatings and the surface sealant show promise for protecting the carbon fibers from the oxidizing environment.
Author
Carbon Fibers; Ceramic Matrix Composites; Silicon Carbides; Coatings; Mechanical Properties; Microstructure

20020073804 NASA Ames Research Center, Moffett Field, CA USA
A Geometry-Centered Approach For Two-Stage-To-Orbit Vehicle Synthesis
Papadopoulos, P. E.; Reuther, J.; Haimes, B.; Harris, S.; 2nd JANNAF Modeling and Simulation Subcommittee Meeting; April 2002; Volume 1; In English
Contract(s)/Grant(s): NAS2-99092; Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

A geometry-centered approach to vehicle analysis and design is demonstrated for classes of two-stage-to-orbit (TSTO) launch configurations. The core capability unique to the new environment is its reliance on a CAD-neutral geometry application program interface. The environment permits a flexible vehicle closure process that includes (a) geometry manipulation, (b) propulsion system selection, (c) weights and sizing, (d) aerodynamic and aero-thermodynamic analysis, and (e) trajectory optimization. Conceptual level analysis tools are integrated into an automated framework using the Phoenix Integration ModelCenter software. The actual vehicle closure is highly iterative with several levels of interface between disciplines. Fundamental to these loops is the geometric manipulation. To provide robust yet accurate and automated geometry, software libraries have been created that allow bidirectional access to and control of a choice of user-chosen CAD kernels. The geometry libraries contain utilities that assemble a vehicle configuration according to prescribed parameter specifications, extract needed geometry parameters, and provide user-controllable water-tight surface tessellations of selected vehicle components. These operations are performed via commands given to the CAD kernel API. The CAPRI programming interface used to broker all the CAD API calls allows the higher-level vehicle assembly and parameter extraction utilities to remain CAD-neutral. The CAPRI interface also provides a consistent (CAD system independent) algorithm for surface tessellations. The geometry-centered philosophy in this analysis and design environment bypasses issues currently associated with CAD output translation, geometry manipulation and grid generation.
Author
Computer Aided Design; Computer Programs; Aerodynamic Configurations

20020073802 SpaceWorks Engineering, Inc., Atlanta, GA USA

System Level Uncertainty Assessment for Collaborative RLV Design

Charania, A. C.; Bradford, John E.; Olds, John R.; Graham, Matthew; 2nd JANNAF Modeling and Simulation Subcommittee Meeting; April 2002; Volume 1; In English; Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

A collaborative design process utilizing Probabilistic Data Assessment (PDA) is showcased. Given the limitation of financial resources by both the government and industry, strategic decision makers need more than just traditional point designs, they need to be aware of the likelihood of these future designs to meet their objectives. This uncertainty, an ever-present character in the design process, can be embraced through a probabilistic design environment. A conceptual design process is presented that encapsulates the major engineering disciplines for a Third Generation Reusable Launch Vehicle (RLV). Toolsets consist of aerospace industry standard tools in disciplines such as trajectory, propulsion, mass properties, cost, operations, safety, and economics. Variations of the design process are presented that use different fidelities of tools. The disciplinary engineering models are used in a collaborative engineering framework utilizing Phoenix Integration's ModelCenter and AnalysisServer environment. These tools allow the designer to join disparate models and simulations together in a unified environment wherein each discipline can interact with any other discipline. The design process also uses probabilistic methods to generate the system level output metrics of interest for a RLV conceptual design. The specific system being examined is the Advanced Concept Rocket Engine 92 (ACRE-92) RLV. Previous experience and knowledge (in terms of input uncertainty distributions from experts and modeling and simulation codes) can be coupled with Monte Carlo processes to best predict the chances of program success.

Author

Computerized Simulation; Probability Theory; Assessments; Design Analysis

20020073535 NASA Ames Research Center, Moffett Field, CA USA

Active Wireless Temperature Sensors for Aerospace Thermal Protection Systems

Milos, Frank S.; Karunaratne, K.; Arnold, Jim, Technical Monitor; Jul. 10, 2002; In English; SPIE 8th Annual International Symposium on NDE for Health Monitoring and Diagnostics, 2-6 Mar. 2003, San Diego, CA, USA; No Copyright; Avail: Other Sources; Abstract Only

Health diagnostics is an area where major improvements have been identified for potential implementation into the design of new reusable launch vehicles in order to reduce life-cycle costs, to increase safety margins, and to improve mission reliability. NASA Ames is leading the effort to advance inspection and health management technologies for thermal protection systems. This paper summarizes a joint project between NASA Ames and Korteks to develop active wireless sensors that can be embedded in the thermal protection system to monitor sub-surface temperature histories. These devices are thermocouples integrated with radio-frequency identification circuitry to enable acquisition and non-contact communication of temperature data through aerospace thermal protection materials. Two generations of prototype sensors are discussed. The advanced prototype collects data from three type-k thermocouples attached to a 2.54-cm square integrated circuit.

Author

Systems Health Monitoring; Temperature Sensors; Temperature Measurement

20020073119 NASA Glenn Research Center, Cleveland, OH USA

Affordable Flight Demonstration of the GTX Air-Breathing SSTO Vehicle Concept

Krivanek, Thomas M.; Roche, Joseph M.; Riehl, John P.; Kosareo, Daniel N.; 26th JANNAF Airbreathing Propulsion Subcommittee Meeting; April 2002; Volume 1; In English; No Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

The rocket based combined cycle (RBCC) powered single-stage-to-orbit (SSTO) reusable launch vehicle has the potential to significantly reduce the total cost per pound for orbital payload missions. To validate overall system performance, a flight demonstration must be performed. This paper presents an overview of the first phase of a flight demonstration program for the GTX SSTO vehicle concept. Phase 1 will validate the propulsion performance of the vehicle configuration over the supersonic and hypersonic airbreathing portions of the trajectory. The focus and goal of Phase 1 is to demonstrate the integration and performance of the propulsion system flowpath with the vehicle aerodynamics over the air-breathing trajectory. This demonstrator vehicle will have dual mode ramjet/scramjets, which include the inlet, combustor, and nozzle with geometrically scaled aerodynamic surface outer mold lines (OML) defining the forebody, boundary layer diverter, wings, and tail. The primary objective of this study is to demonstrate propulsion system performance and operability including the ram to scram transition, as well as to validate vehicle aerodynamics and propulsion airframe integration. To minimize overall risk and development cost the effort will incorporate proven materials, use existing turbomachinery in the propellant delivery

systems, launch from an existing unmanned remote launch facility, and use basic vehicle recovery techniques to minimize control and landing requirements. A second phase would demonstrate propulsion performance across all critical portions of a space launch trajectory (lift off through transition to all-rocket) integrated with flight-like vehicle systems.
Author
Air Breathing Engines; Cost Reduction; Engine Airframe Integration; Propulsion System Configurations; Propulsion System Performance; Reusable Launch Vehicles

20020073118 NASA Glenn Research Center, Cleveland, OH USA
SRM-Assisted Trajectory for the GTX Reference Vehicle
Riehl, John; Trefny, Charles; Kosareo, Daniel; 26th JANNAF Airbreathing Propulsion Subcommittee Meeting; April 2002; Volume 1; In English; No Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

A goal of the GTX effort has been to demonstrate the feasibility of a single stage- to- orbit (SSTO) vehicle that delivers a small payload to low earth orbit. The small payload class was chosen in order to minimize the risk and cost of development of this revolutionary system. A preliminary design study by the GTX team has resulted in the current configuration that offers considerable promise for meeting the stated goal. The size and gross lift-off weight resulting from scaling the current design to closure however may be considered impractical for the small payload. In lieu of evolving the project's reference vehicle to a large-payload class, this paper offers the alternative of using solid-rocket motors in order to close the vehicle at a practical scale. This approach offers a near-term, quasi-reusable system that easily evolves to reusable SSTO following subsequent development and optimization. This paper presents an overview of the impact of the addition of SRM's to the GTX reference vehicle's performance and trajectory. The overall methods of vehicle modeling and trajectory optimization will also be presented. A key element in the trajectory optimization is the use of the program OTIS 3.10 that provides rapid convergence and a great deal of flexibility to the user. This paper will also present the methods used to implement GTX requirements into OTIS modeling.
Author
Solid Propellant Rocket Engines; Single Stage to Orbit Vehicles; Low Earth Orbits

20020073112 NASA Ames Research Center, Moffett Field, CA USA
Cost Per Pound From Orbit
Merriam, M. L.; 26th JANNAF Airbreathing Propulsion Subcommittee Meeting; April 2002; Volume 1; In English; No Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

Traditional studies of Reusable Launch Vehicle (RLV) designs have focused on designs that are completely reusable except for the fuel. This may not be realistic with current technology . An alternate approach is to look at partially reusable launch vehicles. This raises the question of which parts should be reused and which parts should be expendable. One approach is to consider the cost/pound of returning these parts from orbit. With the shuttle, this cost is about three times the cost/pound of launching payload into orbit. A subtle corollary is that RLVs are much less practical for higher orbits, such as the one on which the International Space Station resides, than they are for low earth orbits.
Author
Reusable Launch Vehicles; Costs; Design Optimization

20020073101 NASA Marshall Space Flight Center, Huntsville, AL USA
NASA's Hypersonic Investment Area
Hueter, Uwe; Hutt, John; McClinton, Charles; 26th JANNAF Airbreathing Propulsion Subcommittee Meeting; April 2002; Volume 1; In English; No Copyright; Avail: CPIA, 10630 Little Patuxent Pkwy., Suite 202, Columbia, MD 21044-3320

NASA has established long term goals for access to space. The third generation launch systems are to be fully reusable and operational around 2025. The goal for third-generation launch systems represents significant reduction in cost and improved safety over the current first generation system. The Advanced Space Transportation Office (ASTP) at NASA s Marshall Space Flight Center (MSFC) has the agency lead to develop space transportation technologies. Within ASTP, under the Hypersonic Investment Area (HIA), third generation technologies are being pursued in the areas of propulsion, airframe, integrated vehicle health management (IVHM), avionics, power, operations and system analysis. These technologies are being matured through research and both ground and flight-testing. This paper provides an overview of the HIA program plans and recent accomplishments.
Author
Flight Tests; Hypersonics; Systems Analysis; Space Transportation; Safety; Reusable Launch Vehicles

20020072989 NASA Glenn Research Center, Cleveland, OH USA

Oxidation Behavior of Copper Alloy Candidates for Rocket Engine Applications (Technical Poster)

Ogbuji, Linus U. J.; Humphrey, Donald H.; Barrett, Charles A.; Greenbauer-Seng, Leslie, Technical Monitor; Gray, Hugh R., Technical Monitor; [2002]; In English; TMS IV Meeting, 6-10 Oct. 2002, Columbus, OH, USA; Original contains color illustrations

Contract(s)/Grant(s): NAS3-98008; RTOP 721-20-XX; No Copyright; Avail: CASI; A01, Hardcopy

A rocket engine's combustion chamber is lined with material that is highly conductive to heat in order to dissipate the huge thermal load (evident in a white-hot exhaust plume). Because of its thermal conductivity copper is the best choice of liner material. However, the mechanical properties of pure copper are inadequate to withstand the high stresses, hence, copper alloys are needed in this application. But copper and its alloys are prone to oxidation and related damage, especially 'blanching' (an oxidation-reduction mode of degradation). The space shuttle main engine combustion chamber is lined with a Cu-Ag-Zr alloy, 'NARloy-Z', which exhibits blanching. A superior liner is being sought for the next generation of RLVs (Reusable Launch Vehicles) It should have improved mechanical properties and higher resistance to oxidation and blanching, but without substantial penalty in thermal conductivity. GRCop84, a Cu-8Cr-4Nb alloy (Cr2Nb in Cu matrix), developed by NASA Glenn Research Center (GRC) and Case Western Reserve University, is a prime contender for RLV liner material. In this study, the oxidation resistance of GRCop-84 and other related/candidate copper alloys are investigated and compared

Author

Rocket Engines; Oxidation; Copper Alloys; Oxidation Resistance

20020072848 NASA Glenn Research Center, Cleveland, OH USA

Characterization of C/SiC Ceramic Matrix Composites (CMCs) with Novel Interface Fiber Coatings

Petko, Jeanne F.; Kiser, James Douglas; Verilli, Mike; McCue, Terry; Gray, Hugh R., Technical Monitor; [2002]; In English; 26th Annual Conference on Materials and Structures, 28-31 Jan. 2002, Cape Canaveral, FL, USA

Contract(s)/Grant(s): RTOP 708-73-25; No Copyright; Avail: CASI; A03, Hardcopy

Ceramic Matrix Composites (CMCs) are attractive candidate materials in the aerospace industry due to their high specific strength, low density and higher temperature capabilities. The National Aeronautics and Space Administration (NASA) is pursuing the use of CMC components in advanced Reusable Launch Vehicle (RLV) propulsion applications. Carbon fiber-reinforced silicon carbide (C/SiQ is the primary material of interest for a variety of RLV propulsion applications. These composites offer high-strength carbon fibers and a high modulus, oxidation-resistant matrix. For comparison, two types of carbon fibers were processed with novel types of interface coatings (multilayer and pseudoporous). For RLV propulsion applications, environmental durability will be critical. The coatings show promise of protecting the carbon fibers from the oxidizing environment. The strengths and microstructures of these composite materials are presented.

Author

Ceramic Matrix Composites; Fiber-Matrix Interfaces; Carbon Fibers; Silicon Carbides; Reusable Launch Vehicles; High Strength

20020070918 NASA Glenn Research Center, Cleveland, OH USA

SRM-Assisted Trajectory for the GTX Reference Vehicle

Riehl, John; Trefny, Charles; Kosareo, Daniel, Technical Monitor; June 2002; In e; Combustion, Airbreathing Propulsion, Propulsion Systems Hazards, and Modelling and Simulation Subcommittees Joint Meeting, 8-12 Apr. 2002, Destin, FL, USA

Contract(s)/Grant(s): RTOP 708-90-63

Report No.(s): NASA/TM-2002-211599; E-13386; NAS 1.15:211599; No Copyright; Avail: CASI; A03, Hardcopy

A goal of the GTX effort has been to demonstrate the feasibility of a single stage-to-orbit (SSTO) vehicle that delivers a small payload to low earth orbit. The small payload class was chosen in order to minimize the risk and cost of development of this revolutionary system. A preliminary design study by the GTX team has resulted in the current configuration that offers considerable promise for meeting the stated goal. The size and gross lift-off weight resulting from scaling the current design to closure however may be considered impractical for the small payload. In lieu of evolving the project' reference vehicle to a large-payload class, this paper offers the alternative of using solid-rocket motors in order to close the vehicle at a practical scale. This approach offers a near-term, quasi-reusable system that easily evolves to reusable SSTO following subsequent development and optimization. This paper presents an overview of the impact of the addition of SRM's to the GTX reference vehicle#s performance and trajectory. The overall methods of vehicle modeling and trajectory optimization will also be presented. A key element in the trajectory optimization is the use of the program OTIS 3.10 that provides rapid convergence

and a great deal of flexibility to the user. This paper will also present the methods used to implement GTX requirements into OTIS modeling.
Author
Solid Propellant Rocket Engines; Launch Vehicles; Hypersonic Combustion; Trajectory Optimization

20020070891 NASA Glenn Research Center, Cleveland, OH USA
Fuel Cell Activities at the NASA Glenn Research Center
Kohout, Lisa L.; Lyons, Valerie, Technical Monitor; [2002]; In English; Ohio Technical College, 10 Jul. 2002, Cleveland, OH, USA; No Copyright; Avail: CASI; A03, Hardcopy

Fuel cells have a long history in space applications and may have potential application in aeronautics as well. A fuel cell is an electrochemical energy conversion device that directly transforms the chemical energy of a fuel and oxidant into electrical energy. Alkaline fuel cells have been the mainstay of the U.S. space program, providing power for the Apollo missions and the Space Shuttle. However, Proton Exchange Membrane (PEM) fuel cells offer potential benefits over alkaline systems and are currently under development for the next generation Reusable Launch Vehicle (RLV). Furthermore, primary and regenerative systems utilizing PEM technology are also being considered for future space applications such as surface power and planetary aircraft. In addition to these applications, the NASA Glenn Research Center is currently studying the feasibility of the use of both PEM and solid oxide fuel cells for low- or zero-emission electric aircraft propulsion. These types of systems have potential applications for high altitude environmental aircraft, general aviation and commercial aircraft, and high attitude airships. NASA Glenn has a unique set of capabilities and expertise essential to the successful development of advanced fuel cell power systems for space and aeronautics applications. NASA Glenn's role in past fuel cell development programs as well as current activities to meet these new challenges will be presented
Author
Aircraft Engines; Fuel Cells; Research and Development; Spacecraft Power Supplies; Energy Conversion Efficiency; Aerospace Systems

20020070797 Mississippi Univ., University, MS USA
NASA EPSCoR Preparation Grant
Sukanek, Peter C.; [2002]; In English; American Society of Photogrammetry and Remote Sensing, May 2002, Washington, DC, USA
Contract(s)/Grant(s): NCC5-405; No Copyright; Avail: CASI; A02, Hardcopy

The NASA EPSCoR project in Mississippi involved investigations into three areas of interest to NASA by researchers at the four comprehensive universities in the state. These areas involved: (1) Noninvasive Flow Measurement Techniques, (2) Spectroscopic Exhaust Plume Measurements of Hydrocarbon Fueled Rocket Engines and (3) Integration of Remote Sensing and GIS data for Flood Forecasting on the Mississippi Gulf Coast. Each study supported a need at the Stennis Space Center in Mississippi. The first two addressed needs in rocket testing, and the third, in commercial remote sensing. Students from three of the institutions worked with researchers at Stennis Space Center on the projects.
Author
Flow Measurement; Computational Fluid Dynamics; Reusable Launch Vehicles; Systems Health Monitoring; Exhaust Flow Simulation; Flame Spectroscopy; Flood Predictions; Hydrology Models

20020070788 Georgia Inst. of Tech., Atlanta, GA USA
Permeability of Impacted Coated Composite Laminates
Johnson, W. S.; Findley, Benjamin; August 2002; In English
Contract(s)/Grant(s): NAG8-1819; GIT Proj. 1806675
Report No.(s): E-18-675; No Copyright; Avail: CASI; A07, Hardcopy

Composite materials are being considered for use on future generations of Reusable Launch Vehicles (RLVs) for both fuel tanks and fuel feedlines. Through the use of composite materials NASA can reduce the overall weight of the vehicle dramatically. This weight savings can then be translated into an increase in the weight of payload sent into orbit, reducing the cost per pound of payload. It is estimated that by switching to composite materials for fuel tanks the weight of the tanks can be reduced by 40 percent, which translates to a total vehicle weight savings of 14 percent. In this research, carbon/epoxy composites were studied for fuel feedline applications. There are concerns about using composite materials for feedlines and fuel tanks because these materials are extremely vulnerable to impact in the form of inadvertent bumping or dropped tools both during installation and maintenance. Additionally, it has been found that some of the sample feedlines constructed have had

leaks, and thus there may be a need to seal preexisting leaks in the composite prior to usage.
Derived from text
Composite Materials; Carbon Fibers; Coatings; Epoxy Matrix Composites; Fiber Composites

20020070521 Wyoming Univ., Laramie, WY USA
Reduction of Base Drag on Launch Vehicles
Naughton, Jonathan W.; [2002]; In English
Contract(s)/Grant(s): NAG4-208
Report No.(s): UWAL-2002-02; No Copyright; Avail: CASI; A03, Hardcopy

Current reentry vehicle designs exhibit large amounts of base drag due to large base areas. These large base areas can arise from the integration of the propulsion system (X-33) or control surface placement (X-38). Large base drag limits the vehicle's cross-range capability and causes a large glide-slope angle. Fortunately, there appears to be a possible means of lowering the base drag on these vehicles. Based on early work on the subsonic aerodynamics of lifting bodies, it appears that the addition of small amounts of viscous fore-body drag can produce a significant reduction in base drag. Recent work suggests that this phenomenon also occurs in the transonic and supersonic flight regimes. This report summarizes a study designed to demonstrate the reduction of base drag through the addition of fore-body viscous drag. The present study has focused on the measurement of viscous fore-body drag and the demonstration of the relationship between fore-body viscous drag and base drag at Reynolds Numbers up to 2.5 x 10(exp 6). The results of the present work do not conclusively demonstrate that viscous fore-body drag reduces base drag. The apparent contradictory results of the present study are attributed to the different geometry used in the present study. However, the results suggest that the increased boundary layer thickness at separation caused by larger fore-body viscous drag somehow affects the vortex structure in the wake thereby reducing the base drag. More research is required to confirm this postulated mechanism.
Author
Base Flow; Drag Reduction; Reentry Vehicles; Viscous Drag; Aerodynamic Drag; Forebodies

20020068832 Louisiana State Univ., Shreveport, LA USA
Operations Analysis of Space Shuttle System
Sarker, Bhaba R.; Research Reports: 2001 NASA/ASEE Summer Faculty Fellowship Program; July 2002; In English
Contract(s)/Grant(s): NAG8-1786; No Copyright; Avail: CASI; A01, Hardcopy

The space science program at NASA since 1950's has gone through stages of development and implementation in rocketry and scientific advancement. It has become the nation's largest scientific institution of research and innovation for space exploration and military research, and overall a pride of the nation at an exorbitant cost. After placing man on moon, space shuttle program at NASA is an on-going project with a high success rate at an average cost of about $450m per flight. A future endeavor from both government and private sectors needs to be undertaken for commercialization of this expensive mission. In order to attract private enterprises, it needs to boost up its operations with better technology at lower cost to cope with rapid changes in scientific advancement and economic competition. Thus, a second-generation reusable launch vehicle (2GRLV) will play a major role in future operation of NASA centers. Envisioning this potential way of saving the program by reducing the cost, NASA is currently managing an innovative program called the Space Launch Initiative (SLI) to develop key technologies that will support the development of second-generation reusable launch vehicles (RLV) which will be more economical and safer and reliable than the existing space shuttle system. The selection of which technologies to fund for further development is being based on their likelihood to contribute to providing cost reduction or safety improvements. It is envisioned that in the 2006 timeframe, NASA will make a decision as to whether or not to commit to the replacement of the current space shuttle system with a new RLV. The decision to proceed with a new RLV will be partly based on the likelihood that the new system will be better than the existing space shuttle. Government and private entrepreneurs are currently considering four different types of RLV projects: commercial programs, government programs, international concepts, and X Prize competitors. NASA has already commissioned a series of X-programs to study the future RLV program. Today, NASA decision makers need analytical tools to help determine which technologies to fund the development of this technology. In the 2006 timeframe, these same decision makers will need analytical tools to evaluate and compare various RLV architectures, including the existing space shuttle so as to make the best decision for whether or not to proceed with the development of a new RLV, and if so, then which one. A study is conducted here to establish ground level knowledge from the historical data and expertise experiences of the field personnel. Such information is compiled in the form of mission statement, goals, space shuttle operations, payloads and cargo constraints, resource constraints, and bottlenecks to the enhancement vector.
Derived from text
Reusable Launch Vehicles; Research and Development; Space Shuttles; Assessments; Project Planning; Operations Research

20020068823 College of New Jersey, Pomona, NJ USA
An Overview of PRA with Applications to Aerospace Systems
Navard, Sharon E.; Research Reports: 2001 NASA/ASEE Summer Faculty Fellowship Program; July 2002; In English; No Copyright; Avail: CASI; A01, Hardcopy

Probabilistic Risk Assessment (PRA) is a systematic process for evaluating the probabilities and consequences of undesirable events that can occur in a process or system along with providing a measure of the uncertainty associated with these probability estimates. In the past it was looked at with suspicion by many at NASA, perhaps because of bad experiences with unsuccessful quantitative methods during the Apollo era, but since the Challenger accident NASA has mandated that it be used, and it has been very successful. With NASA's new 'faster, better, cheaper' philosophy, it is vital that a tool be in place that can help to achieve these goals in the reliability area. This paper describes the history of PRA, gives examples of its aerospace applications to date, and gives suggestions for how it can be used in the future, both for space shuttle upgrades and for totally new technologies such as the Second Generation Reusable Launch Vehicle.
Author
Assessments; Probability Theory; Risk

20020068806 Alabama Agricultural and Mechanical Univ., Normal, AL USA
Second Generation Reusable Launch Vehicle Development and Global Competitiveness of US Space Transportation Industry: Critical Success Factors Assessment
Enyinda, Chris I.; Research Reports: 2001 NASA/ASEE Summer Faculty Fellowship Program; July 2002; In English; No Copyright; Avail: CASI; A01, Hardcopy

In response to the unrelenting call in both public and private sectors fora to reduce the high cost associated with space transportation, many innovative partially or fully RLV (Reusable Launch Vehicles) designs (X-34-37) were initiated. This call is directed at all levels of space missions including scientific, military, and commercial and all aspects of the missions such as nonrecurring development, manufacture, launch, and operations. According to Wertz, tbr over thirty years, the cost of space access has remained exceedingly high. The consensus in the popular press is that to decrease the current astronomical cost of access to space, more safer, reliable, and economically viable second generation RLVs (SGRLV) must be developed. Countries such as Brazil, India, Japan, and Israel are now gearing up to enter the global launch market with their own commercial space launch vehicles. NASA and the US space launch industry cannot afford to lag behind. Developing SGRLVs will immeasurably improve the US's space transportation capabilities by helping the US to regain the global commercial space markets while supporting the transportation capabilities of NASA's space missions, Developing the SGRLVs will provide affordable commercial space transportation that will assure the competitiveness of the US commercial space transportation industry in the 21st century. Commercial space launch systems are having difficulty obtaining financing because of the high cost and risk involved. Access to key financial markets is necessary for commercial space ventures. However, public sector programs in the form of tax incentives and credits, as well as loan guarantees are not yet available. The purpose of this paper is to stimulate discussion and assess the critical success factors germane for RLVs development and US global competitiveness.
Derived from text
Cost Reduction; Reusable Launch Vehicles; Space Commercialization; Research and Development; Commercial Spacecraft; NASA Programs

20020068708 NASA Ames Research Center, Moffett Field, CA USA
Progress in Unsteady Turbopump Flow Simulations
Kiris, Cetin C.; Chan, William; Kwak, Dochan; Williams, Robert; [2002]; In English; JANNAF 2002 Meeting, 8-12 Apr. 2002, Destin, FL, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation discusses unsteady flow simulations for a turbopump intended for a reusable launch vehicle (RLV). The simulation process makes use of computational grids and parallel processing. The architecture of the parallel computers used is discussed, as is the scripting of turbopump simulations.
CASI
Turbine Pumps; Unsteady Flow; Computational Grids; Simulation; Systems Analysis

20020068450 NASA Marshall Space Flight Center, Huntsville, AL USA
Estimating the Cost of NASA's Space Launch Initiative: How SLI Cost Stack Up Against the Shuttle
Hamaker, Joseph H.; Roth, Axel, Technical Monitor; [2002]; In English; SSCAG Meeting, 11-13 Jun. 2002, Frascati, Italy; No Copyright; Avail: Other Sources; Abstract Only

NASA is planning to replace the Space Shuttle with a new completely reusable Second Generation Launch System by approximately 2012. Numerous contracted and NASA in-house Space Transportation Architecture Studies and various technology maturation activities are proceeding and have resulted in scores of competing architecture configurations being proposed. Life cycle cost is a key discriminator between all these various concepts. However, the one obvious analogy for costing purposes remains the current Shuttle system. Are there credible reasons to believe that a second generation reusable launch system can be accomplished at less cost than the Shuttle? The need for a credible answer to this question is critical. This paper reviews the cost estimating approaches being used by the contractors and the government estimators to address this issue and explores the rationale behind the numbers.
Author
Space Shuttles; Life Cycle Costs; Cost Estimates; Reusable Launch Vehicles

20020067630 NASA Marshall Space Flight Center, Huntsville, AL USA
Magnetohydrodynamic Augmented Propulsion Experiment
Litchford, Ron J.; Cole, John; Lineberry, John; Chapman, Jim; Schmidt, Harold; Cook, Stephen, Technical Monitor; [2002]; In English; 33rd Plasmadynamics and Lasers Conference, 20-23 May 2002, Maui, HI, USA; No Copyright; Avail: Other Sources; Abstract Only

A fundamental obstacle to routine space access is the specific energy limitations associated with chemical fuels. In the case of vertical take-off, the high thrust needed for vertical liftoff and acceleration to orbit translates into power levels in the 10 GW range. Furthermore, useful payload mass fractions are possible only if the exhaust particle energy (i.e., exhaust velocity) is much greater than that available with traditional chemical propulsion. The electronic binding energy released by the best chemical reactions (e.g., LOX/LH2 for example, is less than 2 eV per product molecule (approx. 1.8 eV per H2O molecule), which translates into particle velocities less than 5 km/s. Useful payload fractions, however, will require exhaust velocities exceeding 15 km/s (i.e., particle energies greater than 20 eV). As an added challenge, the envisioned hypothetical RLV (reusable launch vehicle) should accomplish these amazing performance feats while providing relatively low acceleration levels to orbit (2-3g maximum). From such fundamental considerations, it is painfully obvious that planned and current RLV solutions based on chemical fuels alone represent only a temporary solution and can only result in minor gains, at best. What is truly needed is a revolutionary approach that will dramatically reduce the amount of fuel and size of the launch vehicle. This implies the need for new compact high-power energy sources as well as advanced accelerator technologies for increasing engine exhaust velocity. Electromagnetic acceleration techniques are of immense interest since they can be used to circumvent the thermal limits associated with conventional propulsion systems. This paper describes the Magnetohydrodynamic Augmented Propulsion Experiment (MAPX) being undertaken at NASA Marshall Space Flight Center (MSFC). In this experiment, a 1-MW arc heater is being used as a feeder for a 1-MW magnetohydrodynamic (MHD) accelerator. The purpose of the experiment is to demonstrate that an MHD accelerator can be an effective augmentation system for increasing engine exhaust velocity. More specifically, the experiment is intended to show that electromagnetic effects are effective at producing flow acceleration whereas electrothermal effects do not cause unacceptable heating of the working fluid. The MHD accelerator was designed as an externally diagonalized segmented Faraday channel, which will be inserted into an existing 2-tesla electromagnet. This allows the external power to be connected through two terminals thereby minimizing the complexity and cost associated with powering each segment independently. The design of the accelerator and other components in the flow path has been completed and fabrication activities are underway. This paper provides a full description of MAPX including performance analysis, design, and test plans, and current status.
Author
Magnetohydrodynamics; Electromagnetic Acceleration; Vertical Takeoff; Arc Heating; Magnetohydrodynamic Generators; Thrust Augmentation; Propulsion System Configurations

20020066164 NASA Marshall Space Flight Center, Huntsville, AL USA
A Plan for Advanced Guidance and Control Technology for 2nd Generation Reusable Launch Vehicles
Hanson, John M.; Fogle, Frank, Technical Monitor; [2002]; In English; No Copyright; Avail: CASI; A02, Hardcopy

Advanced guidance and control (AG&C) technologies are critical for meeting safety/reliability and cost requirements for the next generation of reusable launch vehicle (RLV). This becomes clear upon examining the number of expendable launch vehicle failures in the recent past where AG&C technologies would have saved a RLV with the same failure mode, the additional vehicle problems where this technology applies, and the costs associated with mission design with or without all these failure issues. The state-of-the-art in guidance and control technology, as well as in computing technology, is at the point where we can took to the possibility of being able to safely return a RLV in any situation where it can physically be recovered.

This paper outlines reasons for AG&C, current technology efforts, and the additional work needed for making this goal a reality.
Author
Reusable Launch Vehicles; Flight Control; Failure Modes; Mission Planning

20020065568 NASA Marshall Space Flight Center, Huntsville, AL USA
An Automated Method to Compute Orbital Re-Entry Trajectories with Heating Constraints
Zimmerman, Curtis; Dukeman, Greg; Hanson, John; Fogle, Frank R., Technical Monitor; [2002]; In English; No Copyright; Avail: CASI; A03, Hardcopy

Determining how to properly manipulate the controls of a re-entering re-usable launch vehicle (RLV) so that it is able to safely return to Earth and land involves the solution of a two-point boundary value problem (TPBVP). This problem, which can be quite difficult, is traditionally solved on the ground prior to flight. If necessary, a nearly unlimited amount of time is available to find the 'best' solution using a variety of trajectory design and optimization tools. The role of entry guidance during flight is to follow the pre-determined reference solution while correcting for any errors encountered along the way. This guidance method is both highly reliable and very efficient in terms of onboard computer resources. There is a growing interest in a style of entry guidance that places the responsibility of solving the TPBVP in the actual entry guidance flight software. Here there is very limited computer time. The powerful, but finicky, mathematical tools used by trajectory designers on the ground cannot in general be made to do the job. Nonconvergence or slow convergence can result in disaster. The challenges of designing such an algorithm are numerous and difficult. Yet the payoff (in the form of decreased operational costs and increased safety) can be substantial. This paper presents an algorithm that incorporates features of both types of guidance strategies. It takes an initial RLV orbital re-entry state and finds a trajectory that will safely transport the vehicle to a Terminal Area Energy Management (TAEM) region. During actual flight, the computed trajectory is used as the reference to be flown by a more traditional guidance method.
Author
Reusable Launch Vehicles; Reentry Trajectories; Trajectory Optimization; Design Optimization; Boundary Value Problems; Flight Control

20020065525 Science Applications International Corp., Torrance, CA USA
A Thermal Management Systems Model for the NASA GTX RBCC Concept
Traci, Richard M.; Farr, John L., Jr.; Laganelli, Tony; Walker, James, Technical Monitor; June 2002; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS3-99147; RTOP 708-90-63
Report No.(s): NASA/CR-2002-211587; NAS 1.26:211587; E-13372; SAIC-284-002-017; No Copyright; Avail: CASI; A05, Hardcopy

The Vehicle Integrated Thermal Management Analysis Code (VITMAC) was further developed to aid the analysis, design, and optimization of propellant and thermal management concepts for advanced propulsion systems. The computational tool is based on engineering level principles and models. A graphical user interface (GUI) provides a simple and straightforward method to assess and evaluate multiple concepts before undertaking more rigorous analysis of candidate systems. The tool incorporates the Chemical Equilibrium and Applications (CEA) program and the RJPA code to permit heat transfer analysis of both rocket and air breathing propulsion systems. Key parts of the code have been validated with experimental data. The tool was specifically tailored to analyze rocket-based combined-cycle (RBCC) propulsion systems being considered for space transportation applications. This report describes the computational tool and its development and verification for NASA GTX RBCC propulsion system applications.
Author
Aerothermodynamics; Rocket-Based Combined-Cycle Engines; Temperature Control; Aerospace Planes; Mathematical Models; Applications Programs (Computers)

20020063487 NASA Ames Research Center, Moffett Field, CA USA
The NASA Integrated Vehicle Health Management Technology Experiment for X-37
Schwabacher, Mark; Samuels, Jeff; Brownston, Lee; Clancy, Daniel, Technical Monitor; [2002]; In English; Conference on Component and Systems Diagnosis, Prognosis and Health Management II, SPIE AeroSense Meeting, 3 Apr. 2002, Orlando, FL, USA; No Copyright; Avail: CASI; A03, Hardcopy

The NASA Integrated Vehicle Health Management (IVHM) Technology Experiment for X-37 was intended to run IVHM

software on-board the X-37 spacecraft. The X-37 is intended to be an unpiloted vehicle that would orbit the Earth for up to 21 days before landing on a runway. The objectives of the experiment were to demonstrate the benefits of in-flight IVHM to the operation of a Reusable Launch Vehicle, to advance the Technology Readiness Level of this IVHM technology within a flight environment, and to demonstrate that the IVHM software could operate on the Vehicle Management Computer. The scope of the experiment was to perform real-time fault detection and isolation for X-37's electrical power system and electro-mechanical actuators. The experiment used Livingstone, a software system that performs diagnosis using a qualitative, model-based reasoning approach that searches system-wide interactions to detect and isolate failures. Two of the challenges we faced were to make this research software more efficient so that it would fit within the limited computational resources that were available to us on the X-37 spacecraft, and to modify it so that it satisfied the X-37's software safety requirements. Although the experiment is currently unfunded, the development effort had value in that it resulted in major improvements in Livingstone's efficiency and safety. This paper reviews some of the details of the modeling and integration efforts, and some of the lessons that were learned.
Author
Systems Health Monitoring; Management Systems; Technology Assessment; X-37 Vehicle; Applications Programs (Computers)

20020063408 NASA Glenn Research Center, Cleveland, OH USA
Status of the RBCC Direct-Connect Mixer Combustor Experiment
Walker, James F.; Kamhawi, Hani; Krivanek, Thomas M.; Thomas, Scott R.; Smith, Timothy D.; May 2002; In English; Combustion, Airbreathing Propulsion, Propulsion Systems Hazards, and Modelling and Simulation Subcommittes Joint Meeting, 8-12 Apr. 2002, Destin, FL, USA; Original contains color illustrations
Contract(s)/Grant(s): RTOP 708-90-63
Report No.(s): NASA/TM-2002-211555; NAS 1.15:211555; E-13334; No Copyright; Avail: CASI; A03, Hardcopy

The NASA Glenn Research Center is developing hydrogen based combined cycle propulsion technology for a single-stage-to-orbit launch vehicle application under a project called GTX. Rocket Based Combined Cycle (RBCC) propulsion systems incorporate one or more rocket engines into an airbreathing flow path to increase specific impulse as compared to an all rocket-powered vehicle. In support of this effort, an RBCC direct-connect test capability was established at the Engine Components Research Laboratory to investigate low speed, ejector ramjet, and initial ramjet operations and performance. The facility and test article enables the evaluation of two candidate low speed operating schemes; the simultaneous mixing and combustion (SMC) and independent ramjet stream (IRS). The SMC operating scheme is based on the fuel rich operations of the rocket where performance depends upon mixing between the rocket plume and airstream. In contrast, the IRS scheme fuels the airstream separately and uses the rocket plume to ignite the fuel-air mixture. This paper describes the test hardware and facility upgrades installed to support the RBCC tests. It also defines and discusses low speed technical challenges being addressed by the experiments. Finally, preliminary test results, including rocket risk mitigating tests, unfueled airflow tests, and the integrated system hot fire test will be presented.
Author
Rocket-Based Combined-Cycle Engines; Propulsion System Performance; Propulsion System Configurations; Engine Tests; Pressure Distribution; Combustion Chambers

20020060131 NASA Langley Research Center, Hampton, VA USA
Opportunities for Breakthroughs in Large-Scale Computational Simulation and Design
Alexandrov, Natalia; Alter, Stephen J.; Atkins, Harold L.; Bey, Kim S.; Bibb, Karen L.; Biedron, Robert T.; Carpenter, Mark H.; Cheatwood, F. McNeil; Drummond, Philip J.; Gnoffo, Peter A., et al.; June 2002; In English
Contract(s)/Grant(s): RTOP 706-31-21-80
Report No.(s): NASA/TM-2002-211747; L-18196; NAS 1.15:211747; No Copyright; Avail: CASI; A03, Hardcopy

Opportunities for breakthroughs in the large-scale computational simulation and design of aerospace vehicles are presented. Computational fluid dynamics tools to be used within multidisciplinary analysis and design methods are emphasized. The opportunities stem from speedups and robustness improvements in the underlying unit operations associated with simulation (geometry modeling, grid generation, physical modeling, analysis, etc.). Further, an improved programming environment can synergistically integrate these unit operations to leverage the gains. The speedups result from reducing the problem setup time through geometry modeling and grid generation operations, and reducing the solution time through the operation counts associated with solving the discretized equations to a sufficient accuracy. The opportunities are addressed only at a general level here, but an extensive list of references containing further details is included. The opportunities discussed are being addressed through the Fast Adaptive Aerospace Tools (FAAST) element of the Advanced Systems Concept

to Test (ASCoT) and the third Generation Reusable Launch Vehicles (RLV) projects at NASA Langley Research Center. The overall goal is to enable greater inroads into the design process with large-scale simulations.
Author
Aerospace Vehicles; Computational Fluid Dynamics; Simulation; Computer Aided Design

20020053646 NASA Ames Research Center, Moffett Field, CA USA
Time Dependent Simulation of Turbopump Flows
Kwak, Dochan; Kiris, Cetin C.; Chan, William; Williams, Robert; [2001]; In English; Thermal and Fluids Analysis Workshop (TFAWS-2001), 10-14 Sep. 2001, Huntsville, AL, USA; No Copyright; Avail: CASI; A03, Hardcopy

Viewgraphs detailing time-dependent simulations of Turbopump flows are presented. The topics include: 1) Major Drivers of Current Work; 2) Solution Methods; and 3) Unsteady Turbopump Flow.
CASI
Time Dependence; Computerized Simulation; Unsteady Flow; Turbine Pumps

20020051506 Hampton Univ., VA USA
Statistical Methods for Rapid Aerothermal Analysis and Design Technology
Morgan, Carolyn; DePriest, Douglas; Thompson, Richard, Technical Monitor; Jan. 31, 2002; In English
Contract(s)/Grant(s): NAG1-01075; HU Proj. 5-23845; No Copyright; Avail: CASI; A03, Hardcopy

The cost and safety goals for NASA's next generation of reusable launch vehicle (RLV) will require that rapid high-fidelity aerothermodynamic design tools be used early in the design cycle. To meet these requirements, it is desirable to establish statistical models that quantify and improve the accuracy, extend the applicability, and enable combined analyses using existing prediction tools. The research work was focused on establishing the suitable mathematical/statistical models for these purposes. It is anticipated that the resulting models can be incorporated into a software tool to provide rapid, variable-fidelity, aerothermal environments to predict heating along an arbitrary trajectory. This work will support development of an integrated design tool to perform automated thermal protection system (TPS) sizing and material selection.
Author
Reusable Launch Vehicles; Thermal Analysis; Statistical Analysis; Costs; Safety; Aerothermodynamics

20020050420 NASA Marshall Space Flight Center, Huntsville, AL USA
Water Flow Performance of a Superscale Model of the Fastrac Liquid Oxygen Pump
Skelley, Stephen; Zoladz, Thomas; The Tenth Thermal and Fluids Analysis Workshop; July 2001; In English; No Copyright; Avail: CASI; A02, Hardcopy

As part of the National Aeronautics and Space Administration's ongoing effort to lower the cost of access to space, the Marshall Space Flight Center has developed a rocket engine with 60,000 pounds of thrust for use on the Reusable Launch Vehicle technology demonstrator slated for launch in 2000. This gas generator cycle engine, known as the Fastrac engine, uses liquid oxygen and RP-1 for propellants and includes single stage liquid oxygen and RP-1 pumps and a single stage supersonic turbine on a common shaft. The turbopump design effort included the first use and application of new suction capability prediction codes and three-dimensional blade generation codes in an attempt to reduce the turbomachinery design and certification costs typically associated with rocket engine development. To verify the pump's predicted cavitation performance, a water flow test of a superscale model of the Fastrac liquid oxygen pump was conducted to experimentally evaluate the liquid oxygen pump's performance at and around the design point. The water flow test article replicated the flow path of the Fastrac liquid oxygen pump in a 1.582x scale model, including scaled seal clearances for correct leakage flow at a model operating speed of 5000 revolutions per minute. Flow entered the 3-blade axial-flow inducer, transitioned to a shrouded, 6- blade radial impeller, and discharged into a vaneless radial diffuser and collection volute. The test article included approximately 50 total and static pressure measurement locations as well as flush-mounted, high frequency pressure transducers for complete mapping of the pressure environment. The primary objectives of the water flow test were to measure the steady-state and dynamic pressure environment of the liquid oxygen pump versus flow coefficient, suction specific speed, and back face leakage flow rate. Initial results showed acceptable correlation between the predicted and experimentally measured pump head rise at low suction specific speeds. Likewise, only small circumferential variations in steady-state were observed from 80% to 120% of the design flow coefficient, matching the computational predictions and confirming that the integrated design approach has minimized any exit volute-induced distortions. The test article exhibited suction performance trends typically observed in inducer designs with virtually constant head rise with decreasing inlet pressure until complete pump head breakdown. Unfortunately, the net positive suction head at 3% head fall-off occurred far below that predicted at

all tested flow coefficients, resulting in a negative net positive suction head margin at the design point in water. Additional testing to map the unsteady pressure environment was conducted and cavitation-induced flow disturbances at the inducer inlet were observed. Two distinct disturbances were identified, one rotating and one stationary relative to the fixed frame of reference, while the transition from one regime to the next produced significant effects on the steady state pump performance. The impact of the unsteady phenomena and the corresponding energy losses on the unexpectedly poor pump performance is also discussed.
Author
Water Flow; Scale Models; Liquid Oxygen; Pumps; Rocket Engines; Fabrication

20020050402 NASA Marshall Space Flight Center, Huntsville, AL USA
Overview of Fluid Dynamic Activities at the Marshall Space Flight Center
Garcia, Roberto; Griffin, Lisa; Wang, Ten-See; The Tenth Thermal and Fluids Analysis Workshop; July 2001; In English; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

Contents include following: (1) Organizational Changes at MSFC. (2) Recent Program Support & Technology Development: analysis & cold flow testing: Fastrac, X-34, X-33, RLV, LFBB. (3) Ongoing Activities: RLV focused technology, RBCC concepts development, methodology & code development. (4) Future Activities and Direction: hardware design and development; tools development. (5) Concluding remarks: constraints, cooperation, opportunities.
CASI
Support Systems; Research Facilities; Research Management

20020050399 NASA Marshall Space Flight Center, Huntsville, AL USA
A Collaborative Analysis Tool for Thermal Protection Systems for Single Stage to Orbit Launch Vehicles
Alexander, Reginald; Stanley, Thomas Troy; The Tenth Thermal and Fluids Analysis Workshop; July 2001; In English; Original contains color illustrations; No Copyright; Avail: CASI; A02, Hardcopy

Presented is a design tool and process that connects several disciplines which are needed in the complex and integrated design of high performance reusable single stage to orbit (SSTO) vehicles. Every system is linked to all other systems, as is the case with SSTO vehicles with air breathing propulsion, which is currently being studied by the National Aeronautics and Space Administration (NASA). In particular, the thermal protection system (TPS) is linked directly to almost every major system. The propulsion system pushes the vehicle to velocities on the order of 15 times the speed of sound in the atmosphere before pulling up to go to orbit which results in high temperatures on the external surfaces of the vehicle. Thermal protection systems to maintain the structural integrity of the vehicle must be able to mitigate the heat transfer to the structure and be lightweight. Herein lies the interdependency, in that as the vehicle's speed increases, the TPS requirements are increased. And as TPS masses increase the effect on the propulsion system and all other systems is compounded. To adequately calculate the TPS mass of this type of vehicle several engineering disciplines and analytical tools must be used preferably in an environment that data is easily transferred and multiple iterations are easily facilitated.
Author
Thermal Protection; Structural Failure; Single Stage to Orbit Vehicles

20020048698 NASA Marshall Space Flight Center, Huntsville, AL USA, Societe Nationale d'Etude et de Construction de Moteurs d'Aviation, Bordeaux France
Demonstration of Advanced C/SiC Cooled Ramp
Bouquet, Clement; Laithier, Frederic; Lawrence, Timothy; Eckel, Andrew; Munafo, Paul M., Technical Monitor; [2002]; In English; 6th International Symposium on Propulsion for Space Transportation of the 21st Century, 14-16 May 2002, Versailles, France; Original contains color illustrations
Contract(s)/Grant(s): NAS8-99112; No Copyright; Avail: CASI; A02, Hardcopy

Under a NASA contract, SPS is evaluating its C/SiC to metal brazing technique for the development of light, composite, actively cooled panels. The program first consisted of defining a system applicable to the X-33 nozzle ramp. SPS then performed evaluation tests for tube, composite, and braze material selection, and for the adaptation of braze process parameters to the parts geometry. SPS is presently manufacturing a 250x60 millimeter squared specimen, including 10 metallic tubes, which will be cycled in the NASA/GRC-CELL-22 test bed under engine representative conditions.
Author
Silicon Carbides; Carbon; Composite Materials; Manufacturing; Cooling

20020048696 ATK-Thiokol Propulsion, USA

Automation of NDE on RSRM Metal Components

Hartman, John; Kirby, Mark; McCool, Alex, Technical Monitor; [2002]; In English; ASNT Spring Conference, 18-22 Mar. 2002, Portland, OR, USA; Original contains color illustrations

Contract(s)/Grant(s): NAS8-97238; No Copyright; Avail: CASI; A03, Hardcopy

An automated eddy current system has been designed and built, and is being implemented to inspect RSRM (Space Shuttle) metal components. The system provides a significant increase in inspection reliability, as well as other benefits such as data storage, chemical waste reduction and reduction in overall process time. This paper is in viewgraph form.

Derived from text

Nondestructive Tests; Reusable Launch Vehicles; Automation; Metals; Spacecraft Components

20020048273 NASA Ames Research Center, Moffett Field, CA USA

Fractographic Analysis of HfB2-SiC and ZrB2-SiC Composites

Mecholsky, J.J., Jr.; Ellerby, D. T.; Johnson, S. M.; Stackpoole, M. M.; Loehman, R. E.; Arnold, Jim, Technical Monitor; [2001]; In English; 26th Annual International Conference on Advanced Ceramics and Composites, 13-18 Jan. 2002; Copyright; Avail: Other Sources; Abstract Only

Hafnium diboride-silicon carbide and zirconium diboride-silicon carbide composites are potential materials for high temperature leading edge applications on reusable launch vehicles. In order to establish material constants necessary for evaluation of in-situ fracture, bars fractured in four point flexure were examined using fractographic principles. The fracture toughness was determined from measurements of the critical crack sizes and the strength values, and the crack branching constants were established to use in forensic fractography of materials for future flight applications. The fracture toughnesses range from about 13 MPam (sup 1/2) at room temperature to about 6 MPam (sup 1/2) at 1400 C for ZrB2-SiC composites and from about 11 MPam (sup 1/2) at room temperature to about 4 MPam (sup 1/2) at 1400 C for HfB2-SiC composites.

Author

Fracture Strength; Leading Edges; Reusable Launch Vehicles; Refractory Materials

20020047561 NASA Marshall Space Flight Center, Huntsville, AL USA

Permeability After Impact Testing of Composite Laminates

Nettles, A.T.; Munafo, Paul, Technical Monitor; 2002; In English; ASTM Symposia on Composites, 10-13 Mar. 2002, Pittsburgh, PA, USA; No Copyright; Avail: CASI; A03, Hardcopy

Since composite laminates are beginning to be identified for use in reusable launch vehicle propulsion systems, an understanding of their permeance is needed. A foreign object impact event can cause a localized area of permeability (leakage) in a polymer matrix composite and it is the aim of this study to assess a method of quantifying permeability-after-impact results. A simple test apparatus is presented and variables that could affect the measured values of permeability-after-impact were assessed. Once it was determined that valid numbers were being measured, a fiber/resin system was impacted at various impact levels and the resulting permeability measured, first with a leak check solution (qualitative) then using the new apparatus (quantitative). The results showed that as the impact level increased, so did the measured leakage. As the pressure to the specimen was increased, the leak rate was seen to increase in a non-linear fashion for almost all of the specimens tested.

Author

Impact Tests; Laminates; Permeability; Polymer Matrix Composites; Leakage

20020045392 NASA Marshall Space Flight Center, Huntsville, AL USA

Natural Atmospheric Environment Model Development for the National Aeronautics and Space Administration's Second Generation Reusable Launch Vehicle

Roberts, Barry C.; Leahy, Frank; Overbey, Glenn; Batts, Glen W.; Parker, Nelson, Technical Monitor; [2002]; In English; 9th Conference on Aviation, Range and Aerospace Meteorology, 13-16 May 2002, Portland, OR, USA; No Copyright; Avail: CASI; A01, Hardcopy

The National Aeronautics and Space Administration (NASA) recently began development of a new reusable launch vehicle. The program office is located at Marshall Space Flight Center (MSFC) and is called the Second Generation Reusable Launch Vehicle (2GRLV). The purpose of the program is to improve upon the safety and reliability of the first generation reusable launch vehicle, the Space Shuttle. Specifically, the goals are to reduce the risk of crew loss to less than 1-in-10,000 missions and decreased costs by a factor of 10 to approximately $1,000 per pound of payload launched to low Earth orbit. The program is currently in the very early stages of development and many two-stage vehicle concepts will be evaluated. Risk

reduction activities are also taking place. These activities include developing new technologies and advancing current technologies to be used by the vehicle. The Environments Group at MSFC is tasked by the 2GRLV Program to develop and maintain an extensive series of analytical tools and environmental databases which enable it to provide detailed atmospheric studies in support of structural, guidance, navigation and control, and operation of the 2GRLV.
Author
Atmospheric Models; Reusable Launch Vehicles; Technology Utilization

20020045346 Research Inst. for Advanced Computer Science, Moffett Field, CA USA
New V and V Tools for Diagnostic Modeling Environment (DME)
Pecheur, Charles; Nelson, Stacy; Merriam, Marshall, Technical Monitor; April 2002; In English
Contract(s)/Grant(s): NRA-8-30/TA-5-Northrup-Grumman
Report No.(s): NASA/CR-2002-211403; NAS 1.26:211403; No Copyright; Avail: CASI; A03, Hardcopy

The purpose of this report is to provide correctness and reliability criteria for verification and validation (V&V) of Second Generation Reusable Launch Vehicle (RLV) Diagnostic Modeling Environment, describe current NASA Ames Research Center tools for V&V of Model Based Reasoning systems, and discuss the applicability of Advanced V&V to DME. This report is divided into the following three sections: (1) correctness and reliability criteria; (2) tools for V&V of Model Based Reasoning; and (3) advanced V&V applicable to DME. The Executive Summary includes an overview of the main points from each section. Supporting details, diagrams, figures, and other information are included in subsequent sections. A glossary, acronym list, appendices, and references are included at the end of this report.
Author
Reliability Analysis; Reusable Launch Vehicles; Software Development Tools

20020045344 Research Inst. for Advanced Computer Science, Moffett Field, CA USA
V and V of Advanced Systems at NASA
Pecheur, Charles; Nelson, Stacy; April 2002; In English
Contract(s)/Grant(s): NRA-8-30/TA-5/Northrop-Grumman
Report No.(s): NASA/CR-2002-211401; NAS 1.26:211401; No Copyright; Avail: CASI; A05, Hardcopy

The purpose of this report is to provide the following: 1) Overview of Formal Methods beneficial to Verification and Validation (V&V) of 2nd Generation Re-usable Launch Vehicle Integrated Vehicle Health Management (RLV IVHM); 2) Description of current use of Formal Methods at NASA and their applicability to 2nd Generation RLV IVHM software; and 3) Guide for incorporation Formal Methods into NASA V&V Standards for certification of airborne software like 2nd Generation RLV IVHM.
Author
NASA Programs; Reusable Launch Vehicles; Program Verification (Computers); Software Engineering

20020044823 NASA Marshall Space Flight Center, Huntsville, AL USA
X-33 LH2 Tank Failure Investigation Findings
Niedermeyer, Melinda; Munafo, Paul, Technical Monitor; [2002]; In English; Manufacturing Problem Prevention Workshop, 26-27 Feb. 2002, El Segundo, CA, USA; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation gives an overview of the X-33 LH2 tank failure investigation findings. The conclusions of the investigation include the following: (1) the inner skin microcracked and hydrogen infiltrated; (2) the cracks grew larger under pressure; (3) when pressure was removed, the cracks closed slightly; (4) when the tank was drained and warmed, the cracks closed and blocked the leak path; (5) FOD and debond areas provided an opportunity for a leak path; and (6) there is still hydrogen in the the other three lobes today.
Derived from text
Failure Analysis; Leakage; Microcracks; X-33 Reusable Launch Vehicle

20020044103 Research Inst. for Advanced Computer Science, Moffett Field, CA USA, Computer Sciences Corp., Moffett Field, CA USA
Survey of NASA V and V Processes/Methods
Pecheur, Charles; Nelson, Stacy; April 2002; In English
Contract(s)/Grant(s): NRA-8-30/TA-5/Northrop-Grumman
Report No.(s): NASA/CR-2002-211401; NAS 1.26:211401; No Copyright; Avail: CASI; A05, Hardcopy

The purpose of this report is to describe current NASA Verification and Validation (V&V) techniques and to explain how these techniques are applicable to 2nd Generation RLV Integrated Vehicle Health Management (IVHM) software. It also contains recommendations for special V&V requirements for IVHM. This report is divided into the following three sections: 1) Survey - Current NASA V&V Processes/Methods; 2) Applicability of NASA V&V to 2nd Generation RLV IVHM; and 3) Special 2nd Generation RLV IVHM V&V Requirements.
Author
Systems Integration; Program Verification (Computers); Software Engineering; NASA Programs; Surveys

20020043393 NASA Ames Research Center, Moffett Field, CA USA
Thermal Protection Materials for Reentry Applications
Johnson, Sylvia M.; Stackpoole, Mairead; Gusman, Mike; Loehman, Ron; Kotula, Paul; Ellerby, Donald; Arnold, James; Wercinski, Paul; Reuthers, James; Kontinos, Dean, et al.; [2001]; In English; Structural Ceramics and Ceramic Composites for High-Temperature Applications Conference, 11 Oct. 2001, Seville, Spain; No Copyright; Avail: CASI; A03, Hardcopy

Thermal protection materials and systems (IRS) are used to protect spacecraft during reentry into Earth's atmosphere or entry into planetary atmospheres. As such, these materials are subject to severe environments with high heat fluxes and rapid heating. Catalytic effects can increase the temperatures substantially. These materials are also subject to impact damage from micrometeorites or other debris during ascent, orbit, and descent, and thus must be able to withstand damage and to function following damage. Thermal protection materials and coatings used in reusable launch vehicles will be reviewed, including the needs and directions for new materials to enable new missions that require faster turnaround and much greater reusability. The role of ablative materials for use in high heat flux environments, especially for non-reusable applications and upcoming planetary missions, will be discussed. New thermal protection system materials may enable the use of sharp nose caps and leading edges on future reusable space transportation vehicles. Vehicles employing this new technology would have significant increases in maneuverability and out-of-orbit cross range compared to current vehicles, leading to increased mission safety in the event of the need to abort during ascent or from orbit. Ultrahigh temperature ceramics, a family of materials based on HfB_2 and ZrB_2 with SiC, will be discussed. The development, mechanical and thermal properties, and uses of these materials will be reviewed.
Author
Thermal Protection; Spacecraft Reentry; Reusable Launch Vehicles; Planetary Atmospheres; Ablative Materials

20020043183 NASA Ames Research Center, Moffett Field, CA USA
NASA Exhibits
Deardorff, Glenn; Djomehri, M. Jahed; Freeman, Ken; Gambrel, Dave; Green, Bryan; Henze, Chris; Hinke, Thomas; Hood, Robert; Kiris, Cetin; Moran, Patrick; Biegel, Bryan, Technical Monitor, et al.; [2001]; In English; Supercomputing 2001, 10-16 Nov. 2001, Denver, CO, USA
Contract(s)/Grant(s): RTOP 704-40-12; RTOP 725-10-11; RTOP 725-10-31; RTOP 727-01-83; No Copyright; Avail: CASI; A03, Hardcopy

A series of NASA presentations for the Supercomputing 2001 conference are summarized. The topics include: (1) Mars Surveyor Landing Sites 'Collaboratory'; (2) Parallel and Distributed CFD for Unsteady Flows with Moving Overset Grids; (3) IP Multicast for Seamless Support of Remote Science; (4) Consolidated Supercomputing Management Office; (5) Growler: A Component-Based Framework for Distributed/Collaborative Scientific Visualization and Computational Steering; (6) Data Mining on the Information Power Grid (IPG); (7) Debugging on the IPG; (8) Debakey Heart Assist Device: (9) Unsteady Turbopump for Reusable Launch Vehicle; (10) Exploratory Computing Environments Component Framework; (11) OVERSET Computational Fluid Dynamics Tools; (12) Control and Observation in Distributed Environments; (13) Multi-Level Parallelism Scaling on NASA's Origin 1024 CPU System; (14) Computing, Information, & Communications Technology; (15) NAS Grid Benchmarks; (16) IPG: A Large-Scale Distributed Computing and Data Management System; and (17) ILab: Parameter Study Creation and Submission on the IPG.
Derived from text
Supercomputers; Parallel Processing (Computers); Multiprocessing (Computers); Distributed Processing; Conferences; Computation

20020041448 Amain Electronics Co., Inc., Reno, NV USA
Mosad and Stream Vision For A Telerobotic, Flying Camera System
Mandl, William; Apr. 03, 2002; In English; Diskette: 1 3.5-Inch DSHD diskette containing full text document in PDF format
Contract(s)/Grant(s): NAS3-00031
Report No.(s): Amain-Rept-02002; NONP-NASA-DK-2002067346; No Copyright; Avail: CASI; A03, Hardcopy

Two full custom camera systems using the Multiplexed OverSample Analog to Digital (MOSAD) conversion technology for visible light sensing were built and demonstrated. They include a photo gate sensor and a photo diode sensor. The system includes the camera assembly, driver interface assembly, a frame stabler board with integrated decimeter and Windows 2000 compatible software for real time image display. An array size of 320X240 with 16 micron pixel pitch was developed for compatibility with 0.3 inch CCTV optics. With 1.2 micron technology, a 73% fill factor was achieved. Noise measurements indicated 9 to 11 bits operating with 13.7 bits best case. Power measured under 10 milliwatts at 400 samples per second. Nonuniformity variation was below noise floor. Pictures were taken with different cameras during the characterization study to demonstrate the operable range. The successful conclusion of this program demonstrates the utility of the MOSAD for NASA missions, providing superior performance over CMOS and lower cost and power consumption over CCD. The MOSAD approach also provides a path to radiation hardening for space based applications.
Author
Cameras; Telerobotics; Analog to Digital Converters; Fabrication; Sensors; Light (Visible Radiation); Charge Coupled Devices

20020036009 Lee and Associates, LLC, USA
Support to 2nd Generation RLV Propulsion Project Office
Lee, Thomas J.; Mar. 07, 2002; In English
Contract(s)/Grant(s): NASA Order H-34351-D; No Copyright; Avail: CASI; A03, Hardcopy

In this final report regarding support to the second generation RLV (Reusable Launch Vehicle) propulsion project office, a list of tasks accomplished is presented. During this period, Lee & Associates, LLC participated in numerous Systems Requirements Reviews (SRR) related to the Cobra development program.
CASI
Propulsion; Reusable Launch Vehicles

20020034965 NASA Marshall Space Flight Center, Huntsville, AL USA
X-33 Tank Failure During Autoclave Fabrication
Nettles, Alan T.; Munafo, Paul, Technical Monitor; [2001]; In English; US Air Force Manufacturing Problem Prevention Program, 5-6 Jun. 2001, El Segundo, CA, USA; No Copyright; Avail: Other Sources; Abstract Only

During a repair cure cycle on tank #1 of the X-33 liquid hydrogen tanks, a skin to core disbond occurred. Both the inner skin and outer skin of the lobe #1 sandwich panel was noted to have been disbonded and cracked- An investigation was undertaken to determine the cause of this failure. The investigation consisted of reviewing all of the processing data and performing testing on the failed lobe #1, as well as the other lobes, which did not fail during the cure cycle. The tests consisted of residual stress measurements in one of the intact lobes and 'plug-pulls' to assess skin to core strength on all of the remaining lobes. Results showed an extremely low bondline strength due to lack of proper filleting of the adhesive, in addition, tests showed a very rapid decrease in strength with increasing temperature, as well as a further decrease in strength with a larger number of cycles. Also, the honeycomb used was not vented so pressure could build up within the cells. All of these factors appeared to be contributors to the failure.
Author
X-33 Reusable Launch Vehicle; Propellant Tanks; Liquid Hydrogen; Failure

20020034460 NASA Marshall Space Flight Center, Huntsville, AL USA
Findings from the X-33 Hydrogen Tank Failure Investigation
Niedermeyer, Melinda; Munafo, Paul M., Technical Monitor; [2001]; In English; US Air Force Manufacturing Problem Prevention Program, 5-6 Jun. 2001, El Segundo, CA, USA; No Copyright; Avail: Other Sources; Abstract Only

The X-33 Hydrogen tank failed during test in November of 1999 at MSFC. The tank completed the structural loading phase of the test successfully and was drained of hydrogen prior to the failure. The failure initiated in the acreage of Lobe 1 and was instantaneous, peeling the outer skin and core away from the inner skin. It was determined there were several factors that provided the opportunity for the tank to fail in this way. The factor giving life to these opportunistic circumstances was hydrogen infiltration into the core of the tank. The mechanism for this phenomenon will be discussed in this presentation.
Author
X-33 Reusable Launch Vehicle; Propellant Tanks; Liquid Hydrogen; Failure

20020032739 Georgia Inst. of Tech., Atlanta, GA USA

Improvements in Thermal Protection Sizing Capabilities for TCAT: Conceptual Design for Advanced Space Transportation Systems

Olds, John R.; Izon, Stephen James; February 2002; In English

Contract(s)/Grant(s): NAG8-1597

Report No.(s): E-16-P51; No Copyright; Avail: CASI; A05, Hardcopy

The Thermal Calculation Analysis Tool (TCAT), originally developed for the Space Systems Design Lab at the Georgia Institute of Technology, is a conceptual design tool capable of integrating aeroheating analysis into conceptual reusable launch vehicle design. It provides Thermal Protection System (TPS) unit thicknesses and acreage percentages based on the geometry of the vehicle and a reference trajectory to be used in calculation of the total cost and weight of the vehicle design. TCAT has proven to be reasonably accurate at calculating the TPS unit weights for in-flight trajectories; however, it does not have the capability of sizing TPS materials above cryogenic fuel tanks for ground hold operations. During ground hold operations, the vehicle is held for a brief period (generally about two hours) during which heat transfer from the TPS materials to the cryogenic fuel occurs. If too much heat is extracted from the TPS material, the surface temperature may fall below the freezing point of water, thereby freezing any condensation that may be present at the surface of the TPS. Condensation or ice on the surface of the vehicle is potentially hazardous to the mission and can also damage the TPS. It is questionable whether or not the TPS thicknesses provided by the aeroheating analysis would be sufficiently thick to insulate the surface of the TPS from the heat transfer to the fuel. Therefore, a design tool has been developed that is capable of sizing TPS materials at these cryogenic fuel tank locations to augment TCAT's TPS sizing capabilities.

Derived from text

Aerodynamic Heat Transfer; Surface Temperature; Systems Analysis; Systems Engineering; Thermal Protection; Computerized Simulation

20020024753 NASA Marshall Space Flight Center, Huntsville, AL USA

An Advanced Video Sensor for Automated Docking

Howard, Richard T.; Bryan, Thomas C.; Book, Michael L.; Roe, Fred, Technical Monitor; [2001]; In English; 20th Digital Avionics Systems Conference, 14-18 Oct. 2001, Daytona Beach, FL, USA; No Copyright; Avail: Other Sources; Abstract Only

This paper describes the current developments in video-based sensors at the Marshall Space Flight Center. The Advanced Video Guidance Sensor is the latest in a line of video-based sensors designed for use in automated docking systems. The X-33, X-34, X-38, and X-40 are all designed to be unpiloted vehicles; such vehicles will require a sensor system that will provide adequate data for the vehicle to accomplish its mission. One of the primary tasks planned for re-usable launch vehicles is to resupply the space station. In order to approach the space station in a self-guided manner, the vehicle must have a reliable and accurate sensor system to provide relative position and attitude information between the vehicle and the space station. The Advanced Video Guidance Sensor is being designed and built to meet this requirement, as well as requirements for other vehicles docking to a variety of target spacecraft. The Advanced Video Guidance Sensor is being designed to allow range and bearing information to be measured at ranges up to 2 km. The sensor will measure 6-degree-of-freedom information (relative positions and attitudes) from approximately 40 meters all the way in to final contact (approximately 1 meter range). The sensor will have a data output rate of 20 Hz during tracking mode, and will be able to acquire a target within one half of a second. The prototype of the sensor will be near completion at the time of the conference.

Author

Automatic Control; Spacecraft Docking; Reusable Launch Vehicles; Guidance Sensors

20020024007 NASA Glenn Research Center, Cleveland, OH USA

High Voltage Design Guidelines: A Timely Update

Hillard, G. Barry; Kirkici, H.; Ensworth, Clint, Technical Monitor; [2001]; In English; 2001 Conference on Electrical Insulation and Dielectric Phenomena, 14-17 Oct. 2001, Kitchener, Ontario, Canada

Contract(s)/Grant(s): RTOP 376-80-21; No Copyright; Avail: CASI; A01, Hardcopy

The evolving state of high voltage systems and their increasing use in the space program have called for a revision of the High Voltage Design Guidelines, Marshall Space Flight Center technical document MSFC-STD-531, originally issued September 1978 (previously 50 M05189b, October 1972). These guidelines deal in depth with issues relating to the specification of materials, particularly electrical insulation, as well as design practices and test methods. Emphasis is on corona and Paschen breakdown as well as plasma effects for Low Earth Orbiting systems. We will briefly review the history of these guidelines as well as their immediate predecessors and discuss their range of applicability. In addition, this document has served as the basis for several derived works that became focused, program-specific HV guidelines. We will briefly review

two examples, guidelines prepared for the X-33 program and for the Space Shuttle Electric Auxiliary Power Unit (EAPU) upgrade.
Author
High Voltages; Space Shuttles; Auxiliary Power Sources; Design Analysis; Space Programs

20020023442 Boeing Phantom Works, USA
Dynamics and Stability and Control Characteristics of the X-37
Chaudhary, Ashwani; Nguyen, Viet; Tran, Hoi; Poladian, David; Falangas, Eric; Turner, Susan G., Technical Monitor; [2001]; In English; AIAA GN&C Conference, 9 Aug. 2001, Montreal, Canada
Contract(s)/Grant(s): NCC8-190; Copyright; Avail: CASI; A02, Hardcopy

This paper presents the stability and control analysis and the control design results for the Boeing/NASA/AFRL X-37. The X-37 is a flight demonstrator vehicle that will go into space and after its mission, autonomously reenter and land on a conventional runway. This paper studies the dynamics and control of the X-37 from atmospheric reentry through landing. A nominal trajectory that lands on the Edwards Air Force Base Lakebed is considered for all the analysis and design. The X-37's longitudinal and lateral/directional bare-airframe characteristics are presented. The level of maneuvering control power is assessed. Vehicle trim with multiple surfaces is discussed. Special challenges where the wings loose roll effectiveness are discussed and solutions are presented. Aerodynamic uncertainties and flexibility modeling issues are presented. Control design results and robustness analysis methods are presented. Results are provided for the Entry, Terminal Area Energy Management (TAEM), and Approach and Land phases.
Author
Stability; Terminal Area Energy Management; X-37 Vehicle; Control Systems Design; Dynamic Characteristics

20020022529 NASA Glenn Research Center, Cleveland, OH USA
2000 NASA Seal/Secondary Air System Workshop, Volume 1
Steinetz, Bruce M., Editor; Hendricks, Robert C., Editor; October 2001; In English, 25-26 Oct. 2000, Cleveland, OH, USA
Contract(s)/Grant(s): RTOP 706-85-37
Report No.(s): NASA/CP-2001-211208/VOL1; E-13062/VOL1; NAS 1.55:211208/VOL1; No Copyright; Avail: CASI; A21, Hardcopy

The 2000 NASA Seal/Secondary Air System Workshop covered four main areas: (1) overviews of NASA-sponsored Ultra-Efficient Engine Technology (UEET) and Access to Space Programs, with emphasis on program goals and seal needs; (2) review of turbine engine seal issues from the perspective of end users such as United Airlines; (3) reviews of sealing concepts, test results, experimental facilities, and numerical predictions; and (4) reviews of material development programs relevant to advanced seals development. The NASA UEET overview illustrates for the reader the importance of advanced technologies, including seals, in meeting future engine system efficiency and emission goals. GE, Pratt & Whitney, and Honeywell presented advanced seal development work being performed within their organizations. The NASA-funded GE/Stein Seal team has successfully demonstrated a large (3-ft. diam) aspirating seal that can withstand all anticipated pressures, speeds, and rotor runouts anticipated for a GE90 L.P. turbine balance piston location. GE/Stein Seal are fabricating a full-scale seal to be tested in a GE-90 ground test engine in early 2002. Pratt & Whitney and Stein Seal are investigating carbon seals to accommodate large radial movements anticipated in future geared-fan gearbox locations. Honeywell presented a finger seal design being considered for a high-temperature static combustor location incorporating ceramic finger elements. Successful demonstration of the braided carbon rope thermal barriers to extreme temperatures (5500 F) for short durations provide a new form of very high temperature thermal barrier for future Shuttle solid rocket motor nozzle joints. The X-37, X-38, and future highly reusable launch vehicles pose challenging control surface seal demands that require new seal concepts made from emerging high temperature ceramics and other materials.
Author
Aerospace Engineering; Seals (Stoppers); Conferences; Product Development

20020022190 Georgia Inst. of Tech., Atlanta, GA USA
Evaluation of Advanced Thermal Protection Techniques for Future Reusable Launch Vehicles
Olds, John R.; Cowart, Kris; June 2001; In English
Contract(s)/Grant(s): NCC2-5332; No Copyright; Avail: CASI; A09, Hardcopy

A method for integrating Aeroheating analysis into conceptual reusable launch vehicle RLV design is presented in this thesis. This process allows for faster turn-around time to converge a RLV design through the advent of designing an optimized

thermal protection system (TPS). It consists of the coupling and automation of four computer software packages: MINIVER, TPSX, TCAT and ADS. MINIVER is an Aeroheating code that produces centerline radiation equilibrium temperatures, convective heating rates, and heat loads over simplified vehicle geometries. These include flat plates and swept cylinders that model wings and leading edges, respectively. TPSX is a NASA Ames material properties database that is available on the World Wide Web. The newly developed Thermal Calculation Analysis Tool (TCAT) uses finite difference methods to carry out a transient in-depth I-D conduction analysis over the center mold line of the vehicle. This is used along with the Automated Design Synthesis (ADS) code to correctly size the vehicle's thermal protection system JPS). The numerical optimizer ADS uses algorithms that solve constrained and unconstrained design problems. The resulting outputs for this process are TPS material types, unit thicknesses, and acreage percentages. TCAT was developed for several purposes. First, it provides a means to calculate the transient in-depth conduction seen by the surface of the TPS material that protects a vehicle during ascent and reentry. Along with the in-depth conduction, radiation from the surface of the material is calculated along with the temperatures at the backface and interior parts of the TPS material. Secondly, TCAT contributes added speed and automation to the overall design process. Another motivation in the development of TCAT is optimization.
Derived from text
Aerodynamic Heating; Algorithms; Computer Programs; Convective Heat Transfer; Design Analysis; Thermal Analysis

20020022187 NASA Marshall Space Flight Center, Huntsville, AL USA
Thermal/Fluid Analysis of a Composite Heat Exchanger for Use on the RLV Rocket Engine
Nguyen, Dalton; Turner, Larry D., Technical Monitor; [2001]; In English; Thermal Fluid Analysis Workshop, 10-14 Sep. 2001, Huntsville, AL, USA; Original contains color illustrations; No Copyright; Avail: CASI; A02, Hardcopy

As part of efforts to design a regeneratively cooled composite nozzle ramp for use on the reusable vehicle (RLV) rocket engine, a C-SiC composite heat exchanger concept was proposed for thermal performance evaluation. To test the feasibility of the concept, sample heat exchanger panels were made to fit the Glenn Research Center's cell 22 for testing. Operation of the heat exchanger was demonstrated in a combustion environment with high heat fluxes similar to the RLV Aerospike Ramp. Test measurements were reviewed and found to be valuable for the on-going fluid and thermal analysis of the actual RLV composite ramp. Since the cooling fluid for the heat exchanger is water while the RLV Ramp cooling fluid is LH2, fluid and therma models were constructed to correlate to the specific test set-up. The knowledge gained from this work will be helpful for analyzing the thermal response of the actual RLV Composite Ramp. The coolant thermal properties for the models are taken from test data. The heat exchanger's cooling performance was analyzed using the Generalized Fluid System Simulation Program (GFSSP). Temperatures of the heat exchanger's structure were predicted in finite element models using Patran and Sinda. Results from the analytical models and the tests show that RSC's heat exchanger satisfied the combustion environments in a series of 16 tests.
Author
Heat Exchangers; Computerized Simulation; Performance Tests; Carbon-Carbon Composites; Thermal Analysis

20020022180 Boeing Reusable Space Systems, Seal Beach, CA USA
Test Results for an Off-the-Shelf GPS/INS During Approach and Landing Testing of the X-40A
Childers, Dave; Gelderloos, Hendrik; Turner, Susan, Technical Monitor; [2001]; In English; 20th Digital Avionics Conference, 16-18 Oct. 2001, Daytona Beach, FL, USA
Contract(s)/Grant(s): NAS8-190; No Copyright; Avail: CASI; A03, Hardcopy

The X-37 is an unpiloted, reusable space vehicle that will be launched into space, orbit the Earth, reenter the atmosphere, and land autonomously. At the heart of the Guidance, Navigation and Control (GN&C) will be the Space Integrated GPS/INS (SIGI) system, an off the shelf navigation grade GPS/INS that has been enhanced for space and reentry environments. SIGI will provide both navigation and flight control data to the X-37's GN&C. The X-40A is an unpiloted experimental vehicle whose shape and performance are similar to the X-37's and was flown earlier this year to develop and test the approach and landing phase of X-37. On board the X-40A is the X-37's SIGI, which is riding along as an experiment. The X-40A SIGI experiment provided early characterization of SIGI operation and performance with differential GPS (Global Positioning System) during real world approach and landing. Characterization testing was geared toward assessment of reliability and performance of the system. The objectives were to demonstrate performance levels sufficient to meet the X-37 requirements for automatic, autonomous approach and landing, demonstrate reliability over repeated ground and flight tests, and reduce risk for integration of SIGI into the vehicle and support environment. This paper presents a summary of this testing and the results to date.
Author
Flight Tests; Autonomous Navigation; Spacecraft Reliability; X-37 Vehicle; Performance Tests; Systems Integration

20020021989 NASA Marshall Space Flight Center, Huntsville, AL USA

X-33 Tank Failure During Autoclave Fabrication

Nettles, Alan T.; Munafo, Paul, Technical Monitor; [2001]; In English; Joint NASA/AF Meeting on Honeycomb Panels, 2 Oct. 2001, El Segundo, CA, USA; No Copyright; Avail: CASI; A03, Hardcopy

The composite liquid hydrogen tank (tank #1 of 2) for the X-33 flight vehicle is made up of four lobes that have a sandwich construction, bonded to a frame of longerons. Lobes 1 and 4 showed local disbonds to the longerons they were bonded to. The 'bad' areas were cut away and patched with new material. The new material was cured by placing the entire tank in a heated autoclave with no pressure. Upon removal from the autoclave, it was noted that lobe 1 had severe skin/core disbonds on the inner and outer skins. The skins on this lobe were cracked as well. The core was disbonded from the inner skin across the entire acreage, except for spots around the lobe perimeter. The outer skin was separated from the core in a region near the center of the lobe. Lobe 1 was removed from the tank on January 13, 1999. Bolts were placed through the lobe to hold it together and the cuts on the inner skin were not continuous, but 'tabs' were left for final cutting and removal. Upon closer inspection of the disbonded basesheet, it was noted that there was a lack of filleting into the honeycomb core. Good fillets are critical to bond strength.

Derived from text

Bolts; Debonding (Materials); Defects; Joints (Junctions); Sandwich Structures

20020021567 NASA Marshall Space Flight Center, Huntsville, AL USA

NASA's Space Launch Initiative Targets Toxic Propellants

Hurlbert, Eric; McNeal, Curtis; Davis, Daniel J., Technical Monitor; [2001]; In English; 4th International Peroxide Propulsion Conference, 20-22 Jun. 2001, Noordwijk, Netherlands

Report No.(s): CP-50; No Copyright; Avail: CASI; A02, Hardcopy

When manned and unmanned space flight first began, the clear and overriding design consideration was performance. Consequently, propellant combinations of all kinds were considered, tested, and, when they lifted the payload a kilometer higher, or an extra kilogram to the same altitude, they became part of our operational inventory. Cost was not considered. And with virtually all of the early work being performed by the military, safety was hardly a consideration. After all, fighting wars has always been dangerous. Those days are past now. With space flight, and the products of space flight, a regular part of our lives today, safety and cost are being reexamined. NASA's focus turns naturally to its Shuttle Space Transportation System. Designed, built, and flown for the first time in the 1970s, this system remains today America's workhorse for manned space flight. Without its tremendous lift capability and mission flexibility, the International Space Station would not exist. And the Hubble telescope would be a monument to shortsighted management, rather than the clear penetrating eye on the stars it is today. But the Shuttle system fully represents the design philosophy of its period: it is too costly to operate, and not safe enough for regular long term access to space. And one of the key reasons is the utilization of toxic propellants. This paper will present an overview of the utilization of toxic propellants on the current Shuttle system.

Derived from text

Propellants; Space Transportation System; Reusable Launch Vehicles; Cryogenics

20020020963 NASA Langley Research Center, Hampton, VA USA

Advanced Metallic Thermal Protection System Development

Blosser, M. L.; Chen, R. R.; Schmidt, I. H.; Dorsey, J. T.; Poteet, C. C.; Bird, R. K.; [2002]; In English; 40th Aerospace Sciences Meeting and Exhibit, 14-17 Jan. 2002, Reno, NV, USA; Original contains color illustrations

Report No.(s): AIAA Paper 2002-0504; Copyright; Avail: CASI; A03, Hardcopy

A new Adaptable, Robust, Metallic, Operable, Reusable (ARMOR) thermal protection system (TPS) concept has been designed, analyzed, and fabricated. In addition to the inherent tailorable robustness of metallic TPS, ARMOR TPS offers improved features based on lessons learned from previous metallic TPS development efforts. A specific location on a single-stage-to-orbit reusable launch vehicle was selected to develop loads and requirements needed to design prototype ARMOR TPS panels. The design loads include ascent and entry heating rate histories, pressures, acoustics, and accelerations. Additional TPS design issues were identified and discussed. An iterative sizing procedure was used to size the ARMOR TPS panels for thermal and structural loads as part of an integrated TPS/cryogenic tank structural wall. The TPS panels were sized to maintain acceptable temperatures on the underlying structure and to operate under the design structural loading. Detailed creep analyses were also performed on critical components of the ARMOR TPS panels. A lightweight, thermally compliant TPS support system (TPSS) was designed to connect the TPS to the cryogenic tank structure. Four 18-inch-square ARMOR TPS panels were fabricated. Details of the fabrication process are presented. Details of the TPSS for connecting the ARMOR

TPS panels to the externally stiffened cryogenic tank structure are also described. Test plans for the fabricated hardware are presented.
Author
Fabrication; Panels; Thermal Protection; Cryogenic Fluid Storage; Creep Analysis

20020020427 NASA Marshall Space Flight Center, Huntsville, AL USA
NASA Alternate Access to Station Service Concept
Bailey, Michelle D.; Crumbly, Chris; [2002]; In English; 53rd International Astronautical Congress Conference, 10-19 Oct. 2002, Houston, TX, USA; No Copyright; Avail: Other Sources; Abstract Only

The evolving nature of the NASA space enterprise compels the agency to develop new and innovative space systems concepts. NASA, working with increasingly strained budgets and a declining manpower base, is attempting to transform from operational activities to procurement of commercial services. NASA's current generation reusable launch vehicle, the Shuttle, is in transition from a government owned and operated entity to a commercial venture to reduce the civil servant necessities for that program. NASA foresees its second generation launch vehicles being designed and operated by industry for commercial and government services. The 'service' concept is a pioneering effort by NASA. The purpose the 'service' is not only to reduce the civil servant overhead but will free up government resources for further research - and enable industry to develop a space business case so that industry can sustain itself beyond government programs. In addition, NASA desires a decreased responsibility thereby decreasing liability. The Second Generation Reusable Launch Vehicle (RLV) program is implementing NASA's Space Launch Initiative (SLI) to enable industry to develop the launch vehicles of the future. The Alternate Access to Station (AAS) project office within this program is chartered with enabling industry to demonstrate an alternate access capability for the International Space Station (ISS). The project will not accomplish this by traditional government procurement methods, not by integrating the space system within the project office, or by providing the only source of business for the new capability. The project funds will ultimately be used to purchase a service to take re-supply cargo to the ISS, much the same as any business might purchase a service from FedEx to deliver a package to its customer. In the near term, the project will fund risk mitigation efforts for enabling technologies. AAS is in some ways a precursor to the 2nd Generation RLV. By accomplishing ISS resupply with existing technologies, not only will a new category of autonomous vehicles deliver cargo, but a commercial business base will be incubated that will improve the likelihood of commercial convergence with the next generation of RLVs. Traditional paradigms in government management and acquisition philosophy are being challenged in order to bring about the objective of the AAS project. The phased procurement approach is proving to be the most questionable aspect to date. This work addresses the fresh approach AAS is adopting in management and procurement through a study of the AAS history, current solutions, key technologies, procurement complications, and an incremental forward plan leading to the purchase of a service to deliver goods to ISS. Included in this work is a discussion of the Commercial Space Act of 1998 and how it affects government purchase of space launch and space vehicle services. Industry should find these topics pertinent to their current state of business.
Author
Reusable Launch Vehicles; International Space Station; Space Commercialization

20020020362 NASA Marshall Space Flight Center, Huntsville, AL USA
Second Generation RLV Space Vehicle Concept
Bailey, Michelle; Daniel, Charles; Throckmorton, David A., Technical Monitor; [2002]; In English; 53rd International Aeronautical Congress Conference, 10-19 Oct. 2002, Houston, TX, USA; No Copyright; Avail: Other Sources; Abstract Only

NASA has a long history of conducting development programs and projects in a consistent fashion. Systems Engineering within those programs and projects has also followed a given method outlined by such documents as the NASA Systems Engineering Handbook. The relatively new NASA Space Launch Initiative (SLI) is taking a new approach to developing a space vehicle, with innovative management methods as well as new Systems Engineering processes. With the program less than a year into its life cycle, the efficacy of these new processes has yet to be proven or disproven. At $776M for phase 1, SLI represents a major portion of the NASA focus; however, the new processes being incorporated are not reflected in the training provided by NASA to its engineers. The NASA Academy of Program and Project Leadership (APPL) offers core classes in program and project management and systems engineering to NASA employees with the purpose of creating a 'knowledge community where ideas, skills, and experiences are exchanged to increase each other's capacity for strong leadership'. The SLI program is, in one sense, a combination of a conceptual design program and a technology program. The program as a whole doesn't map into the generic systems engineering project cycle as currently, and for some time, taught. For example, the NASA APPL Systems Engineering training course teaches that the 'first step in developing an architecture is to define the external boundaries of the system', which will require definition of the interfaces with other systems and the

next step will be to 'define all the components that make up the next lower level of the system hierarchy' where fundamental requirements are allocated to each component. Whereas, the SLI technology risk reduction approach develops architecture subsystem technologies prior to developing architectures. The higher level architecture requirements are not allowed to fully develop and undergo decomposition and allocation down to the subsystems before the subsystems must develop allocated requirements based on the highest level of requirements. In the vernacular of the project cycles prior to the mid 1990's, the architecture definition portion of the program appears to be at a generic Phase A stage, while the subsystems are operating at Phase B. Even the management structure of the SLI program is innovative in its approach to Systems Engineering and is not reflected in the APPL training modules. The SLI program has established a Systems Engineering office as an office separate from the architecture development or the subsystem technology development, while that office does have representatives within these other offices. The distributed resources of the Systems Engineering Office are co-located with the respective Project Offices. This template is intended to provide systems engineering as an integrated function at the Program Level. the program management of SLI and the MAT agree that 'program/project managers and the systems engineering team must work closely together towards the single objective of delivering quality products that meet the customer needs'. This paper will explore the differences between the methods being taught by NASA, which represent decades of ideas, and those currently in practice in SLI. Time will tell if the innovation employed by SLI will prove to be the model of the future. For now, it is suggested that the training of the present exercise the flexibility of recognizing the new processes employed by a major new NASA program.
Author
Reusable Launch Vehicles; Spacecraft Launching; Management Methods; Systems Engineering

20020019044 NASA Marshall Space Flight Center, Huntsville, AL USA
Lightweight Chambers for Thrust Assemblies
Elam, S.; Lee, J.; Holmes, R.; Zimmerman, F.; Turner, Jim, Technical Monitor; Jul. 12, 2001; In English; 52nd International Astronautical Congress, 1-5 Oct. 2001, Toulouse, France
Contract(s)/Grant(s): NRA8-21
Report No.(s): IAF-01-S.3.05; Copyright; Avail: Other Sources

New materials and fabrication techniques have been successfully used to create actively cooled thrust chambers that weigh less, and potentially operate hotter, than conventional designs. Results could improve the performance and payload capacity of many vehicles, including new Reusable Launch Vehicle (RLV) concepts. Polymer Matrix Composite (PMC) and Metal Matrix Composite (MMC) materials replaced traditional metal alloys for structural jackets and manifolds. Ceramic Matrix Composite (CMC) materials and an advanced copper alloy (GRCop-84) were used for chamber liners to significantly increase operating temperatures. Several small chambers were fabricated and tested with promising results to demonstrate that designs over 50% lighter are feasible. New fabrication techniques were demonstrated, including an advanced casting technique and a low cost vacuum plasma spray process. Based on the favorable results for the MMC, PMC and GRCop-84 materials, similar concepts were applied to larger chamber designs. These larger units are currently being fabricated for testing in 2001. Hot-fire testing on the larger units will directly compare performance results between a conventional chamber design and these 'lightweight' alternatives. The technology developed and demonstrated by this effort could benefit new RLV programs, as well as other existing and future engine designs.
Author
Engine Design; Thrust Chambers; Cooling Systems; Design Analysis; Metal Matrix Composites; Combustion Chambers

20020017580 NASA Marshall Space Flight Center, Huntsville, AL USA
X-33 XRS-2200 Linear Aerospike Engine Sea Level Plume Radiation
DAgostino, Mark G.; Lee, Young C.; Wang, Ten-See; Turner, Jim, Technical Monitor; [2001]; In English; 26th JANNAF Exhaust Plume Technology Subcommittee Meeting, 5-9 Nov. 2001, San Antonio, TX, USA; No Copyright; Avail: CASI; A02, Hardcopy

Wide band plume radiation data were collected during ten sea level tests of a single XRS-2200 engine at the NASA Stennis Space Center in 1999 and 2000. The XRS-2200 is a liquid hydrogen/liquid oxygen fueled, gas generator cycle linear aerospike engine which develops 204,420 lbf thrust at sea level. Instrumentation consisted of six hemispherical radiometers and one narrow view radiometer. Test conditions varied from 100% to 57% power level (PL) and 6.0 to 4.5 oxidizer to fuel (O/F) ratio. Measured radiation rates generally increased with engine chamber pressure and mixture ratio. One hundred percent power level radiation data were compared to predictions made with the FDNS and GASRAD codes. Predicted levels ranged from 42% over to 7% under average test values.
Author
Aerospike Engines; Plumes; X-33 Reusable Launch Vehicle; Heat Flux

20020016181 NASA Marshall Space Flight Center, Huntsville, AL USA
An Approach to Establishing System Benefits for Technology in NASA's Hypersonics Investment Area
Hueter, Uwe; Pannell, Bill; Cook, Stephen, Technical Monitor; Aug. 17, 2001; In English; 52nd International Astronautical Congress, 1-5 Oct. 2001, Toulouse, France
Report No.(s): IAF-01-V.4.05; No Copyright; Avail: CASI; A02, Hardcopy

NASA's has established long term goals for access-to-space. The third generation launch systems are to be fully reusable and operational around 2025. The goals for the third generation launch system are to significantly reduce cost and improve safety over current systems. The Advanced Space Transportation Program (ASTP) Office at the NASA's Marshall Space Flight Center in Huntsville, AL has the agency lead to develop space transportation technologies. Within ASTP, under the Hypersonics Investment Area, third generation technologies are being pursued. The Hypersonics Investment Area's primary objective is to mature vehicle technologies to enable substantial increases in the design and operating margins of third generation RLVs (current Space Shuttle is considered the first generation RLV) by incorporating advanced propulsion systems, materials, structures, thermal protection systems, power, and avionics technologies. The paper describes the system process, tools and concepts used to determine the technology benefits. Preliminary results will be presented along with the current technology investments that are being made by ASTP's Hypersonics Investment Area.
Author
Hypersonics; Space Transportation; Technology Assessment; Space Shuttles; Propulsion System Performance; NASA Space Programs; Avionics

20020012536 NASA Glenn Research Center, Cleveland, OH USA
Testing of Densified Liquid Hydrogen Stratification in a Scale Model Propellant Tank
Jurns, John M.; Tomsik, Thomas M.; Greene, William D.; September 2001; In English; 1999 Cryogenic Engineering and International Cryogenic Materials Conference, 12-16 Jul. 1999, Montreal, Quebec, Canada
Contract(s)/Grant(s): RTOP 721-20-00
Report No.(s): NASA/TM-2001-209391; NAS 1.15:209391; E-11820; No Copyright; Avail: CASI; A03, Hardcopy

This paper describes a test program that was conducted at NASA to demonstrate the ability to load densified LH2 into a subscale propellant tank. This work was done through a collaborative effort between NASA Glenn Research Center and the Lockheed Martin Michoud Space Systems (LMMSS). The Multilobe tank, which was made from composite materials similar to that to be used on X-33, was formed from two lobes with a center septum. Test results are shown for data that was collected on filling the subscale tank with densified liquid hydrogen (DLH2) propellant that was produced at the NASA Plum Brook Station. Data is compared to analytical predictions. Data collected for this test series agrees well with analytical predictions of the environmental heat leak into the tank and the thermal stratification characteristics of the hydrogen propellant in the tank as it was filled with DLH2.
Author
Liquid Hydrogen; Propellant Tanks; Scale Models; Stratification; Cryogenic Fluid Storage; Performance Tests

20020012437 NASA Ames Research Center, Moffett Field, CA USA
The Evolution of Flexible Insulation as Thermal Protection Systems for Reusable Launch Vehicles: AFRSI (Advanced Flexible Reusable Surface Insulation) to CRI (Conformal Reusable Insulation)
Rezin, Marc; Oka, Kris; Arnold, Jim, Technical Monitor; [2001]; In English; National Space and Missile Materials Symposium, 25 Jun. 2001; No Copyright; Avail: CASI; A03, Hardcopy

This viewgraph presentation gives an overview of the evolution of flexible insulation as a thermal protection system for reusable launch vehicles, Advanced Flexible Reusable Surface Insulation (AFRSI) to Conformal Reusable Insulation (CRI). Details are given on the approved use of AFRIS on the Shuttle Orbiter in June 1980, the first flight of AFRIS on STS-6, windward blanket development, composite flexible blanket insulation, and flight demonstrations.
CASI
Thermal Insulation; Space Shuttle Orbiters

20020008402 NASA, Washington, DC USA
NASA IVHM Technology Experiment for X-vehicles (NITEX)
Sandra, Hayden; Bajwa, Anupa; [2001]; In English; No Copyright; Avail: CASI; A01, Hardcopy

The purpose of the NASA IVHM Technology Experiment for X-vehicles (NITEX) is to advance the development of selected IVHM technologies in a flight environment and to demonstrate the potential for reusable launch vehicle ground

processing savings. The technologies to be developed and demonstrated include system-level and detailed diagnostics for real-time fault detection and isolation, prognostics for fault prediction, automated maintenance planning based on diagnostic and prognostic results, and a microelectronics hardware platform. Complete flight The Evolution of Flexible Insulation as IVHM consists of advanced sensors, distributed data acquisition, data processing that includes model-based diagnostics, prognostics and vehicle autonomy for control or suggested action, and advanced data storage. Complete ground IVHM consists of evolved control room architectures, advanced applications including automated maintenance planning and automated ground support equipment. This experiment will advance the development of a subset of complete IVHM.
Derived from text
Reusable Launch Vehicles; Diagnosis; Distributed Processing; Fault Detection; Ground Based Control; Real Time Operation; Technology Assessment

20010110022 Oklahoma State Univ., Stillwater, OK USA
Using Neural Networks in Decision Making for a Reconfigurable Electro Mechanical Actuator (EMA)
Latino, Carl D.; 2000 Research Reports: NASA/ASEE Summer Faculty Fellowship Program; October 2001; In English; No Copyright; Avail: CASI; A01, Hardcopy

The objectives of this project were to demonstrate applicability and advantages of a neural network approach for evaluating the performance of an electro-mechanical actuator (EMA). The EMA in question was intended for the X-37 Advanced Technology Vehicle. It will have redundant components for safety and reliability. The neural networks for this application are to monitor the operation of the redundant electronics that control the actuator in real time and decide on the operating configuration. The system we proposed consists of the actuator, sensors, control circuitry and dedicated (embedded) processors. The main purpose of the study was to develop suitable hardware and neural network capable of allowing real time reconfiguration decisions to be made. This approach was to be compared to other methods such as fuzzy logic and knowledge based systems considered for the same application. Over the course of the project a more general objective was the identification of the other neural network applications and the education of interested NASA personnel on the topic of Neural Networks.
Author
Electromechanical Devices; Actuators; Neural Nets; Decision Making; Redundant Components; Real Time Operation

20010106933 NASA Ames Research Center, Moffett Field, CA USA
Wireless Subsurface Sensors for Health Monitoring of Thermal Protection Systems on Hypersonic Vehicles
Milos, Frank S.; Arnold, Jim, Technical Monitor; [2001]; In English; 3rd International Workshop on Structural Health Monitoring, 12-14 Sep. 2001, Stanford, CA, USA; No Copyright; Avail: Other Sources; Abstract Only

Health diagnostics is an area where major improvements have been identified for potential implementation into the design of new reusable launch vehicles (RLVs) in order to reduce life cycle costs, to increase safety margins, and to improve mission reliability. NASA Ames is leading the effort to develop inspection and health management technologies for thermal protection systems. This paper summarizes a joint project between NASA Ames and industry partners to develop 'wireless' devices that can be embedded in the thermal protection system to monitor temperature or other quantities of interest. These devices are sensors integrated with radio-frequency identification (RFID) microchips to enable non-contact communication of sensor data to an external reader that may be a hand-held scanner or a large portal. Both passive and active prototype devices have been developed. The passive device uses a thermal fuse to indicate the occurrence of excessive temperature. This device has a diameter under 0.13 cm. (suitable for placement in gaps between ceramic TPS tiles on an RLV) and can withstand 370 C for 15 minutes. The active device contains a small battery to provide power to a thermocouple for recording a temperature history during flight. The bulk of the device must be placed beneath the TPS for protection from high temperature, but the thermocouple can be placed in a hot location such as near the external surface.
Author
Thermal Protection; Hypersonic Vehicles; Reusable Launch Vehicles; Diagnosis

20010106932 NASA Ames Research Center, Moffett Field, CA USA
Thermal Protection Materials for Reentry and Planetary Applications
Johnson, Sylvia M.; Arnold, Jim, Technical Monitor; [2001]; In English; Conference on Structural Ceramics and Ceramic Composites for High Temperature Applications, Oct. 2001, Spain; No Copyright; Avail: Other Sources; Abstract Only

Thermal protection materials and systems (TPS) are used to protect spacecraft during reentry into Earth's atmosphere or entry into planetary atmospheres. As such, these materials are subject to severe environments with high heat fluxes and rapid

heating. Catalytic effects can increase the temperatures substantially. These materials are also subject to impact damage from micrometeorites or other debris during ascent, orbit, and descent, and thus must be able to withstand damage and function following damage. Thermal protection materials and coatings used in reusable launch vehicles will be reviewed, including the needs and directions for new materials to enable new missions that require faster turnaround and much greater reusability. The role of ablative materials for use in high heat flux environments, especially for non-reusable applications and upcoming planetary missions, will be discussed. New thermal protection system materials may enable the use of sharp nose caps and leading edges on future reusable space transportation vehicles. Vehicles employing this new technology would have significant increases in maneuverability and out-of-orbit cross range compared to current vehicles, leading to increased mission safety in the event of the need to abort during ascent or from orbit. Ultrahigh temperature ceramics, a family of materials based on HfB2and ZrB2 with SiC, will be discussed. The development, mechanical and thermal properties, and uses of these materials will be reviewed.
Author
Thermal Protection; Heat Flux; Reusable Launch Vehicles; Atmospheric Entry; Heating; Planetary Atmospheres

20010098302 NASA Langley Research Center, Hampton, VA USA
Structures for the 3rd Generation Reusable Concept Vehicle
Hrinda, Glenn A.; [2001]; In English; AIAA Space 2001 Conference and Exposition, 28-30 Aug. 2001, Albuquerque, NM, USA; Copyright; Avail: CASI; A02, Hardcopy

A major goal of NASA is to create an advance space transportation system that provides a safe, affordable highway through the air and into space. The long-term plans are to reduce the risk of crew loss to 1 in 1,000,000 missions and reduce the cost of Low-Earth Orbit by a factor of 100 from today's costs. A third generation reusable concept vehicle (RCV) was developed to assess technologies required to meet NASA's space access goals. The vehicle will launch from Cape Kennedy carrying a 25,000 lb. payload to the International Space Station (ISS). The system is an air breathing launch vehicle (ABLV) hypersonic lifting body with rockets and uses triple point hydrogen and liquid oxygen propellant. The focus of this paper is on the structural concepts and analysis methods used in developing the third generation reusable launch vehicle (RLV). Member sizes, concepts and material selections will be discussed as well as analysis methods used in optimizing the structure. Analysis based on the HyperSizer structural sizing software will be discussed. Design trades required to optimize structural weight will be presented.
Author
Structural Design; Reusable Spacecraft; Spacecraft Design; Design Optimization; Structural Analysis

20010097314 NASA Ames Research Center, Moffett Field, CA USA
Wireless Subsurface Microsensors for Health Monitoring of Thermal Protection Systems on Hypersonic Vehicles
Milos, Frank S.; Watters, David G.; Pallix, Joan B.; Bahr, Alfred J.; Huestis, David L.; Arnold, Jim, Technical Monitor; [2001]; In English; SPIE 6th Annual International Symposium on Health Monitoring and Diagnostics, 4-8 Mar. 2001, Newport Beach, CA, USA
Contract(s)/Grant(s): NAS2-99092; NAS2-10730; NAS2-10731; No Copyright; Avail: CASI; A02, Hardcopy

Health diagnostics is an area where major improvements have been identified for potential implementation into the design of new reusable launch vehicles in order to reduce life cycle costs, to increase safety margins, and to improve mission reliability. NASA Ames is leading the effort to develop inspection and health management technologies for thermal protection systems. This paper summarizes a joint project between NASA Ames and SRI International to develop 'SensorTags,' radio frequency identification devices coupled with event-recording sensors, that can be embedded in the thermal protection system to monitor temperature or other quantities of interest. Two prototype SensorTag designs containing thermal fuses to indicate a temperature overlimit are presented and discussed.
Author
Damage Assessment; Microinstrumentation; Hypersonic Vehicles; Smart Materials; Thermal Protection; Wireless Communication

20010095443 Science Applications International Corp., Huntsville, AL USA
Pathfinder Technologies Specialist, X-37
French, James R.; Sep. 30, 2001; In English
Contract(s)/Grant(s): NAS8-99060; No Copyright; Avail: CASI; A04, Hardcopy

The X-37 is a technology demonstrator sponsored by NASA. It includes a number of experiments both imbedded (i.e.,

essential aspects of the vehicle) and separate. The technologies demonstrated will be useful in future operational versions as well as having broad applications to other programs. Mr. James R. French, of JRF Engineering Services and as a consultant to SAIC, has provided technical support to the X-37 NASA Program office since the beginning of the program. In providing this service, Mr. French has maintained close contact with the Boeing Seal Beach and Rocketdyne technical teams via telephone, e-mail, and periodic visits. His interfaces were primarily with the working engineers in order to provide NASA sponsors with a different view than that achieved through management channels. Mr. French's periodic and highly detailed technical reports were submitted to NASA and SAIC (Science Applications International Corporation) on a weekly/monthly basis. These reports addressed a wide spectrum of programmatic and technical interests related to the X-37 Program including vehicle design, flight sciences, propulsion, thermal protection, Guidance Navigation & Control (GN&C), structures, and operations. This deliverable is presented as a consolidation of the twelve monthly reports submitted during the Contract's Option Year,
Author
X-37 Vehicle; Research and Development; NASA Programs; Contract Management; Technology Utilization

20010095419 Science Applications International Corp., Huntsville, AL USA
Space Transportation Systems Technologies
Laue, Jay H.; Sep. 30, 2001; In English
Contract(s)/Grant(s): NAS8-99060; NRA8-21; No Copyright; Avail: CASI; A05, Hardcopy

This document is the final report by the Science Applications International Corporation (SAIC) on contracted support provided to the National Aeronautics and Space Administration (NASA) under Contract NAS8-99060, 'Space Transportation Systems Technologies'. This contract, initiated by NASA's Marshall Space Flight Center (MSFC) on February 8, 1999, was focused on space systems technologies that directly support NASA's space flight goals. It was awarded as a Cost-Plus-Incentive-Fee (CPIF) contract to SAIC, following a competitive procurement via NASA Research Announcement, NRA 8-21. This NRA was specifically focused on tasks related to Reusable Launch Vehicles (RLVs). Through Task Area 3 (TA-3), 'Other Related Technology' of this NRA contract, SAIC extensively supported the Space Transportation Directorate of MSFC in effectively directing, integrating, and setting its mission, operations, and safety priorities for future RLV-focused space flight. Following an initially contracted Base Year (February 8, 1999 through September 30, 1999), two option years were added to the contract. These were Option Year 1 (October 1, 1999 through September 30, 2000) and Option Year 2 (October 1, 2000 through September 30, 2001). This report overviews SAIC's accomplishments for the Base Year, Option Year 1, and Option Year 2, and summarizes the support provided by SAIC to the Space Transportation Directorate, NASA/MSFC.
Author
Aerospace Systems; Reusable Launch Vehicles; Space Transportation System

20010092480 NASA Glenn Research Center, Cleveland, OH USA
Performance Evaluation of the NASA GTX RBCC Flowpath
Thomas, Scott R.; Palac, Donald T.; Trefny, Charles J.; Roche, Joseph M.; July 2001; In English; Fifteenth International Symposium on Airbreathing Engines, 2-7 Sep. 2001, Bangalore, India; Original contains color illustrations
Contract(s)/Grant(s): RTOP 708-73-20
Report No.(s): NASA/TM-2001-210953; E-12807; NAS 1.15:210953; ISABE-2001-1070; No Copyright; Avail: CASI; A02, Hardcopy

The NASA Glenn Research Center serves as NASAs lead center for aeropropulsion. Several programs are underway to explore revolutionary airbreathing propulsion systems in response to the challenge of reducing the cost of space transportation. Concepts being investigated include rocket-based combined cycle (RBCC), pulse detonation wave, and turbine-based combined cycle (TBCC) engines. The GTX concept is a vertical launched, horizontal landing, single stage to orbit (SSTO) vehicle utilizing RBCC engines. The propulsion pod has a nearly half-axisymmetric flowpath that incorporates a rocket and ram-scramjet. The engine system operates from lift-off up to above Mach 10, at which point the airbreathing engine flowpath is closed off, and the rocket alone powers the vehicle to orbit. The paper presents an overview of the research efforts supporting the development of this RBCC propulsion system. The experimental efforts of this program consist of a series of test rigs. Each rig is focused on development and optimization of the flowpath over a specific operating mode of the engine. These rigs collectively establish propulsion system performance over all modes of operation, therefore, covering the entire speed range. Computational Fluid Mechanics (CFD) analysis is an important element of the GTX propulsion system development and validation. These efforts guide experiments and flowpath design, provide insight into experimental data, and extend results to

conditions and scales not achievable in ground test facilities. Some examples of important CFD results are presented.
Author
Air Breathing Engines; Computational Fluid Dynamics; Propulsion System Configurations; Propulsion System Performance

20010091019 NASA Ames Research Center, Moffett Field, CA USA
Health Monitoring Technology for Thermal Protection Systems on Reusable Hypersonic Vehicles
Milos, Frank S.; Watters, D. G.; Heinemann, J. M.; Karunaratne, K. S.; Arnold, Jim, Technical Monitor; [2001]; In English; National Space and Missile Materials Symposium, 25-28 Jun. 2001, Monterey, CA, USA; No Copyright; Avail: CASI; A03, Hardcopy

Integrated subsystem health diagnostics is an area where major improvements have been identified for potential implementation into the design of new reusable launch vehicles (RLVs) in order to reduce life cycle costs, to increase safety margins, and to improve mission reliability. This talk summarizes a joint effort between NASA Ames and industry partners to develop rapid non-contact diagnostic tools for health and performance monitoring of thermal protection systems (TPS) on future RLVs. The specific goals for TPS health monitoring are to increase the speed and reliability of TPS inspections for improved operability at lower cost. The technology being developed includes a 3-D laser scanner for examining the exterior surface of the TPS, and a subsurface microsensor suite for monitoring the health and performance of the TPS. The sensor suite consists of passive overlimit sensors and sensors for continuous parameter monitoring in flight. The sensors are integrated with radio-frequency identification (RFID) microchips to enable wireless communication of-the sensor data to an external reader that may be a hand-held scanner or a large portal. Prototypes of the laser system and both types of subsurface sensors have been developed. The laser scanner was tested on Shuttle Orbiter Columbia and was able to dimension surface chips and holes on a variety of TPS materials. The temperature-overlimit microsensor has a diameter under 0.05 inch (suitable for placement in gaps between ceramic TPS tiles) and can withstand 700 F for 15 minutes.
Author
Thermal Protection; Microinstrumentation; Diagnosis; Chips (Electronics); Ceramics; Systems Health Monitoring

20010086424 Science Applications International Corp., Huntsville, AL USA
Second Generation RLV Program
Laue, Jay; Feb. 26, 2001; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS8-99060; NRA8-21
Report No.(s): NAS8-99060-MPR26; No Copyright; Avail: CASI; A03, Hardcopy

During the time period covered by this report, SAIC: 1) Continued to develop and assess processes and approaches that can be applied to Second Generation Reuseable Launch Vehicles (RLV) technologies prioritization. An approach based on the use of analytic Saaty scale functions has been defined and is being investigated. 2) Planned and facilitated technologies prioritization workshops, supported development of systems program algorithms based on the concept of influence diagramming, and assessment of analogies between aircraft and space systems developments. 3) Video interviews held with X-37 personnel at Dryden. The CD-ROM is being concluded and a near-final review disc is expected soon. 4) CD-ROMS were produced by Engineered Multimedia, Inc. (EMI) under subcontract to SAIC. These two CD-ROM products, ‘Microgravity’ and ‘New Horizons’, were delivered to MSFS technical representatives at a briefing at SAID on January 18, 2001. 5) Presentation brochure, ‘Aviation/Space Analog Team Interim Report’ was provided to Space Propulsion Synergy Team (SPST) members and to NASA personnel. Wrap-up of effort will be a mid-April briefing to both MSFC and the full SPST membership. Support to Phase 2 of CCPD ends 3/31/01. Phase 3 continuation effort planned with emphasis on technical info content. 6) Recommended that eliminating the B-52 flights of the X-37 in favor of alternate approached be evaluated. Discovered that a required change to the thruster valves of the X-37 had not been made. Repair work continues on the lower fuselage section of the X-37, in the areas that experienced core collapse. The currently catalogued potential weight increases and decreases are about equal, but this does not include the impact of the cable weight underestimate reported earlier. The CFD for RCS testing correlates well with the wind tunnel data. 7) The Level IV CCB approved SCN-4 to the MC-1 Engine SPecification. Approved changes have been incorporated into the specification, and Revision D has been released to the Repository. Billy Gonterman, SAIC, traveled to SSC to perform a five-day readiness assessment on the progress of facilities activation. Implemented modifications to the MC-1 Engine TPA Component Database. 8) Replaced 25 Combined-Cycle Propulsion Database (CCPD) documents; modified the low-on screen to include a direct email link for user contact; prepared a summary presentation of current and proposed tasks.
Derived from text
Reusable Launch Vehicles; Microgravity; X-37 Vehicle; Combined Cycle Power Generation; Spacecraft Propulsion; Systems Engineering; Rocket Engines

20010084776 NASA Dryden Flight Research Center, Edwards, CA USA
Full Envelope Reconfigurable Control Design for the X-33 Vehicle
Cotting, M. Christopher; Burken, John J.; Lee, Seung-Hee, Technical Monitor; [2001]; In English; AIAA Guidance, Navigation and Control Conference, 6-10 Aug. 2001, Montreal, Quebec, Canada
Report No.(s): AIAA Paper 2001-1234; Copyright; Avail: CASI; A02, Hardcopy

In the event of a control surface failure, the purpose of a reconfigurable control system is to redistribute the control effort among the remaining working surfaces such that satisfactory stability and performance are retained. An Off-line Nonlinear General Constrained Optimization (ONCO) approach was used for the reconfigurable X-33 control design method. Three example failures are shown using a high fidelity 6 DOF simulation (case I ascent with a left body flap jammed at 25 deg.; case 2 entry with a right inboard elevon jam at 25 deg.; and case 3, landing (TAEM) with a left rudder jam at -30 deg.) Failure comparisons between responses with the nominal controller and reconfigurable controllers show the benefits of reconfiguration. Single jam aerosurface failures were considered, and failure detection and identification is considered accomplished in the actuator controller. The X-33 flight control system will incorporate reconfigurable flight control in the baseline system.
Author
X-33 Reusable Launch Vehicle; Flight Control; Control Systems Design; Failure; Design Analysis; Aerodynamic Configurations; Aircraft Design; Controllers

20010081320 NASA Dryden Flight Research Center, Edwards, CA USA
Reconfigurable Control Design for the Full X-33 Flight Envelope
Cotting, M. Christopher; Burken, John J.; August 2001; In English; AIAA Guidance, Navigation and Control Conference, 6-10 Aug. 2001, Montreal, Quebec, Canada
Contract(s)/Grant(s): RTOP 715-33-02-E8-23
Report No.(s): NASA/TM-2001-210396; H-2458; NAS 1.15:210396; AIAA Paper 2001-4379; No Copyright; Avail: CASI; A03, Hardcopy

A reconfigurable control law for the full X-33 flight envelope has been designed to accommodate a failed control surface and redistribute the control effort among the remaining working surfaces to retain satisfactory stability and performance. An offline nonlinear constrained optimization approach has been used for the X-33 reconfigurable control design method. Using a nonlinear, six-degree-of-freedom simulation, three example failures are evaluated: ascent with a left body flap jammed at maximum deflection; entry with a right inboard elevon jammed at maximum deflection; and landing with a left rudder jammed at maximum deflection. Failure detection and identification are accomplished in the actuator controller. Failure response comparisons between the nominal control mixer and the reconfigurable control subsystem (mixer) show the benefits of reconfiguration. Single aerosurface jamming failures are considered. The cases evaluated are representative of the study conducted to prove the adequate and safe performance of the reconfigurable control mixer throughout the full flight envelope. The X-33 flight control system incorporates reconfigurable flight control in the existing baseline system.
Author
X-33 Reusable Launch Vehicle; Configuration Management; Control Systems Design; Flight Envelopes; Flight Control; Control Theory

20010080460 NASA Dryden Flight Research Center, Edwards, CA USA
A Monte Carlo Dispersion Analysis of the X-33 Simulation Software
Williams, Peggy S.; [2001]; In English; AIAA Atmospheric Flight Mechanics Conference, 6-9 Aug. 2001, Montreal, Canada
Report No.(s): AIAA Paper 2001-4067; Copyright; Avail: CASI; A03, Hardcopy

A Monte Carlo dispersion analysis has been completed on the X-33 software simulation. The simulation is based on a preliminary version of the software and is primarily used in an effort to define and refine how a Monte Carlo dispersion analysis would have been done on the final flight-ready version of the software. This report gives an overview of the processes used in the implementation of the dispersions and describes the methods used to accomplish the Monte Carlo analysis. Selected results from 1000 Monte Carlo runs are presented with suggestions for improvements in future work.
Author
Computerized Simulation; Monte Carlo Method; Computer Programs; Software Development Tools; Dispersion

20010080458 NASA Marshall Space Flight Center, Huntsville, AL USA
Evaluation of Microcracking in Two Carbon-Fiber/Epoxy-Matrix Composite Cryogenic Tanks
Hodge, A. J.; August 2001; In English; Original contains color illustrations
Report No.(s): NASA/TM-2001-211194; M-1024; NAS 1.15:211194; No Copyright; Avail: CASI; A03, Hardcopy

Two graphite/epoxy cryogenic pressure vessels were evaluated for microcracking. The X-33 LH2 tank lobe skins were extensively examined for microcracks. Specimens were removed from the inner skin of the X-33 tank for tensile testing. The data obtained from these tests were used to model expected microcrack density as a function of stress. Additionally, the laminate used in the Marshall Space Flight Center (MSFC) Composite Conformal, Cryogenic, Common Bulkhead, Aerogel-Insulated Tank (CBAT) was evaluated. Testing was performed in an attempt to predict potential microcracking during testing of the CBAT.
Author
Graphite-Epoxy Composites; Pressure Vessels; X-33 Reusable Launch Vehicle; Storage Tanks; Cryogenic Fluid Storage; Microcracks; Liquid Hydrogen

20010079105 NASA Marshall Space Flight Center, Huntsville, AL USA
National Rocket Propulsion Materials Plan: A NASA, Department of Defense, and Industry Partnership
Clinton, Raymond G., Jr.; Munafo, Paul M., Technical Monitor; [2001]; In English; 2001 ASM/TMS Spring Symposium, 1-2 May 2001, Schenectady, NY, USA; No Copyright; Avail: Other Sources; Abstract Only

NASA, Department of Defense, and rocket propulsion industry representatives are working together to create a national rocket propulsion materials development roadmap. This ‘living document’ will facilitate collaboration among the partners, leveraging of resources, and will be a highly effective tool for technology development planning. The structuring of the roadmap, and development plan, which will combine the significant efforts of the Integrated High Payoff Rocket Propulsion Technology (IHPRPT) Program, and NASA’s Integrated Space Transportation Plan (ISTP), is being lead by the IHPRPT Materials Working Group (IMWG). The IHPRPT Program is a joint DoD, NASA, and industry effort to dramatically improve the nation’s rocket propulsion capabilities. This phased program is structured with increasingly challenging goals focused on performance, reliability, and cost to effectively double rocket propulsion capabilities by 2010. The IHPRPT program is focused on three propulsion application areas: Boost and Orbit Transfer (both liquid rocket engines and solid rocket motors), Tactical, and Spacecraft. Critical to the success of this initiative is the development and application of advanced materials, processes, and manufacturing technologies. NASA’s ISTP is a comprehensive strategy focusing on the aggressive safety, reliability, and affordability goals for future space transportation systems established by the agency. Key elements of this plan are the 2 nd and 3 d Generation Reusable Launch Vehicles (RLV). The affordability and safety goals of these generational systems are, respectively, 10X cheaper and 100X safer by 2010, and 100X cheaper and 10,000X safer by 2025. Accomplishment of these goals requires dramatic and sustained breakthroughs, particularly in the development and the application of advanced material systems. The presentation will provide an overview of the IHPRPT materials initiatives, NASA’s 2nd and 3 rd Generation RLV propulsion materials projects, and the approach for the development of the national rocket propulsion materials roadmap.
Author
Technological Forecasting; Solid Propellant Rocket Engines; Safety; Reliability; Propulsion; Planning; Liquid Propellant Rocket Engines

20010071334 NASA Langley Research Center, Hampton, VA USA
High Temperature Polyimide Materials in Extreme Temperature Environments
Johnson, Theodore F.; Gates, Thomas S.; [2001]; In English; 42nd AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference and Exhibit, 16-19 Apr. 2001, Seattle, WA, USA
Report No.(s): AIAA Paper 2001-1214; Copyright; Avail: CASI; A03, Hardcopy

At the end of the NASA High Speed Research (HSR) Program, NASA Langley Research Center (LaRC) began a program to screen the high-temperature Polymeric Composite Materials (PMCs) characterized by the HSR Durability Program for possible use in Reusable Launch Vehicles (RLVs) operating under extreme temperature conditions. The HSR Program focused on developing material-related technologies to enable a High Speed Civil Transport (HSCT) capable of operating temperatures ranging from 54 C (-65 F) to 177 C (350 F). A high-temperature polymeric resin, PETI-5 was used in the HSR Program to satisfy the requirements for performance and durability for a PMC. For RLVs, it was anticipated that this high temperature material would contribute to reducing the overall weight of a vehicle by eliminating or reducing the thermal protection required to protect the internal structural elements of the vehicle and increasing the structural strain limits. The tests were performed to determine temperature-dependent mechanical and physical proper-ties of IM7/PETI-5 composite over a temperature range from cryogenic temperature -253 C (-423F) to the material’s maximum use temperature of 230 C (450 F). This paper presents results from the test program for the temperature-dependent mechanical and physical properties of IM7/PETI-5 composite in the temperature range from -253 C (-423 F) to 27 C (80 F).
Author
High Temperature; Reusable Launch Vehicles; Polyimides; Refractory Materials; Supersonic Transports; Cryogenic Temperature

20010067392 NASA Marshall Space Flight Center, Huntsville, AL USA

Predictor-Corrector Entry Guidance for Reusable Launch Vehicles

Youssef, Hussein; Chowdhry, Rajiv; Lee, Howard; Zimmerman, Curtis; Brandon, Larry, Technical Monitor; [2001]; In English; AIAA GNC Conference, 6-9 Aug. 2001, Montreal, Canada

Contract(s)/Grant(s): NCC8-115; No Copyright; Avail: CASI; A02, Hardcopy

An online entry guidance algorithm has been developed using a predictor-corrector approach. The algorithm is designed for the Reusable Launch Vehicle (RLV) and is demonstrated by using, the X-33 model. The objective of the design is to handle widely dispersed entry conditions and deliver the vehicle at the Terminal Area Energy Management (TAEM) interface box within an acceptable tolerance and without violating any of the vehicle physical constraints. Combination of several control variables is used in testing the performance and computational requirement of the algorithm. The control variables are the bank angle, angle-of-attack and the time for roll reversal. The bank angle and angle-of-attack profiles are the nominal profiles plus the perturbations in each direction. The initial guess of the bank profile is a 45 degrees bank angle with reversal at 360 seconds from liftoff. A six-element state vector is propagated to the TAEM interface box through the integration of the equations of motion (EOM). Altitude, heading and range errors are computed between the desired and the achieved state at the TAEM interface. These errors are used to correct the initial guess of the control variables. This process is repeated until the errors meet an acceptable level at the TAEM interface. Several numerical optimization methods are used to evaluate the convergent property of the predictor-predictor methodology. Successful results are demonstrated using the X-33 model.

Author

X-33 Reusable Launch Vehicle; Predictor-Corrector Methods; Numerical Analysis; Algorithms

20010067274 NASA Marshall Space Flight Center, Huntsville, AL USA

X-33 LH2 Tank Failure Investigation Findings

Niedermeyer, M.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 liquid hydrogen tank failure investigation found the following: (1) The inner skin microcracked and hydrogen infiltrated into it; (2) The cracks grew larger under pressure; (3) When pressure was removed, the cracks closed slightly; (4) When the tank was drained and warmed, the cracks closed and blocked the leak path; (5) Foreign object debris (FOD) and debond areas provided an opportunity for a leak path; and (6) There is still hydrogen in the other three lobes today.

Derived from text

Microcracks; X-33 Reusable Launch Vehicle; Accident Investigation; Propellant Tanks

20010067271 NASA Marshall Space Flight Center, Huntsville, AL USA

Development of Metal Matrix Composites for NASA's Advanced Propulsion Systems

Lee, J.; Elam, S.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; No Copyright; Avail: CASI; A02, Hardcopy

The state-of-the-art development of several Metal Matrix Composites (MMC) for NASA's advanced propulsion systems will be presented. The goal is to provide an overview of NASA-Marshall Space Flight Center's on-going activities in MMC components for advanced liquid rocket engines such as the X-33 vehicle's Aerospike engine and X-34's Fastrac engine. The focus will be on lightweight, low cost, and environmental compatibility with oxygen and hydrogen of key MMC materials, within each of NASA's new propulsion application, that will provide a high payoff for NASA's Reusable Launch Vehicles and space access vehicles. In order to fabricate structures from MMC, effective joining methods must be developed to join MMC to the same or to different monolithic alloys. Therefore, a qualitative assessment of MMC's welding and joining techniques will be outlined.

Author

Metal Matrix Composites; Welding; Joints (Junctions)

20010067262 NASA Marshall Space Flight Center, Huntsville, AL USA

Composite,Cryogenic, Conformal, Common Bulkhead, Aerogel-Insulated Tank (CBAT)

Roberts, J. K.; Kovach, M. P.; McMahon, W. M.; Finckenor, J. L.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; Copyright; Avail: Other Sources

The objective of the Composite, Cryogenic, Conformal, Common Bulkhead, Aerogel-insulated Tank (CBAT) Program is to evaluate the potential for using various new technologies in next generation Reusable Launch Vehicles (RLVs) through design, fabrication, and testing of a subscale system. The new technologies include polymer matrix composites (PMCs),

conformal propellant storage, common bulkhead packaging, and aerogel insulation. The National Aeronautics and Space Administration (NASA) and Thiokol Propulsion from Cordant Technologies are working together to develop a design and the processing methodologies which will allow integration of these technologies into a single structural component assembly. Such integration will significantly decrease subsystem weight and reduce shape, volume, and placement restrictions, thereby enhancing overall launch system performance. This paper/presentation focuses on the challenges related to materials and processes that were encountered and overcome during this program to date.
Author
Technology Assessment; Reusable Launch Vehicles; Technology Utilization; Fabrication; Propellant Tanks

20010067258 NASA Marshall Space Flight Center, Huntsville, AL USA
Fabrication and Testing of Ceramic Matrix Composite Rocket Propulsion Components
Effinger, M. R.; Clinton, R. C., Jr.; Dennis, J.; Elam, S.; Genge, G.; Eckel, A.; Jaskowiak, M. H.; Kiser, J. D.; Lang, J.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; No Copyright; Avail: CASI; A03, Hardcopy

NASA has established goals for Second and Third Generation Reusable Launch Vehicles. Emphasis has been placed on significantly improving safety and decreasing the cost of transporting payloads to orbit. Ceramic matrix composites (CMC) components are being developed by NASA to enable significant increases in safety and engineer performance, while reducing costs. The development of the following CMC components are being pursued by NASA: (1) Simplex CMC Blisk; (2) Cooled CMC Nozzle Ramps; (3) Cooled CMC Thrust Chambers; and (4) CMC Gas Generator. These development efforts are application oriented, but have a strong underpinning of fundamental understanding of processing-microstructure-property relationships relative to structural analyses, nondestructive characterization, and material behavior analysis at the coupon and component and system operation levels. As each effort matures, emphasis will be placed on optimizing and demonstrating material/component durability, ideally using a combined Building Block Approach and Build and Bust Approach.
Derived from text
Ceramic Matrix Composites; Reusable Launch Vehicles; Fabrication; Performance Tests; Engine Parts

20010067250 NASA Marshall Space Flight Center, Huntsville, AL USA
Large Composite Structures Processing Technologies for Reusable Launch Vehicles
Clinton, R. C., Jr.; Vickers, J. H.; McMahon, W. M.; Hulcher, A. B.; Johnston, N. J.; Cano, R. J.; Belvin, H. L.; McIver, K.; Franklin, W.; Sidwell, D., et al.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; No Copyright; Avail: CASI; A03, Hardcopy

Significant efforts have been devoted to establishing the technology foundation to enable the progression to large scale composite structures fabrication. We are not capable today of fabricating many of the composite structures envisioned for the second generation reusable launch vehicle (RLV). Conventional ‘aerospace’ manufacturing and processing methodologies (fiber placement, autoclave, tooling) will require substantial investment and lead time to scale-up. Out-of-autoclave process techniques will require aggressive efforts to mature the selected technologies and to scale up. Focused composite processing technology development and demonstration programs utilizing the building block approach are required to enable envisioned second generation RLV large composite structures applications. Government/industry partnerships have demonstrated success in this area and represent best combination of skills and capabilities to achieve this goal.
Derived from text
Composite Structures; Fabrication; Reusable Launch Vehicles

20010067229 NASA Marshall Space Flight Center, Huntsville, AL USA
Thermographic Analysis of Composite Cobonds on the X-33
Russell, S. S.; Walker, J. L.; Lansing, M. D.; Proceedings of The 4th Conference on Aerospace Materials, Processes, and Environmental Technology; February 2001; In English; No Copyright; Avail: CASI; A02, Hardcopy

During the manufacture of the X-33 liquid hydrogen (LH2) Tank 2, a total of 36 reinforcing caps were inspected thermographically. The cured reinforcing sheets of graphite/epoxy were bonded to the tank using a wet cobond process with vacuum bagging and low temperature curing. A foam filler material wedge separated the reinforcing caps from the outer skin of the tank. Manufacturing difficulties caused by a combination of the size of the reinforcing caps and their complex geometry lead to a potential for trapping air in the bond line. An inspection process was desired to ensure that the bond line was free of voids before it had cured so that measures could be taken to rub out the entrapped air or remove the cap and perform additional surface matching. Infrared thermography was used to perform the procure ‘wet bond’ inspection as well a to

document the final 'cured' condition of the caps. The thermal map of the bond line was acquired by heating the cap with either a flash lamp or a set of high intensity quartz lamps and then viewing it during cool down. The inspections were performed through the vacuum bag and voids were characterized by localized hot spots. In order to ensure that the cap had bonded to the tank properly, a post cure 'flash heating' thermographic investigation was performed with the vacuum bag removed. Any regions that had opened up after the preliminary inspection or that were hidden during the bagging operation were marked and filled by drilling small holes in the cap and injecting resin. This process was repeated until all critical sized voids were filled.
Author
Thermography; Voids; Joints (Junctions); Infrared Inspection; Nondestructive Tests

20010063955 NASA Dryden Flight Research Center, Edwards, CA USA
Two Reconfigurable Flight-Control Design Methods: Robust Servomechanism and Control Allocation
Burken, John J.; Lu, Ping; Wu, Zheng-Lu; Bahm, Cathy; Journal of Guidance, Control and Dynamics; May-June 2001; ISSN 0731-5090; Volume 24, No. 3; In English; AIAA Guidance Navigation and Control Conference, 9-11 Aug. 1999, Portland, OR, USA
Contract(s)/Grant(s): NCC4-117; RTOP 242-33-02-00-23
Report No.(s): H-2433; AIAA Paper 99-4134; Copyright; Avail: CASI; A03, Hardcopy

Two methods for control system reconfiguration have been investigated. The first method is a robust servomechanism control approach (optimal tracking problem) that is a generalization of the classical proportional-plus-integral control to multiple input-multiple output systems. The second method is a control-allocation approach based on a quadratic programming formulation. A globally convergent fixed-point iteration algorithm has been developed to make onboard implementation of this method feasible. These methods have been applied to reconfigurable entry flight control design for the X-33 vehicle. Examples presented demonstrate simultaneous tracking of angle-of-attack and roll angle commands during failures of the fight body flap actuator. Although simulations demonstrate success of the first method in most cases, the control-allocation method appears to provide uniformly better performance in all cases.
Author
Flight Control; Servomechanisms; Proportional Control; Mimo (Control Systems); Quadratic Programming; X-33 Reusable Launch Vehicle

20010060161 Old Dominion Univ., Norfolk, VA USA
N-Factor Computations for the X-33 Vehicle
Balakumar, P.; Hamilton, H. Harris, Technical Monitor; May 24, 2001; In English
Contract(s)/Grant(s): NAG1-2297
Report No.(s): ODURF-103331; No Copyright; Avail: CASI; A05, Hardcopy

N-Factor computations for the X-33 vehicle on the lower side of the body were per-formed. The transition prediction code e-Malik was used to compute the N-Factors and the meanflow field was computed using the CFD code LAURA. The computations are done at two angles of attack 30 and 40 degrees and the freestream Mach number and Reynolds number are M= 6 and Re = 7.7* 10(exp 6). The N-Factors obtained are very low in the range of 2-3 near the experimentally observed transition onset regions. The stability computations show that at 30 degrees angle of attack, the transition occurs due to crossflow instability and at higher angles of attack the transition occurs due to first mode inviscid instability.
Author
Angle of Attack; Computational Fluid Dynamics; Inviscid Flow; Stability; X-33 Reusable Launch Vehicle; Boundary Layers

20010059347 NASA Marshall Space Flight Center, Huntsville, AL USA
Progress and Status of the 2nd Generation Reusable Launch Vehicle Program
Dumbacher, Daniel L.; Rogacki, John R., Technical Monitor; [2001]; In English; 52nd International Astronautical Congress, 1-5 Oct. 2001, Toulouse, France; No Copyright; Avail: Other Sources; Abstract Only

Safe, low-cost space transportation is the key enabler of the commercial development and civil exploration of space. Human space flight remains a hazardous endeavor in spite of advances in aerospace technology. NASA intends to reduce the resources devoted to routine space operations, thus enabling additional science research, technology development, and exploration activities. NASA developed the Integrated Space Transportation Plan (ISTP) to address the needs for space access. ISTP is a balanced and comprehensive plan that embodies strategies for achieving goals for earth-to-orbit and in-space transportation operations. ISTP calls for near-term safety related upgrades to the Space Shuttle, development of a Crew Return Vehicle, a new Space Launch Initiative (SLI), and development of an Aero-Space Base to provide foundation technology for

future space systems. NASA formed the SLI to coordinate the development of the 2nd Generation Reusable Launch Vehicle (RLV) architecture, The SLI is implemented by the 2nd Generation RLV Program Office.
Author
Reusable Launch Vehicles; Spacecraft Launching; Aerospace Systems; Space Transportation; Low Cost; Space Shuttles

20010054791 Georgia Inst. of Tech., Atlanta, GA USA
A Method of Integrating Aeroheating into Conceptual Reusable Launch Vehicle Design: Evaluation of Advanced Thermal Protection Techniques for Future Reusable Launch Vehicles
Olds, John R.; Cowart, Kris; June 2001; In English
Contract(s)/Grant(s): NCC2-5332; No Copyright; Avail: CASI; A09, Hardcopy

A method for integrating Aeroheating analysis into conceptual reusable launch vehicle (RLV) design is presented in this thesis. This process allows for faster turn-around time to converge a RLV design through the advent of designing an optimized thermal protection system (TPS). It consists of the coupling and automation of four computer software packages: MINIVER, TPSX, TCAT, and ADS. MINIVER is an Aeroheating code that produces centerline radiation equilibrium temperatures, convective heating rates, and heat loads over simplified vehicle geometries. These include flat plates and swept cylinders that model wings and leading edges, respectively. TPSX is a NASA Ames material properties database that is available on the World Wide Web. The newly developed Thermal Calculation Analysis Tool (TCAT) uses finite difference methods to carry out a transient in-depth 1-D conduction analysis over the center mold line of the vehicle. This is used along with the Automated Design Synthesis (ADS) code to correctly size the vehicle's thermal protection system (TPS). The numerical optimizer ADS uses algorithms that solve constrained and unconstrained design problems. The resulting outputs for this process are TPS material types, unit thicknesses, and acreage percentages. TCAT was developed for several purposes. First, it provides a means to calculate the transient in-depth conduction seen by the surface of the TPS material that protects a vehicle during ascent and reentry. Along with the in-depth conduction, radiation from the surface of the material is calculated along with the temperatures at the backface and interior parts of the TPS material. Secondly, TCAT contributes added speed and automation to the overall design process. Another motivation in the development of TCAT is optimization. In some vehicles, the TPS accounts for a high percentage of the overall vehicle dry weight. Optimizing the weight of the TPS will thereby lower the percentage of the dry weight accounted for by the TPS. Also, this will lower the cost of the TPS and the overall cost of the vehicle.
Derived from text
Aerodynamic Heating; Computer Programs; Thermal Protection

20010050134 Old Dominion Univ., Norfolk, VA USA
Panel Flutter and Sonic Fatigue Analysis for RLV
Mei, Chuh; Cheng, Guangfeng; May 2001; In English
Contract(s)/Grant(s): NAG1-2150; ODURF Proj. 191881; No Copyright; Avail: CASI; A03, Hardcopy

A methodology is presented for the flutter analysis of the seal of thermal protection system (TPS) panel of X-33 Advanced Technology Demonstrator test vehicle. The seal is simulated as a two-dimensional cantilevered panel with an elastic stopper, which is modeled as an equivalent spring. This cantilever beam-spring model under the aerodynamic pressure at supersonic speeds turns out to be an impact nonlinear dynamic system. The flutter analysis of the seal is thus carried out using, time domain numerical simulation with a displacement stability criterion. The flutter boundary of the seal is further verified with a family of three traditional and one nontraditional panel flutter models. The frequency domain method that applies eigenanalysis on the traditional panel flutter problem was used. The results showed that the critical dynamic pressure could be more than doubled with properly chosen material for the base stopper. The proposed methodology can be easily extended to three-dimensional panel seals with flow angularity.
Author
Acoustic Fatigue; Boundaries; Displacement; Flutter Analysis; Panel Flutter; Supersonic Speed; Thermal Protection

20010048747 NASA Marshall Space Flight Center, Huntsville, AL USA
An Approach to Establishing System Benefits for Technologies In NASA's Spaceliner Investment Area
Hueter, Uwe; Pannell, Bill; Lyles, Garry M., Technical Monitor; [2001]; In English; 52nd International Aeronautical Congress, 1-5 Oct. 2001, Toulouse, France; No Copyright; Avail: Other Sources; Abstract Only

NASA's has established long term goals for access-to-space. The third generation launch systems are to be fully reusable and operational around 2025. The goals for the third generation launch system are to significantly reduce cost and improve

safety over current systems. The Advanced Space Transportation Program Office (ASTP) at the NASA's Marshall Space Flight Center in Huntsville, AL has the agency lead to develop space transportation technologies. Within ASTP, under the Spaceliner Investment Area, third generation technologies are being pursued. The Spaceliner Investment Area's primary objective is to mature vehicle technologies to enable substantial increases in the design and operating margins of third generation RLVs (current Space Shuttle is considered the first generation RLV) by incorporating advanced propulsion systems, materials, structures, thermal protection systems, power, and avionics technologies. Advancements in design tools and better characterization of the operational environment will result in reduced design and operational variabilities leading to improvements in margins. Improvements in operational efficiencies will be obtained through the introduction of integrated vehicle health management, operations and range technologies. Investments in these technologies will enable the reduction in the high operational costs associated with today's vehicles by allowing components to operate well below their design points resulting in improved component operating life, reliability, and safety which in turn reduces both maintenance and refurbishment costs. The introduction of advanced technologies may enable horizontal takeoff by significantly reducing the takeoff weight and allowing use of existing infrastructure. This would be a major step toward the goal of airline-like operation. These factors in conjunction with increased flight rates, resulting from reductions in transportation costs, will result in significant improvements of future vehicles. The real-world problem is that resources are limited and technologies need to be prioritized to assure the resources are spent on technologies that provide the highest system level benefits. Toward that end, a systems approach is being taken to determine the benefits of technologies for the Spaceliner Investment Area. Technologies identified to be enabling will be funded. However, the other technologies will be funded based on their system's benefits. Since the final launch system concept will not be decided for many years, several vehicle concepts are being evaluated to determine technology benefits. Not only performance, but also cost and operability are being assessed. This will become an annual process to assess these technologies against their goals and the benefits to various launch systems concepts. The paper describes the system process, tools and concepts used to determine the technology benefits. Preliminary results will be presented along with the current technology investments that are being made by ASTP's Spaceliner Investment Area.
Author
Cost Reduction; Operating Costs; Space Transportation System; Reusable Launch Vehicles; Reusable Spacecraft; Cost Effectiveness; Cost Analysis

20010046963 NASA Marshall Space Flight Center, Huntsville, AL USA
An Overview of In-Stu Treatability Studies at Marshall Space Flight Center, Huntsville, Alabama
McElroy, Bill; Keith, Amy; Glasgow, J. K.; Dasappa, Srini; McCaleb, Rebecca, Technical Monitor; [2001]; In English; Oxidation and Reduction Technologies for In-Situ Treatment of Soil and Groundwater, 26-29 Jun. 2001, Niagra Falls, Canada; No Copyright; Avail: Other Sources; Abstract Only

Marshall Space Flight Center (MSFC) is located in Huntsville, Alabama (north-central Alabama), on approximately 1,840 acres near the center of the U.S. Army's Redstone Arsenal (RSA). MSFC is the National Aeronautics and Space Administration's (NASA's) principal propulsion development center. Its scientists, engineers, and support personnel play a major role in the National Space Transportation System by managing space shuttle mission activities, including the microgravity laboratory. In addition, MSFC will be a significant contributor to several of NASA's future programs, including the Reusable Launch Vehicle (X-33), International Space Station, and Advanced X-ray Astrophysics Facility, as well as research on a variety of space science applications. MSFC has been used to develop, test and manufacture space vehicles and components since 1960, when civilian rocketry and missile activities were transferred from RSA to MSFC. In 1994, MSFC was placed on the National Priority List for the management of hazardous waste sites, under the requirements of the Comprehensive Environmental Response, Compensation, and Liability Act (CERCLA). One requirement of the CERCLA program is to evaluate the nature and extent of environmental contamination resulting from identified CERCLA sites, assess the public health and environmental risks associated with the identified contamination, and identify potential remedial actions. A CERCLA remedial investigation (RI) for the groundwater system has identified at least five major plumes of chlorinated volatile organic compounds (CVOCs) in the groundwater beneath the facility. These plumes are believed to be the result of former management practices at 14 main facility locations (termed 'source areas') where CVOCs were released to the subsurface. Trichloroethene (TCE) is the predominant CVOC and is common to all the plumes. Perchloroethene (PCE) also exists in two of the plumes. In addition to TCE and PCE, carbon tetrachloride and 1,1,2,2-tetrachloroethane are contained in one of the plumes. The CVOCs are believed to exist as dense non-aqueous phase liquids (DNAPLs) beneath many of the source areas.
Author
NASA Programs; Environment Pollution

20010046959 NASA Marshall Space Flight Center, Huntsville, AL USA

X-33 XRS-2200 Linear Aerospike Engine Sea Level Plume Radiation

DAgostino, Mark; Lee, Young C.; Wang, Ten-See; Turner, Jim E., Technical Monitor; [2001]; In English; Fluids Workshop, 5 Apr. 2001, Huntsville, AL, USA; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents radiation data for the linear aerospike engine.

CASI

Aerospike Engines; Plumes

20010046955 NASA Marshall Space Flight Center, Huntsville, AL USA

2nd Generation Reusable Launch Vehicle Potential Commercial Development Scenarios

Creech, Stephen D.; Rogacki, John R., Technical Monitor; [2001]; In English; IAA 31st Symposium of Economics and Commercialization of Space Activities; No Copyright; Avail: Other Sources; Abstract Only

The presentation will discuss potential commercial development scenarios for a Second Generation Reusable Launch Vehicle. The analysis of potential scenarios will include commercial rates of return, government return on investment, and market considerations. The presentation will include policy considerations in addition to analysis of Second Generation Reusable Launch Vehicle economics. The data discussed is being developed as a part of NASA's Second Generation Reusable Launch Vehicle Program, for consideration as potential scenarios for enabling a next generation system. Material will include potential scenarios not previously considered by NASA or presented at other conferences. Candidate paper has not been presented at a previous meeting, and conference attendance of the author has been approved by NASA.

Author

Reusable Launch Vehicles; Economics; Commercialization; Conferences

20010045816 NASA Marshall Space Flight Center, Huntsville, AL USA

2nd Generation RLV: Program Goals and Acquisition Strategy

Graham, J. Bart; Dumbacher, D. L., Technical Monitor; [2001]; In English; Space Technology and Applications International Forum, 11-14 Feb. 2001, Albuquerque, NM, USA; No Copyright; Avail: CASI; A02, Hardcopy

The risk to loss of life for Space Shuttle crewmembers is approximately one in 245 missions. U.S. launch service providers captured nearly 100%, of the commercial launch market revenues in the mid 1980s. Today, the U.S. captures less than 50% of that market. A launch system architecture is needed that will dramatically increase the safety of space flight while significantly reducing the cost. NASA's Space Launch Initiative, which is implemented by the 2nd Generation RLV Program Office at Marshall Space Flight Center, seeks to develop technology and reusable launch vehicle concepts which satisfy the commercial launch market needs and the unique needs of NASA. Presented in this paper are the five primary elements of NASA's Integrated Space Transportation Plan along with the highest level goals and the acquisition strategy of the 2nd Generation RLV Program. Approval of the Space Launch Initiative FY01 budget of $290M is seen as a major commitment by the Agency and the Nation to realize the commercial potential that space offers and to move forward in the exploration of space.

Author

Space Transportation; Reusable Launch Vehicles; Market Research

20010043670 NASA Langley Research Center, Hampton, VA USA

Detection of Micro-Leaks Through Complex Geometries Under Mechanical Load and at Cryogenic Temperature

Rivers, H. Kevin; Sikora, J. G.; Sankaran, S. N.; [2001]; In English; 42nd Structures, Structural Dynamics, and Materials Conference and Exhibit, 16-19 Apr. 2001, Seattle, WA, USA

Report No.(s): AIAA Paper 2001-1218; Copyright; Avail: CASI; A01, Hardcopy

Polymer Matrix Composite (PMC) hydrogen tanks have been proposed as an enabling technology for reducing the weight of Single-Stage-to-Orbit reusable launch vehicles where structural mass has a large impact on vehicle performance. A key development issue of these lightweight structures is the leakage of hydrogen through the composite material. The rate of hydrogen leakage can be a function of the material used, method of 6 fabrication used to manufacture the tank, mechanical load the tank must react, internal damage-state of the material, and the temperatures at which the tank must operate. A method for measuring leakage through a geometrically complex structure at cryogenic temperature and under mechanical load was developed, calibrated and used to measure hydrogen leakage through complex X-33 liquid-hydrogen tank structure sections.

Author

Polymer Matrix Composites; Leakage; Propellant Tanks; Cryogenic Temperature; Liquid Hydrogen; Fault Detection; Composite Structures

20010041325 NASA Marshall Space Flight Center, Huntsville, AL USA

Development of Aluminum Composites for a Rocket Engine's Lightweight Thrust Cell

Lee, Jonathan A.; Elam, Sandy; Munafo, Paul M., Technical Monitor; [2001]; In English; 25th Annual Conference on Composites, Materials and Structures, 21-26 Jan. 2001, Cocoa Beach, FL, USA

Contract(s)/Grant(s): RTOP 713-23-53; No Copyright; Avail: Other Sources; Abstract Only

The Aerospike liquid fueled rocket engine for the X-33 aerospace vehicle consists of several thrust cells, which can comprise as much as 25% of the engine weight. The interior wall of the thrust cell chamber is exposed to high temperature combustion products and must be cooled by using liquid hydrogen. Ultimately, reducing engine weight would increase vehicle performance and allow heavier payload capabilities. Currently, the thrust cell's structural jacket and manifolds are made of stainless steel 347, which can potentially be replaced by a lighter material such as an Aluminum (Al) Metal Matrix Composites (MMC). Up to 50% weight reduction can be expected for each of the thrust cell chambers using particulate SiC reinforced Al MMC. Currently, several Al MMC thrust cell structural jackets have been produced, using cost-effective processes such as gravity casting and plasma spray deposition, to demonstrate MMC technology readiness for NASA's advanced propulsion systems.

Author

Aerospike Engines; Aluminum; Metal Matrix Composites; Thrust Chambers; Weight Reduction

20010041080 NASA Ames Research Center, Moffett Field, CA USA

A Two-Stage-to-Orbit Spaceplane Concept With Growth Potential

Mehta, Unmeel B.; Bowles, Jeffrey V.; February 2001; In English; 10 International Space Planes and Hypersonic Systems and Technologies Conference, 24-27 Apr. 2001, Kyoto, Japan; Original contains color illustrations

Contract(s)/Grant(s): RTOP 713-74-00

Report No.(s): NASA/TM-2001-209620; NAS 1.15:209620; A-00V0029; No Copyright; Avail: CASI; A03, Hardcopy

A two-stage-to-orbit (TSTO) spaceplane concept developed in 1993 is revisited, and new information is provided to assist in the development of the next-generation space transportation vehicles. The design philosophy, TSTO spaceplane concept, and the design method are briefly described. A trade study between cold and hot structures leads to the choice of cold structures with external thermal protection systems. The optimal Mach number for staging the second stage of the TSTO spaceplane (with air-breathing propulsion on the first stage) is 10, based on life-cycle cost analysis. The performance and specification of a prototype/experimental (P/X) TSTO spaceplane with a turbo/ram/scramjet propulsion system and built-in growth potential are presented and discussed. The internal rate of return on investment is the highest for the proposed TSTO spaceplane, vis-A-vis a single-stage-to-orbit (SSTO) rocket vehicle and a TSTO spaceplane without built-in growth. Additional growth potentials for the proposed spaceplane are suggested. This spaceplane can substantially decrease access-to-space cost and risk, and increase safety and reliability in the near term It can be a serious candidate for the next-generation space transportation system.

Author

Single Stage to Orbit Vehicles; Space Transportation System; Air Breathing Engines; Aerospace Planes; Design Analysis

20010039539 NASA Marshall Space Flight Center, Huntsville, AL USA

Permeability Testing of Impacted Composite Laminates for Use on Reusable Launch Vehicles

Nettles, Alan T.; Munafo, Paul, Technical Monitor; February 2001; In English

Report No.(s): NASA/TM-2001-210799; M-1001; NAS 1.15:210799; No Copyright; Avail: Other Sources; Abstract Only

Since composite laminates are beginning to be identified for use in reusable launch vehicle propulsion systems, an understanding of their permeance is needed. A foreign object impact event can cause a localized area of permeability (leakage) in a polymer matrix composite, and it is the aim of this study to assess a method of quantifying permeability-after-impact results. A simple test apparatus is presented, and variables that could affect the measured values of permeability-after-impact were assessed. Once it was determined that valid numbers were being measured, a fiber/resin system was impacted at various impact levels and the resulting permeability measured, first with a leak check solution (qualitative) then using the new apparatus (quantitative). The results showed that as the impact level increased, so did the measured leakage. As the pressure to the specimen was increased, the leak rate was seen to increase in a nonlinear fashion for almost all the specimens tested.

Author

Laminates; Permeability; Polymer Matrix Composites; Impact Tests

20010039031 Boeing Co., Huntington Beach, CA USA, NASA Marshall Space Flight Center, Huntsville, AL USA
A Cryogenic Propellant Production Depot for Low Earth Orbit
Potter, Seth D.; Henley, Mark; Guitierrez, Sonia; Fikes, John; Carrington, Connie; Smitherman, David; Gerry, Mark; Sutherlin, Steve; Beason, Phil; Howell, Joe, Technical Monitor; [2001]; In English; International Space Development Conference, 24-28 May 2001, Albuquerque, NM, USA; No Copyright; Avail: CASI; A03, Hardcopy

The cost of access to space beyond low Earth orbit can be lowered if vehicles can refuel in orbit. The power requirements for a propellant depot that electrolyzes water and stores cryogenic oxygen and hydrogen can be met using technology developed for space solar power. A propellant depot is described that will be deployed in a 400 km circular equatorial orbit, receive tanks of water launched into a lower orbit from Earth by gun launch or reusable launch vehicle, convert the water to liquid hydrogen and oxygen, and store Lip to 500 metric tonnes of cryogenic propellants. The propellant stored in the depot can support transportation from low Earth orbit to geostationary Earth orbit, the Moon, LaGrange points, Mars, etc. The tanks are configured in an inline gravity-gradient configuration to minimize drag and settle the propellant. Temperatures can be maintained by body-mounted radiators; these will also provide some shielding against orbital debris. Power is supplied by a pair of solar arrays mounted perpendicular to the orbital plane, which rotate once per orbit to track the Sun. In the longer term, cryogenic propellant production technology can be applied to a larger LEO depot, as well as to the use of lunar water resources at a similar depot elsewhere.
Author
Cryogenic Rocket Propellants; Low Earth Orbits; Refueling

20010038424 NASA Marshall Space Flight Center, Huntsville, AL USA
Unshrouded Impeller Technology Task Status
Williams, Robert W.; Turner, James E., Technical Monitor; [2001]; In English; TD Fluids Workshop, 4-5 Apr. 2001, Huntsville, AL, USA; Original contains color illustrations; No Copyright; Avail: CASI; A03, Hardcopy

The objective of this presentation is to develop an unshrouded impeller design that will meet the performance requirements of a 3-stage fuel pump with a 2-stage pump design.
CASI
Shrouded Propellers; Impellers; Fuel Pumps; Turbine Pumps

20010038264 NASA Marshall Space Flight Center, Huntsville, AL USA
NASA's Spaceliner Investment Area Technology Activities
Hueter, Uwe; Lyles, Garry M., Technical Monitor; [2001]; In English; 10th International Space Planes and Hypersonics Systems and Technologies Conference, 24-27 Apr. 2001, Kyoto, Japan; Copyright; Avail: CASI; A02, Hardcopy

NASA's has established long term goals for access-to-space. The third generation launch systems are to be fully reusable and operational around 2025. The goals for the third generation launch system are to significantly reduce cost and improve safety over current conditions. The Advanced Space Transportation Program Office (ASTP) at the NASA's Marshall Space Flight Center in Huntsville, AL has the agency lead to develop space transportation technologies. Within ASTP, under the Spaceliner Investment Area, third generation technologies are being pursued in the areas of propulsion, airframes, integrated vehicle health management (IVHM), avionics, power, operations, and range. The ASTP program will mature these technologies through both ground and flight system testing. The Spaceliner Investment Area plans to mature vehicle technologies to reduce the implementation risks for future commercially developed reusable launch vehicles (RLV). The plan is to substantially increase the design and operating margins of the third generation RLV (the Space Shuttle is the first generation) by incorporating advanced technologies in propulsion, materials, structures, thermal protection systems, avionics, and power. Advancements in design tools and better characterization of the operational environment will allow improvements in design margins. Improvements in operational efficiencies will be provided through use of advanced integrated health management, operations, and range technologies. The increase in margins will allow components to operate well below their design points resulting in improved component operating life, reliability, and safety which in turn reduces both maintenance and refurbishment costs. These technologies have the potential of enabling horizontal takeoff by reducing the takeoff weight and achieving the goal of airline-like operation. These factors in conjunction with increased flight rates from an expanding market will result in significant improvements in safety and reductions in operational costs of future vehicles. The paper describes current status, future plans and technologies that are being matured by the Spaceliner Investment Area under the Advanced Space Transportation Program Office.
Author
Space Transportation; Technology Utilization; NASA Programs; Avionics; Systems Analysis; Spacecraft Propulsion

20010037825 Engineering Sciences, Inc., Huntsville, AL USA

Unic Unstructured CFD Methodology Development

Chen, Y. S.; Liu, J.; Zhang, S.; Mallapragada, P.; Turner, James E., Technical Monitor; Wang, T. S., Technical Monitor; D'Agostino, M., Technical Monitor; [2001]; In English; Fluids, 4-5 Apr. 2001, Huntsville, AL, USA; Original contains color illustrations

Contract(s)/Grant(s): NAS8-00002; No Copyright; Avail: CASI; A03, Hardcopy

Base heating characteristics is crucial to the success and the overall performance of the X-33 engine. Base heating is important throughout the entire flight trajectory due to the aerospike engine design of X-33. The base region is surrounded by the hot-gas plume, which expands, circulates and impinges on the base. An advanced computation fluid dynamics method is employed in an effort to develop a robust, accurate and efficient tool for the X-33 base heating performance predictions. This computational tool is developed based on a Navier-Stokes flow solver, which is suitable for general complex geometry and includes turbulence, finite-rate chemistry, and radiation models. To fulfill the fast turnaround requirement as a design analysis tool, adaptive mesh refinement method and parallel-computing algorithm are also incorporated in the present model. Case study for the X-33 base-region fluid dynamics and heat transfer characteristics are presented.

Author

Computational Fluid Dynamics; X-33 Reusable Launch Vehicle; Base Heating; Engine Design; Heat Transfer; Design Analysis

20010028815 NASA Langley Research Center, Hampton, VA USA

Evaluation of Computational Aeroelasticity Codes for Loads and Flutter

Huttsell, Larry; Schuster, Dave; Volk, John; Giesing, Joe; Love, Mike; [2001]; In English

Report No.(s): AIAA Paper 01-0569; Copyright; Avail: CASI; A03, Hardcopy

This paper presents an overview of a joint effort to evaluate computational aeroelasticity codes for loads and flutter. Computations are performed on realistic problems for static aeroelasticity, classic flutter, Limit Cycle Oscillations (LCO), and control surface buzz. The codes being evaluated involve a transonic small disturbance code with an interactive boundary layer method and two Euler/Navier-Stokes codes. To evaluate the accuracy of these codes, comparisons are made with available wind tunnel or flight test data. This paper presents the results/status of the following applications: static aeroelasticity - aeroelastic tailored wing, flutter - AV-8B-like wing and F-I5-like tail, LCO - B-1-like and B-2, and control surface buzz - NASP wind tunnel model and Global Hawk wing.

Author

Aeroelasticity; Flutter; Loads (Forces); Control Surfaces; Wing Oscillations

20010025824 Pratt and Whitney Aircraft, West Palm Beach, FL USA

2nd Generation Reusable Launch Vehicle (2G RLV)

Matlock, Steve; Sides, Steve; Kmiec, Tom; Arbogast, Tim; Mayers, Tom; Doehnert, Bill; Feb. 23, 2001; In English; Original contains color illustrations

Contract(s)/Grant(s): NAS8-00168; NRA8-27

Report No.(s): LS-50-Rev; No Copyright; Avail: CASI; A06, Hardcopy

This is a revised final report and addresses all of the work performed on this program. Specifically, it covers vehicle architecture background, definition of six baseline engine cycles, reliability baseline (space shuttle main engine QRAS), and component level reliability/performance/cost for the six baseline cycles, and selection of 3 cycles for further study. This report further addresses technology improvement selection and component level reliability/performance/cost for the three cycles selected for further study, as well as risk reduction plans, and recommendation for future studies.

Author

Reusable Launch Vehicles; Rocket Engine Design; Liquid Rocket Propellants; Space Shuttle Main Engine; Performance Prediction

20010021342 NASA Marshall Space Flight Center, Huntsville, AL USA

Lightweight Chambers for Thrust Assemblies

Elam, Sandra K.; Lee, Jonathan; Holmes, Richard; Zimmerman, Frank; Effinger, Mike; Turner, James E., Technical Monitor; [2001]; In English; 52nd International Astronautical Congress, 1-5 Oct. 2001, Toulouse, France

Contract(s)/Grant(s): NRA8-21; No Copyright; Avail: Other Sources; Abstract Only

The Marshall Space Flight Center (MSFC) of the National Aeronautics and Space Administration (NASA) has

successfully applied new materials and fabrication techniques to create actively cooled thrust chambers that operate 200-400 degrees hotter and weigh 50% lighter than conventional designs. In some vehicles, thrust assemblies account for as much as 20% of the engine weight. So, reducing the weight of these components and increasing their operating range will benefit many engines and vehicle designs, including Reusable Launch Vehicle (RLV) concepts. Obviously, copper and steel alloys have been used successfully for many years in the chamber components of thrust assemblies. Yet, by replacing the steel alloys with Polymer Matrix Composite (PMC) and/or Metal Matrix Composite (MMC) materials, design weights can be drastically reduced. In addition, replacing the traditional copper alloys with a Ceramic Matrix Composite (CMC) or an advanced copper alloy (Cu-8Cr-4Nb, also known as GRCop-84) significantly increases allowable operating temperatures. Several small MMC and PMC demonstration chambers have recently been fabricated with promising results. Each of these designs included GRCop-84 for the cooled chamber liner. These units successfully verified that designs over 50% lighter are feasible. New fabrication processes, including advanced casting technology and a low cost vacuum plasma spray (VPS) process, were also demonstrated with these units. Hot-fire testing at MSFC is currently being conducted on the chambers to verify increased operating temperatures available with the GRCop-84 liner. Unique CMC chamber liners were also successfully fabricated and prepared for hot-fire testing. Yet, early results indicate these CMC liners need significantly more development in order to use them in required chamber designs. Based on the successful efforts with the MMC and PMC concepts, two full size 'lightweight' chambers are currently being designed and fabricated for hot-fire testing at MSFC in 2001. These 'full size' chambers will be similar in size to those used on the X33 engine (RS2200). One will be fabricated with a MMC structural jacket, while the other uses a PMC jacket. Each will be designed for thrust levels of 15,000 pounds in an oxygen/hydrogen environment with liquid hydrogen coolant. Both chambers will use GRCop-84 for its channel wall liner. Each unit is expected to be at least 60% lighter than a conventional design with traditional materials. Hot-fire testing on the full size units in late 2001 will directly compare performance results between a conventional chamber design and these 'lightweight' alternatives. The technology developed and demonstrated by this effort will not only benefit next generation RLV programs, but it can be applied to other existing and future engine programs, as well. Efforts were sponsored by the Advanced Space Transportation Program for RLV Focused Technologies. The task team was led by MSFC with additional members from NASA-Glenn Research Center and the Rocketdyne Division of The Boeing Company. Specific materials development and fabrication processes were provided by Aerojet, Lockheed Martin Astronautics, Composite Optics, Inc., Hyper-Therm, Ceramic Composites, Inc., MSE Technology Applications, and Plasma Processes, Inc.
Author
Technology Assessment; Fabrication; Ceramic Matrix Composites; Copper Alloys; Metal Matrix Composites; Polymer Matrix Composites; Technology Utilization; Thrust Chambers

20010020948 NASA Marshall Space Flight Center, Huntsville, AL USA
Permeability Testing of Impacted Composite Laminates for Use on Reusable Launch Vehicles
Nettles, A. T.; February 2001; In English
Report No.(s): NASA/TM-2001-210799; M-1001; NAS 1.15:210799; No Copyright; Avail: CASI; A03, Hardcopy

Since composite laminates are beginning to be identified for use in reusable launch vehicle propulsion systems, an understanding of their permeance is needed. A foreign object impact event can cause a localized area of permeability (leakage) in a polymer matrix composite, and it is the aim of this study to assess a method of quantifying permeability-after-impact results. A simple test apparatus is presented, and variables that could affect the measured values of permeability-after-impact were assessed. Once it was determined that valid numbers were being measured, a fiber/resin system was impacted at various impact levels and the resulting permeability measured, first with a leak check solution (qualitative) then using the new apparatus (quantitative). The results showed that as the impact level increased, so did the measured leakage. As the pressure to the specimen was increased, the leak rate was seen to increase in a nonlinear fashion for almost all the specimens tested.
Author
Laminates; Leakage; Permeability; Microcracks; Impact Tests

20010020386 NASA Marshall Space Flight Center, Huntsville, AL USA
RLV Turbine Performance Optimization
Griffin, Lisa W.; Dorney, Daniel J.; [2000]; In English; 12th PERC Propulsion Symposium, 26-27 Oct. 2000, Cleveland, OH, USA; Original contains color illustrations
Contract(s)/Grant(s): NRA8-21; No Copyright; Avail: CASI; A02, Hardcopy

A task was developed at NASA/Marshall Space Flight Center (MSFC) to improve turbine aerodynamic performance through the application of advanced design and analysis tools. There are four major objectives of this task: 1) to develop, enhance, and integrate advanced turbine aerodynamic design and analysis tools; 2) to develop the methodology for application

of the analytical techniques; 3) to demonstrate the benefits of the advanced turbine design procedure through its application to a relevant turbine design point; and 4) to verify the optimized design and analysis with testing. Final results of the preliminary design and the results of the two-dimensional (2D) detailed design of the first-stage vane of a supersonic turbine suitable for a reusable launch vehicle (R-LV) are presented. Analytical techniques for obtaining the results are also discussed.
Author
Reusable Launch Vehicles; Computational Fluid Dynamics; Gas Turbine Engines; Aerodynamic Characteristics; Design Optimization

20010019998 NASA Marshall Space Flight Center, Huntsville, AL USA
Testing the Preliminary X-33 Navigation System
Lomas, James J.; Mitchell, Daniel W.; Freestone, Todd M.; Lee, Charles; Lessman, Craig; Foster, Lee D., Technical Monitor; [2001]; In English; 24th Guidance and Control, 31 Jan. - 4 Feb. 2001, Breckenridge, CO, USA; No Copyright; Avail: Other Sources; Abstract Only

The X-33 Reusable Launch Vehicle (RLV) must meet the demanding requirements of landing autonomously on a narrow landing strip following a flight that reaches an altitude of up to 200,000 feet and a speed in excess of Mach 9 with significant in-flight energy bleed-off maneuvers. To execute this flight regimen a highly reliable avionics system has been designed that includes three LN-100G Inertial Navigation System/Global Positioning System (INS/GPS) units as the primary navigation system for the X-33. NASA's Marshall Space Flight Center (MSFC) tested an INS/GPS system in real-time simulations to determine the ability of this navigation suite to meet the in flight and autonomous landing requirements of the X-33 RLV. A total of sixty-one open loop tests were performed to characterize the navigation accuracy of the LN-100G. Twenty-seven closed-loop tests were also performed to evaluate the performance of the X-33 Guidance, Navigation and Control (GN&C) algorithms with the real navigation hardware. These closed-loop tests were also designed to expose any integration or operational issues with the real-time X-33 vehicle simulation. Dynamic road tests of the INS/GPS were conducted by Litton to assess the performance of differential and nondifferential INS/GPS hybrid navigation solutions. The results of the simulations and road testing demonstrate that this novel solution is capable of meeting the demanding requirements of take-off, in-flight navigation, and autonomous landing of the X-33 RLV. This paper describes the test environment developed to stimulate the LN-100G and discusses the results of this test effort. This paper also presents recommendations for a navigation system suitable to an operational RLV system.
Author
X-33 Reusable Launch Vehicle; Autonomy; Spacecraft Landing; Navigation Aids; Performance Tests

20010018568 NASA Langley Research Center, Hampton, VA USA
Infrared Sensing Aeroheating Flight Experiement: STS-96 Flight Results
Blanchard, Robert C.; Wilmoth, Richard G.; Glass, Christopher E.; Merski, N. Ronald, Jr.; Berry, Scott A.; Bozung, Timothy J.; Tietjen, Alan; Wendt, Jodean; Dawson, Don; [2001]; In English; 39th Aerospace Sciences, 8-11 Jan. 2001, Reno, NV, USA; Original contains color illustrations
Report No.(s): AIAA Paper 2001-0352; Copyright; Avail: CASI; A03, Hardcopy

Major elements of an experiment called the Infrared Sensing Aeroheating Flight Experiment are discussed. The primary experiment goal is to provide reentry global temperature images from infrared measurements to define the characteristics of hypersonic boundary-layer transition during flight. Specifically, the experiment is to identify, monitor, and quantify hypersonic boundary layer windward surface transition of the X-33 vehicle during flight. In addition, the flight data will serve as a calibration and validation of current boundary layer transition prediction techniques, provide benchmark laminar, transitional, and fully turbulent global aeroheating data in order to validate existing wind tunnel and computational results, and to advance aeroheating technology. Shuttle Orbiter data from STS-96 used to validate the data acquisition and data reduction to global temperatures, in order to mitigate the experiment risks prior to the maiden flight of the X-33, is discussed. STS-96 reentry mid-wave (3-5 Pm) infrared data were collected at the Ballistic Missile Defense Organization/Innovative Sciences and Technology Experimentation Facility site at NASA-Kennedy Space Center and subsequently mapped into global temperature contours using ground calibrations only. A series of image mapping techniques have been developed in order to compare each frame of infrared data with thermocouple data collected during the flight. Comparisons of the ground calibrated global temperature images with the corresponding thermocouple data are discussed. The differences are shown to be generally less than about 5%, which is comparable to the expected accuracy of both types of aeroheating measurements.
Author
Infrared Radiation; Performance Tests; Heating; Reentry Physics; Temperature Measurement; Surface Temperature; Data Acquisition; Flight Characteristics

20010017162 NASA Glenn Research Center, Cleveland, OH USA

Preliminary Sizing of Vertical Take-Off Rocket-Based Combined-Cycle Powered Launch Vehicles

Roche, Joseph M.; McCurdy, David R.; January 2001; In English; 37th Combustion, 13-17 Nov. 2000, Monterey, CA, USA
Contract(s)/Grant(s): RTOP 708-73-20
Report No.(s): NASA/TM-2001-210668; E-12503-1; NAS 1.15:210668; No Copyright; Avail: CASI; A03, Hardcopy

The task of single-stage-to-orbit has been an elusive goal due to propulsion performance, materials limitations, and complex system integration. Glenn Research Center has begun to assemble a suite of relationships that tie Rocket-Based Combined-Cycle (RBCC) performance and advanced material data into a database for the purpose of preliminary sizing of RBCC-powered launch vehicles. To accomplish this, a near optimum aerodynamic and structural shape was established as a baseline. The program synthesizes a vehicle to meet the mission requirements, tabulates the results, and plots the derived shape. A discussion of the program architecture and an example application is discussed herein.
Author

Single Stage to Orbit Vehicles; Mission Planning; Aerodynamics

20010016865 NASA Glenn Research Center, Cleveland, OH USA

Structural Sizing of a 25,000-lb Payload, Air-Breathing Launch Vehicle for Single-Stage-to-Orbit

Roche, Joseph M.; Kosareo, Daniel N.; January 2001; In English; 37th Combustion, 13-17 Nov. 2000, Monterey, CA, USA
Contract(s)/Grant(s): RTOP 708-73-20
Report No.(s): NASA/TM-2001-210667; E-12501-1; NAS 1.15:210667; No Copyright; Avail: CASI; A03, Hardcopy

In support of NASA's Air-Breathing Launch Vehicle (ABLV) study, a 25,000-lb payload version of the GTX (formerly Trailblazer) reference vehicle concept was developed. The GTX is a vertical lift-off, reusable, single-stage-to-orbit launch vehicle concept that uses hypersonic air-breathing propulsion in a rocket-based combined-cycle (RBCC) propulsion system to reduce the required propellant fraction. To achieve this goal the vehicle and propulsion system must be well integrated both aerodynamically and structurally to reduce weight. This study demonstrates the volumetric and structural efficiency of a vertical takeoff, horizontal landing, hypersonic vehicle with a circular cross section. A departure from the lifting body concepts, this design philosophy is even extended to the engines, which have semicircular nacelles symmetrically mounted on the vehicle. Material candidates with a potential for lightweight and simplicity have been selected from a set of near term technologies (five to ten years). To achieve the mission trajectory, preliminary weight estimates show the vehicle's gross lift-off weight is 1.26 x 10(exp 6) lb. The structural configuration of the GTX vehicle and its propulsion system are described. The vehicle design benefits are presented, and key technical issues are highlighted.
Author

Air Breathing Boosters; Reusable Launch Vehicles; Propulsion System Configurations; Space Transportation

20010014167 NASA Dryden Flight Research Center, Edwards, CA USA

Wind-Tunnel Investigations of Blunt-Body Drag Reduction Using Forebody Surface Roughness

Whitmore, Stephen A.; Sprague, Stephanie; Naughton, Jonathan W.; Curry, Robert E., Technical Monitor; January 2001; In English; 39th Aerospace Sciences, 8-11 Jan. 2001, Reno, NV, USA; Original contains color illustrations
Contract(s)/Grant(s): RTOP 718-20-00-E8-53; RTOP 529-35-24-M1-00-RR
Report No.(s): NASA/TM-2001-210390; H-2439; NAS 1.15:210390; AIAA Paper 2001-0252; Copyright; Avail: CASI; A03, Hardcopy

This paper presents results of wind-tunnel tests that demonstrate a novel drag reduction technique for blunt-based vehicles. For these tests, the forebody roughness of a blunt-based model was modified using micomachined surface overlays. As forebody roughness increases, boundary layer at the model aft thickens and reduces the shearing effect of external flow on the separated flow behind the base region, resulting in reduced base drag. For vehicle configurations with large base drag, existing data predict that a small increment in forebody friction drag will result in a relatively large decrease in base drag. If the added increment in forebody skin drag is optimized with respect to base drag, reducing the total drag of the configuration is possible. The wind-tunnel tests results conclusively demonstrate the existence of a forebody dragbase drag optimal point. The data demonstrate that the base drag coefficient corresponding to the drag minimum lies between 0.225 and 0.275, referenced to the base area. Most importantly, the data show a drag reduction of approximately 15% when the drag optimum is reached. When this drag reduction is scaled to the X-33 base area, drag savings approaching 45,000 N (10,000 lbf) can be realized.
Author

Wind Tunnel Tests; Aerodynamic Drag; Base Flow; Drag Reduction

20010003921 NASA Langley Research Center, Hampton, VA USA

Infrared Sensing Aeroheating Flight Experiment: STS-96 Flight Results

Blanchard, Robert C.; Wilmoth, Richard G.; Glass, Christopher E.; Merski, N. Ronald, Jr.; Berry, Scott A.; Bozung, Timothy J.; Tietjen, Alan; Wendt, Jodean; Dawson, Don; [2001]; In English; 39th Aerospace Sciences, 8-11 Jan. 2001, Reno, NV, USA; Original contains color illustrations

Report No.(s): AIAA Paper 2001-0352; Copyright; Avail: CASI; A03, Hardcopy

Major elements of an experiment called the Infrared Sensing Aeroheating Flight Experiment are discussed. The primary experiment goal is to provide reentry global temperature images from infrared measurements to define the characteristics of hypersonic boundary-layer transition during flight. Specifically, the experiment is to identify, monitor, and quantity hypersonic boundary layer windward surface transition of the X-33 vehicle during flight. In addition, the flight data will serve as a calibration and validation of current boundary layer transition prediction techniques, provide benchmark laminar, transitional, and fully turbulent global aeroheating data in order to validate existing wind tunnel and computational results, and to advance aeroheating technology. Shuttle Orbiter data from STS-96 used to validate the data acquisition and data reduction to global temperatures, in order to mitigate the experiment risks prior to the maiden flight of the X-33, is discussed. STS-96 reentry midwave (3-5 micron) infrared data were collected at the Ballistic Missile Defense Organization/Innovative Sciences and Technology Experimentation Facility site at NASA-Kennedy Space Center and subsequently mapped into global temperature contours using ground calibrations only. A series of image mapping techniques have been developed in order to compare each frame of infrared data with thermocouple data collected during the flight. Comparisons of the ground calibrated global temperature images with the corresponding thermocouple data are discussed. The differences are shown to be generally less than about 5%, which is comparable to the expected accuracy of both types of aeroheating measurements.

Author

Infrared Detectors; Aerodynamic Heating; Boundary Layer Transition; Turbulent Boundary Layer; Hypersonic Boundary Layer; X-33 Reusable Launch Vehicle; Space Shuttle Orbiters

20010000337 Lee and Associates, LLC, USA

Support to X-33/Resusable Launch Vehicle Technology Program

Dec. 05, 2000; In English

Contract(s)/Grant(s): NASA Order H-29854-D; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 Guidance, Navigation, and Control (GN&C) Peer Review Team (PRT) was formed to assess the integrated X-33 vehicle GN&C system in order to identify any areas of disproportionate risk for initial flight. The eventual scope of the PRT assessment encompasses the GN&C algorithms, software, avionics, control effectors, applicable models, and testing. The initial (phase 1) focus of the PRT was on the GN&C algorithms and the Flight Control Actuation Subsystem (FCAS). The PRT held meetings during its phase 1 assessment at X-33 assembly facilities in Palmdale, California on May 17-18, 2000 and at Honeywell facilities in Tempe, Arizona on June 7, 2000. The purpose of these meetings was for the PRT members to get background briefings on the X-33 vehicle and for the PRT team to be briefed on the design basis and current status of the X-33 GN&C algorithms as well as the FCAS. The following material is covered in this PRT phase 1 final report. Some significant GN&C-related accomplishments by the X-33 development team are noted. Some topics are identified that were found during phase 1 to require fuller consideration when the PRT reconvenes in the future. Some new recommendations by the PRT to the X-33 program will likely result from a thorough assessment of these subjects. An initial list of recommendations from the PRT to the X-33 program is provided. These recommendations stem from topics that received adequate review by the PRT in phase 1. Significant technical observations by the PRT members as a result of the phase 1 meetings are detailed. (These are covered in an appendix.) There were many X-33 development team members who contributed to the technical information used by the PRT during the phase 1 assessment, who supported presentations to the PRT, and who helped to address the many questions posed by the PRT members at and after the phase 1 meetings. In all instances the interaction between the PRT and the X-33 development team members was cordial and very professional. The members of the PRT are grateful for the time and effort applied by all of these individuals and hope that the contents of this report will help to make the X-33 program a success.

Derived from text

X-33 Reusable Launch Vehicle; Guidance (Motion); Flight Control; Navigation

20000120758 Lee and Associates, LLC, USA

Support to X-33/Reusable Launch Vehicle Technology Program

Nov. 29, 2000; In English

Contract(s)/Grant(s): NASA Order H-31417-D; No Copyright; Avail: CASI; A03, Hardcopy

The Primary activities of Lee & Associates for the referenced Purchase Order has been in direct support of the

X-33/Reusable Launch Vehicle Technology Program. An independent review to evaluate the X-33 liquid hydrogen fuel tank failure, which recently occurred after-test of the starboard tank has been provided. The purpose of the Investigation team was to assess the tank design modifications, provide an assessment of the testing approach used by MSFC (Marshall Space Flight Center) in determining the flight worthiness of the tank, assessing the structural integrity, and determining the cause of the failure of the tank. The approach taken to satisfy the objectives has been for Lee & Associates to provide the expertise of Mr. Frank Key and Mr. Wayne Burton who have relevant experience from past programs and a strong background of experience in the fields critical to the success of the program. Mr. Key and Mr. Burton participated in the NASA established Failure Investigation Review Team to review the development and process data and to identify any design, testing or manufacturing weaknesses and potential problem areas. This approach worked well in satisfying the objectives and providing the Review Team with valuable information including the development of a Fault Tree. The detailed inputs were made orally in real time in the Review Team daily meetings. The results of the investigation were presented to the MSFC Center Director by the team on February 15, 2000. Attached are four charts taken from that presentation which includes 1) An executive summary, 2) The most probable cause, 3) Technology assessment, and 4) Technology Recommendations for Cryogenic tanks.
Author
X-33 Reusable Launch Vehicle; Fuel Tanks; Technology Assessment; Cryogenic Fluid Storage

20000120402 Boeing Co., Seattle, WA USA
Advanced High Temperature Structural Seals
Newquist, Charles W.; Verzemnieks, Juris; Keller, Peter C.; Shorey, Mark W.; Steinetz, Bruce, Technical Monitor; November 2000; In English
Contract(s)/Grant(s): F33615-95-D-3203; RTOP 242-90-0D; AF Proj. 0012
Report No.(s): NASA/CR-2000-210522; E-12489; NAS 1.26:210522; No Copyright; Avail: CASI; A03, Hardcopy

This program addresses the development of high temperature structural seals for control surfaces for a new generation of small reusable launch vehicles. Successful development will contribute significantly to the mission goal of reducing launch cost for small, 200 to 300 lb payloads. Development of high temperature seals is mission enabling. For instance, ineffective control surface seals can result in high temperature (3100 F) flows in the elevon area exceeding structural material limits. Longer sealing life will allow use for many missions before replacement, contributing to the reduction of hardware, operation and launch costs. During the first phase of this program the existing launch vehicle control surface sealing concepts were reviewed, the aerothermal environment for a high temperature seal design was analyzed and a mock up of an arc-jet test fixture for evaluating seal concepts was fabricated.
Author
Control Surfaces; Reusable Launch Vehicles; Seals (Stoppers); Structural Design; Research and Development

20000120039 FDC/NYMA, Inc., Hampton, VA USA
International Space Station Evolution Data Book, Volume 2, Evolution Concepts
Jorgensen, Catherine A., Editor; Antol, Jeffrey, Technical Monitor; October 2000; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS1-96013; RTOP 953-20-00
Report No.(s): NASA/SP-2000-6109/VOL2/REV1; L-18039B/VOL2/REV1; NAS 1.21:6109/VOL2/REV1; No Copyright; Avail: CASI; A07, Hardcopy

This report provides a focused and in-depth look at the opportunities and drivers for the enhancement and evolution of the International Space Station (ISS) during assembly and beyond the assembly complete stage. These enhancements would expand and improve the current baseline capabilities of the ISS and help to facilitate the commercialization of the ISS by the private sector. Volume 1 provides the consolidated overview of the ISS baseline systems; information on the current facilities available for pressurized and unpressurized payloads; and information on current plans for crew availability and utilization, resource timelines and margin summaries including power, thermal, and storage volumes; and an overview of the vehicle traffic model. Volume 2 includes discussions of advanced technologies being investigated for use on the ISS and potential commercial utilization activities being examined including proposed design reference missions (DRM's) and the technologies being assessed by the Pre-planned Program Improvement (P(sup 3) I) Working Group. This information is very high level and does not provide the relevant information necessary for detailed design efforts. This document is meant to educate readers on the ISS and to stimulate the generation of ideas for enhancement and utilization of the ISS, either by or for the government, academia, and commercial industry.
Author
International Space Station; Evolution (Development); Space Commercialization; Space Station Modules; Space Station Structures; Structural Design; Space Industrialization

20000108738 NASA Marshall Space Flight Center, Huntsville, AL USA

A Damage Resistance Comparison Between Candidate Polymer Matrix Composite Feedline Materials

Nettles, A. T; September 2000; In English

Report No.(s): NASA/TM-2000-210482; NAS 1.15:210482; M-992; No Copyright; Avail: CASI; A03, Hardcopy

As part of NASAs focused technology programs for future reusable launch vehicles, a task is underway to study the feasibility of using the polymer matrix composite feedlines instead of metal ones on propulsion systems. This is desirable to reduce weight and manufacturing costs. The task consists of comparing several prototype composite feedlines made by various methods. These methods are electron-beam curing, standard hand lay-up and autoclave cure, solvent assisted resin transfer molding, and thermoplastic tape laying. One of the critical technology drivers for composite components is resistance to foreign objects damage. This paper presents results of an experimental study of the damage resistance of the candidate materials that the prototype feedlines are manufactured from. The materials examined all have a 5-harness weave of IM7 as the fiber constituent (except for the thermoplastic, which is unidirectional tape laid up in a bidirectional configuration). The resin tested were 977-6, PR 520, SE-SA-1, RS-E3 (e-beam curable), Cycom 823 and PEEK. The results showed that the 977-6 and PEEK were the most damage resistant in all tested cases.

Author

Composite Materials; Curing; Damage; Technology Assessment; Harnesses; Manufacturing; Polymer Matrix Composites

20000064109 NASA Glenn Research Center, Cleveland, OH USA

Ceramic Matrix Composites (CMC) Life Prediction Method Development

Levine, Stanley R.; Calomino, Anthony M.; Ellis, John R.; Halbig, Michael C.; Mital, Subodh K.; Murthy, Pappu L.; Opila, Elizabeth J.; Thomas, David J.; Thomas-Ogbuji, Linus U.; Verrilli, Michael J.; May 2000; In English; 24th Composites, Materials and Structures, 24-28 Jan. 2000, Cape Canaveral, FL, USA

Contract(s)/Grant(s): RTOP 242-82-77

Report No.(s): NASA/TM-2000-210052; E-12253; NAS 1.15:210052; No Copyright; Avail: CASI; A03, Hardcopy

Advanced launch systems (e.g., Reusable Launch Vehicle and other Shuttle Class concepts, Rocket-Based Combine Cycle, etc.), and interplanetary vehicles will very likely incorporate fiber reinforced ceramic matrix composites (CMC) in critical propulsion components. The use of CMC is highly desirable to save weight, to improve reuse capability, and to increase performance. CMC candidate applications are mission and cycle dependent and may include turbopump rotors, housings, combustors, nozzle injectors, exit cones or ramps, and throats. For reusable and single mission uses, accurate prediction of life is critical to mission success. The tools to accomplish life prediction are very immature and not oriented toward the behavior of carbon fiber reinforced silicon carbide (C/SiC), the primary system of interest for a variety of space propulsion applications. This paper describes an approach to satisfy the need to develop an integrated life prediction system for CMC that addresses mechanical durability due to cyclic and steady thermomechanical loads, and takes into account the impact of environmental degradation.

Author

Ceramic Matrix Composites; Technology Utilization; Silicon Carbides; Prediction Analysis Techniques; Life (Durability); Fiber Composites; Durability

20000062016 NASA Langley Research Center, Hampton, VA USA

Parametric Weight Comparison of Advanced Metallic, Ceramic Tile, and Ceramic Blanket Thermal Protection Systems

Myers, David E.; Martin, Carl J.; Blosser, Max L.; June 2000; In English

Contract(s)/Grant(s): RTOP 242-33-01-50

Report No.(s): NASA/TM-2000-210289; L-17954; NAS 1.15:210289; No Copyright; Avail: CASI; A03, Hardcopy

A parametric weight assessment of advanced metallic panel, ceramic blanket, and ceramic tile thermal protection systems (TPS) was conducted using an implicit, one-dimensional (I-D) finite element sizing code. This sizing code contained models to account for coatings fasteners, adhesives, and strain isolation pads. Atmospheric entry heating profiles for two vehicles, the Access to Space (ATS) vehicle and a proposed Reusable Launch Vehicle (RLV), were used to ensure that the trends were not unique to a certain trajectory. Ten TPS concepts were compared for a range of applied heat loads and substructural heat capacities to identify general trends. This study found the blanket TPS concepts have the lightest weights over the majority of their applicable ranges, and current technology ceramic tiles and metallic TPS concepts have similar weights. A proposed, state-of-the-art metallic system which uses a higher temperature alloy and efficient multilayer insulation was predicted to be

significantly lighter than the ceramic tile stems and approaches blanket TPS weights for higher integrated heat loads.
Author
Ceramics; Fasteners; Isolation; Loads (Forces); Multilayer Insulation; Thermal Protection; Tiles; Trajectories

20000052923 NASA Langley Research Center, Hampton, VA USA
Fatigue Crack Growth Rate Test Results for Al-Li 2195 Parent Metal, Variable Polarity Plasma Arc Welds and Friction Stir Welds
Hafley, Robert A.; Wagner, John A.; Domack, Marcia S.; May 2000; In English
Contract(s)/Grant(s): RTOP 242-20-05
Report No.(s): NASA/TM-2000-210098; NAS 1.15:210098; L-17984; No Copyright; Avail: CASI; A03, Hardcopy

The fatigue crack growth rate of aluminum-lithium (Al-Li) alloy 2195 plate and weldments was determined at 200-F, ambient temperature and -320-F. The effects of stress ratio (R), welding process, orientation and thickness were studied. Results are compared with plate data from the Space Shuttle Super Lightweight Tank (SLWT) allowables program. Data from the current series of tests, both plate and weldment, falls within the range of data generated during the SLWT allowables program.
Author
Aluminum-Lithium Alloys; Crack Propagation; Welded Joints; Weld Tests; Cracking (Fracturing); Fatigue (Materials); Cracks

20000037777 NASA Marshall Space Flight Center, Huntsville, AL USA
Test Report for NASA MSFC Support of the Linear Aerospike SR-71 Experiment (LASRE)
Elam, S. K.; February 2000; In English
Report No.(s): NASA/TM-2000-210076; NAS 1.15:210076; M-973; No Copyright; Avail: CASI; A07, Hardcopy

The Linear Aerospike SR-71 Experiment (LASRE) was performed in support of the Reusable Launch Vehicle (RLV) program to help develop a linear aerospike engine. The objective of this program was to operate a small aerospike engine at various speeds and altitudes to determine how slipstreams affect the engine's performance. The joint program between government and industry included NASA!s Dryden Flight Research Center, The Air Force's Phillips Laboratory, NASA's Marshall Space Flight Center, Lockheed Martin Skunkworks, Lockheed-Martin Astronautics, and Rocketdyne Division of Boeing North American. Ground testing of the LASRE engine produced two successful hot-fire tests, along with numerous cold flows to verify sequencing and operation before mounting the assembly on the SR-71. Once installed on the aircraft, flight testing performed several cold flows on the engine system at altitudes ranging from 30,000 to 50,000 feet and Mach numbers ranging from 0.9 to 1.5. The program was terminated before conducting hot-fires in flight because excessive leaks in the propellant supply systems could not be fixed to meet required safety levels without significant program cost and schedule impacts.
Author
Aerospike Engines; Reusable Launch Vehicles; SR-71 Aircraft; Liquid Propellant Rocket Engines; Propulsion System Performance; Engine Tests

20000033847 NASA Glenn Research Center, Cleveland, OH USA
Recent Advances and Applications in Cryogenic Propellant Densification Technology
Tomsik, Thomas M.; March 2000; In English; 12th Intersociety Cryogenic, 5-9 Mar. 2000, Atlanta, GA, USA
Contract(s)/Grant(s): RTOP 242-33-0C
Report No.(s): NASA/TM-2000-209941; E-12189; NAS 1.15:209941; No Copyright; Avail: CASI; A03, Hardcopy

This purpose of this paper is to review several historical cryogenic test programs that were conducted at the NASA Glenn Research Center (GRC), Cleveland, Ohio over the past fifty years. More recently these technology programs were intended to study new and improved denser forms of liquid hydrogen (LH2) and liquid oxygen (LO2) cryogenic rocket fuels. Of particular interest are subcooled cryogenic propellants. This is due to the fact that they have a significantly higher density (eg. triple-point hydrogen, slush etc.), a lower vapor pressure and improved cooling capacity over the normal boiling point cryogen. This paper, which is intended to be a historical technology overview, will trace the past and recent development and testing of small and large-scale propellant densification production systems. Densifier units in the current GRC fuels program, were designed and are capable of processing subcooled LH2 and L02 propellant at the X33 Reusable Launch Vehicle (RLV) scale. One final objective of this technical briefing is to discuss some of the potential benefits and application which propellant densification technology may offer the industrial cryogenics production and end-user community. Density enhancements to

cryogenic propellants (LH2, LO2, CH4) in rocket propulsion and aerospace application have provided the opportunity to either increase performance of existing launch vehicles or to reduce the overall size, mass and cost of a new vehicle system.
Author
Cryogenic Rocket Propellants; Densification; Technology Assessment; Liquid Hydrogen; Liquid Oxygen

20000024930 NASA Kennedy Space Center, Cocoa Beach, FL USA
Design, Development,and Testing of Umbillical System Mechanisms for the X-33 Advanced Technology Demonstrator
Littlefield, Alan C.; Melton, Gregory S.; Dec. 1999; In English; 34th Aerospace Mechanisms, 10-12 May 2000, Greenbelt, MD, USA
Report No.(s): NASA/TM-1999-208562; NAS1.15:208562; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 Advanced Technology Demonstrator is an un-piloted, vertical take-off, horizontal landing spacecraft. The purpose of the X-33 program is to demonstrate technologies that will dramatically lower the cost of access to space. The rocket-powered X-33 will reach an altitude of up to 100 km and speeds between Mach 13 and 15. Fifteen flight tests are planned, beginning in 2000. Some of the key technologies demonstrated will be the linear aerospike engine, improved thermal protection systems, composite fuel tanks and reduced operational timelines. The X-33 vehicle umbilical connections provide monitoring, power, cooling, purge, and fueling capability during horizontal processing and vertical launch operations. Two 'rise-ofF' umbilicals for the X-33 have been developed, tested, and installed. The X-33 umbilical systems mechanisms incorporate several unique design features to simplify horizontal operations and provide reliable disconnect during launch.
Author
Cost Reduction; Technology Assessment; Flight Tests; Proving; Spacecraft Launching; Vertical Landing; X-33 Reusable Launch Vehicle

20000017931 NASA Glenn Research Center, Cleveland, OH USA
Advanced Electric Propulsion for RLV Launched Geosynchronous Spacecraft
Oleson, Steven; December 1999; In English; 26th 26th International Electric Propulsion Conference, 17-21 Oct. 1999, Kitakyushu, Japan
Contract(s)/Grant(s): RTOP 632-1B-1C
Report No.(s): NASA/TM-1999-209646; NAS 1.15:209646; IEPC-99-185; E-11994; No Copyright; Avail: CASI; A03, Hardcopy

Solar Electric Propulsion (SEP) when used for station keeping and final orbit insertion has been shown to increase a geostationary satellite's payload when launched by existing expendable launch vehicles. In the case of reusable launch vehicles or expendable launch vehicles where an upper stage is an expensive option, this methodology can be modified by using the existing on-board apogee chemical system to perform a perigee burn and then letting the electric propulsion system complete the transfer to geostationary orbit. The elimination of upper stages using on-board chemical and electric propulsion systems was thus examined for GEO spacecraft. Launch vehicle step-down from an Atlas IIAR to a Delta 7920 (no upper stage) was achieved using expanded on-board chemical tanks, 40 kW payload power for electric propulsion, and a 60 day elliptical to GEO SEP orbit insertion. Optimal combined chemical and electric trajectories were found using SEPSPOT. While Hall and ion thrusters provided launch vehicle step-down and even more payload for longer insertion times, NH3 arcjets had insufficient performance to allow launch vehicle step-down. Degradation levels were only 5% to 7% for launch step-down cases using advanced solar arrays. Results were parameterized to allow comparisons for future reusable launch vehicles. Results showed that for an 8 W/kg initial power/launch mass power density spacecraft, 50% to 100% more payload can be launched using this method.
Author
Electric Propulsion; Propulsion System Configurations; Propulsion System Performance; Spacecraft Propulsion; Reusable Launch Vehicles; Geosynchronous Orbits; Synchronous Platforms; Synchronous Satellites

20000004220 NASA Langley Research Center, Hampton,VA USA, NASA Langley Research Center, Hampton,VA USA
Compute as Fast as the Engineers Can Think! ULTRAFAST COMPUTING TEAM FINAL REPORT
Biedron, R. T.; Mehrotra, P.; Nelson, M. L.; Preston, M. L.; Rehder, J. J.; Rogersm J. L.; Rudy, D. H.; Sobieski, J.; Storaasli, O. O.; Sep. 1999; In English
Contract(s)/Grant(s): RTOP 509-10-31-03
Report No.(s): NASA/TM-1999-209715; L-17891; NAS 1.15:209715; No Copyright; Avail: CASI; A04, Hardcopy

This report documents findings and recommendations by the Ultrafast Computing Team (UCT). In the period 10-12/98,

UCT reviewed design case scenarios for a supersonic transport and a reusable launch vehicle to derive computing requirements necessary for support of a design process with efficiency so radically improved that human thought rather than the computer paces the process. Assessment of the present computing capability against the above requirements indicated a need for further improvement in computing speed by several orders of magnitude to reduce time to solution from tens of hours to seconds in major applications. Evaluation of the trends in computer technology revealed a potential to attain the postulated improvement by further increases of single processor performance combined with massively parallel processing in a heterogeneous environment. However, utilization of massively parallel processing to its full capability will require redevelopment of the engineering analysis and optimization methods, including invention of new paradigms. To that end UCT recommends initiation of a new activity at LaRC called Computational Engineering for development of new methods and tools geared to the new computer architectures in disciplines, their coordination, and validation and benefit demonstration through applications.
Author
Computer Systems Design; Architecture (Computers); Axioms; Heterogeneity; Massively Parallel Processors

19990117247 NASA Langley Research Center, Hampton, VA USA
Multidisciplinary Optimization Branch Experience Using iSIGHT Software
Padula, S. L.; Korte, J. J.; Dunn, H. J.; Salas, A. O.; November 1999; In English
Contract(s)/Grant(s): RTOP 242-33-03-05; RTOP 581-20-31-01; RTOP 509-10-11-01
Report No.(s): NASA/TM-1999-209714; NAS 1.15:209714; L-17914; No Copyright; Avail: CASI; A03, Hardcopy

The Multidisciplinary Optimization (MDO) Branch at NASA Langley is investigating frameworks for supporting multidisciplinary analysis and optimization research. A framework provides software and system services to integrate computational tasks and allows the researcher to concentrate more on the application and less on the programming details. A framework also provides a common working environment and a full range of optimization tools, and so increases the productivity of multidisciplinary research teams. Finally, a framework enables staff members to develop applications for use by disciplinary experts in other organizations. This year, the MDO Branch has gained experience with the iSIGHT framework. This paper describes experiences with four aerospace applications, including: (1) reusable launch vehicle sizing, (2) aerospike nozzle design, (3) low-noise rotorcraft trajectories, and (4) acoustic liner design. Brief overviews of each problem are provided, including the number and type of disciplinary codes and computation time estimates. In addition, the optimization methods, objective functions, design variables, and constraints are described for each problem. For each case, discussions on the advantages and disadvantages of using the iSIGHT framework are provided as well as notes on the ease of use of various advanced features and suggestions for areas of improvement.
Author
Design Analysis; Multidisciplinary Design Optimization; Computer Aided Design; Systems Engineering; Computer Programs; Evaluation; Computer Systems Performance

19990113121 NASA Dryden Flight Research Center, Edwards, CA USA
Evaluation of the Linear Aerospike SR-71 Experiment (LASRE) Oxygen Sensor
Ennix, Kimberly A.; Corpening, Griffin P.; Jarvis, Michele; Chiles, Harry R.; November 1999; In English; 9th 9th International Space Planes and Hypersonic Systems, 1-5 Nov. 1999, Norfolk, VA, USA
Contract(s)/Grant(s): RTOP 529-31-44-00-32
Report No.(s): NASA/TM-1999-206589; NAS 1.15:206589; H-2377; No Copyright; Avail: CASI; A03, Hardcopy

The Linear Aerospike SR-71 Experiment (LASRE) was a propulsion flight experiment for advanced space vehicles such as the X-33 and reusable launch vehicle. A linear aerospike rocket engine was integrated into a semi-span of an X-33-like lifting body shape (model), and carried on top of an SR-71 aircraft at NASA Dryden Flight Research Center. Because no flight data existed for aerospike nozzles, the primary objective of the LASRE flight experiment was to evaluate flight effects on the engine performance over a range of altitudes and Mach numbers. Because it contained a large quantity of energy in the form of fuel, oxidizer, hypergolics, and gases at very high pressures, the LASRE propulsion system posed a major hazard for fire or explosion. Therefore, a propulsion-hazard mitigation system was created for LASRE that included a nitrogen purge system. Oxygen sensors were a critical part of the nitrogen purge system because they measured purge operation and effectiveness. Because the available oxygen sensors were not designed for flight testing, a laboratory study investigated oxygen-sensor characteristics and accuracy over a range of altitudes and oxygen concentrations. Laboratory test data made it possible to properly calibrate the sensors for flight. Such data also provided a more accurate error prediction than the manufacturer's

specification. This predictive accuracy increased confidence in the sensor output during critical phases of the flight. This paper presents the findings of this laboratory test.
Author
Aerospike Engines; Liquid Nitrogen; Gas Detectors; Oxygen Analyzers; Purging

19990113117 Starcraft Boosters, Inc., Houston, TX USA, Starcraft Boosters, Inc., Canyon Lake, TX USA
The StarBooster System: A Cargo Aircraft for Space
Davis, Hubert P.; Dula, Arthur M.; McLaughlin, Don; Frassanito, John; Andrews, Jason, Editor; Oct. 27, 1999; In English
Contract(s)/Grant(s): NASA Order H-30550-D
Report No.(s): SBI-1999-001A; No Copyright; Avail: CASI; A04, Hardcopy

Starcraft Boosters has developed a different approach for lowering the cost of access to space. We propose developing a new aircraft that will house an existing expendable rocket stage. This vehicle, termed StarBooster, will be the first stage of a family of launch vehicles. By combining these elements, we believe we can reduce the cost and risk of fielding a new partially reusable launch system. This report summarizes the work performed on the StarBooster concept since the company's inception in 1996. Detailed analyses are on-going and future reports will focus on the maturation of the vehicle and system design.
Derived from text
Booster Rocket Engines; Reusable Launch Vehicles; Propulsion System Performance; Space Transportation

19990110310 NASA Dryden Flight Research Center, Edwards, CA USA
Flush Airdata Sensing (FADS) System Calibration Procedures and Results for Blunt Forebodies
Cobleigh, Brent R.; Whitmore, Stephen A.; Haering, Edward A., Jr.; Borrer, Jerry; Roback, V. Eric; November 1999; In English; 9th International Space Planes and Hypersonic Systems and Technologies, 1-5 Nov. 1999, Norfolk, VA, USA
Contract(s)/Grant(s): RTOP 242-33-02-00-23
Report No.(s): NASA/TP-1999-209012; NAS 1.60:209012; H-2379; AIAA Paper 99-4816; No Copyright; Avail: CASI; A03, Hardcopy

Blunt-forebody pressure data are used to study the behavior of the NASA Dryden Flight Research Center flush airdata sensing (FADS) pressure model and solution algorithm. The model relates surface pressure measurements to the airdata state. Spliced from the potential flow solution for uniform flow over a sphere and the modified Newtonian impact theory, the model was shown to apply to a wide range of blunt-forebody shapes and Mach numbers. Calibrations of a sphere, spherical cones, a Rankine half body, and the F-14, F/A-18, X-33, X-34, and X-38 configurations are shown. The three calibration parameters are well-behaved from Mach 0.25 to Mach 5.0, an angle-of-attack range extending to greater than 30 deg, and an angle-of-sideslip range extending to greater than 15 deg. Contrary to the sharp calibration changes found on traditional pitot-static systems at transonic speeds, the FADS calibrations are smooth, monotonic functions of Mach number and effective angles of attack and sideslip. Because the FADS calibration is sensitive to pressure port location, detailed measurements of the actual pressure port locations on the flight vehicle are required and the wind-tunnel calibration model should have pressure ports in similar locations. The procedure for calibrating a FADS system is outlined.
Author
Calibrating; Detection; Forebodies; Mathematical Models; Pressure Measurement

19990108597 NASA Dryden Flight Research Center, Edwards,CA USA
Evaluation of the Linear Aerospike SR-71 Experiment (LASRE) Oxygen Sensor
Ennix, Kimberly A.; Corpening, Griffin P.; Jarvis, Michele; Chiles, Harry R.; November 1999; In English; 9th Space Planes and Hypersonic Systems, 1-5 Nov. 1999, Norfolk, VA, USA; Original contains color illustration
Contract(s)/Grant(s): RTOP 529-31-44
Report No.(s): NASA/TM-1999-206589; H-2377; NAS 1.15:206589; Copyright; Avail: CASI; A03, Hardcopy

The Linear Aerospike SR-71 Experiment (LASRE) was a propulsion flight experiment for advanced space vehicles such as the X-33 and reusable launch vehicle. A linear aerospike rocket engine was integrated into a semi-span of an X-33-like lifting body shape (model), and carried on top of an SR-71 aircraft at NASA Dryden Flight Research Center. Because no flight data existed for aerospike nozzles, the primary objective of the LASRE flight experiment was to evaluate flight effects on the engine performance over a range of altitudes and Mach numbers. Because it contained a large quantity of energy in the form of fuel, oxidizer, hypergolics, and gases at very high pressures, the LASRE propulsion system posed a major hazard for fire or explosion. Therefore, a propulsion-hazard mitigation system was created for LASRE that included a nitrogen purge system.

Oxygen sensors were a critical part of the nitrogen purge system because they measured purge operation and effectiveness. Because the available oxygen sensors were not designed for flight testing, a laboratory study investigated oxygen-sensor characteristics and accuracy over a range of altitudes and oxygen concentrations. Laboratory test data made it possible to properly calibrate the sensors for flight. Such data also provided a more accurate error prediction than the manufacturer's specification. This predictive accuracy increased confidence in the sensor output during critical phases of the flight. This paper presents the findings of this laboratory test.
Author
Aerospike Engines; Oxygen; SR-71 Aircraft; Gas Detectors; Propulsion System Configurations

19990106559 NASA Dryden Flight Research Center, Edwards, CA USA
Propellant Feed System Leak Detection: Lessons Learned From the Linear Aerospike SR-71 Experiment (LASRE)
Hass, Neal; Mizukami, Masashi; Neal, Bradford A.; St. John, Clinton; Beil, Robert J.; Griffin, Timothy P.; November 1999; In English; 9th International Space Planes and Hypersonic Systems, 1-5 Nov. 1999, Norfolk, VA, USA
Contract(s)/Grant(s): RTOP 243-33-02
Report No.(s): NASA/TM-1999-206590; NAS 1.15:206590; H-2378; Copyright; Avail: CASI; A03, Hardcopy

This paper presents pertinent results and assessment of propellant feed system leak detection as applied to the Linear Aerospike SR-71 Experiment (LASRE) program flown at the NASA Dryden Flight Research Center, Edwards, California. The LASRE was a flight test of an aerospike rocket engine using liquid oxygen and high-pressure gaseous hydrogen as propellants. The flight safety of the crew and the experiment demanded proven technologies and techniques that could detect leaks and assess the integrity of hazardous propellant feed systems. Point source detection and systematic detection were used. Point source detection was adequate for catching gross leakage from components of the propellant feed systems, but insufficient for clearing LASRE to levels of acceptability. Systematic detection, which used high-resolution instrumentation to evaluate the health of the system within a closed volume, provided a better means for assessing leak hazards. Oxygen sensors detected a leak rate of approximately 0.04 cubic inches per second of liquid oxygen. Pressure sensor data revealed speculated cryogenic boiloff through the fittings of the oxygen system, but location of the source(s) was indeterminable. Ultimately, LASRE was cancelled because leak detection techniques were unable to verify that oxygen levels could be maintained below flammability limits.
Author
Leakage; Aerospike Engines; Gas Detectors; Pressure Sensors; Feed Systems; Flight Tests

19990105883 NASA Dryden Flight Research Center, Edwards, CA USA
Flush Airdata Sensing (FADS) System Calibration Procedures and Results for Blunt Forebodies
Cobleigh, Brent R.; Whitmore, Stephen A.; Haering, Edward A., Jr.; Borrer, Jerry; Roback, V. Eric; November 1999; In English; 9th Space Planes and Hypersonic Systems and Technologies, 1-5 Nov. 1999, Norfolk, VA, USA
Contract(s)/Grant(s): RTOP 242-33-02-00-23
Report No.(s): NASA/TP-1999-209012; NAS 1.60:209012; H-2379; No Copyright; Avail: CASI; A03, Hardcopy

Blunt-forebody pressure data are used to study the behavior of the NASA Dryden Flight Research Center flush airdata sensing (FADS) pressure model and solution algorithm. The model relates surface pressure measurements to the airdata state. Spliced from the potential flow solution for uniform flow over a sphere and the modified Newtonian impact theory, the model was shown to apply to a wide range of blunt-forebody shapes and Mach numbers. Calibrations of a sphere, spherical cones, a Rankine half body, and the F-14, F/A-18, X-33, X-34, and X-38 configurations are shown. The three calibration parameters are well-behaved from Mach 0.25 to Mach 5.0, an angle-of-attack range extending to greater than 30 deg, and an angle-of-sideslip range extending to greater than 15 deg. Contrary to the sharp calibration changes found on traditional pitot-static systems at transonic speeds, the FADS calibrations are smooth, monotonic functions of Mach number and effective angles of attack and sideslip. Because the FADS calibration is sensitive to pressure port location, detailed measurements of the actual pressure port locations on the flight vehicle are required and the wind-tunnel calibration model should have pressure ports in similar locations. The procedure for calibrating a FADS system is outlined.
Author
Procedures; Calibrating; Forebodies; Mathematical Models; Algorithms; Potential Flow; Pressure Measurement; Newton Theory

19990100643 Lockheed Martin Engineering and Sciences Co., Hampton, VA USA
An Inviscid Computational Study of an X-33 Configuration at Hypersonic Speeds
Prabhu, Ramadas K.; July 1999; In English
Contract(s)/Grant(s): NAS1-96014; RTOP 242-80-01-01
Report No.(s): NASA/CR-1999-209366; NAS 1.26:209366; No Copyright; Avail: CASI; A03, Hardcopy

This report documents the results of a study conducted to compute the inviscid longitudinal aerodynamic characteristics of a simplified X-33 configuration. The major components of the X-33 vehicle, namely the body, the canted fin, the vertical fin, and the body-flap, were simulated in the CFD (Computational Fluid Dynamic) model. The rear-ward facing surfaces at the base including the aerospike engine surfaces were not simulated. The FELISA software package consisting of an unstructured surface and volume grid generator and two inviscid flow solvers was used for this study. Computations were made for Mach 4.96, 6.0, and 10.0 with perfect gas air option, and for Mach 10 with equilibrium air option with flow condition of a typical point on the X-33 flight trajectory. Computations were also made with CF4 gas option at Mach 6.0 to simulate the CF4 tunnel flow condition. An angle of attack range of 12 to 48 deg was covered. The CFD results were compared with available wind tunnel data. Comparison was good at low angles of attack; at higher angles of attack (beyond 25 deg) some differences were found in the pitching moment. These differences progressively increased with increase in angle of attack, and are attributed to the viscous effects. However, the computed results showed the trends exhibited by the wind tunnel data.
Author

Computational Fluid Dynamics; Inviscid Flow; Viscous Flow; Unstructured Grids (Mathematics); X-33 Reusable Launch Vehicle; Venturestar Launch Vehicle; Aerospike Engines

19990094229 NASA Langley Research Center, Hampton,VA USA
Cyclic Cryogenic Thermal-Mechanical Testing of an X-33/RLV Liquid Oxygen Tank Concept
Rivers, H. Kevin; September 1999; In English; Original contains color illustrations
Contract(s)/Grant(s): RTOP 242-23-02-13
Report No.(s): NASA/TM-1999-209560; L-17818; NAS 1.15:209560; No Copyright; Avail: CASI; A03, Hardcopy

An important step in developing a cost-effective, reusable, launch vehicle is the development of durable, lightweight, insulated, cryogenic propellant tanks. Current cryogenic tanks are expendable so most of the existing technology is not directly applicable to future launch vehicles. As part of the X-33/Reusable Launch Vehicle (RLV) Program, an experimental apparatus developed at the NASA Langley Research Center for evaluating the effects of combined, cyclic, thermal and mechanical loading on cryogenic tank concepts was used to evaluate cryogenic propellant tank concepts for Lockheed-Martin Michoud Space Systems. An aluminum-lithium (Al 2195) liquid oxygen tank concept, insulated with SS-1171 and PDL-1034 cryogenic insulation, is tested under simulated mission conditions, and the results of those tests are reported. The tests consists of twenty-five simulated Launch/Abort missions and twenty-five simulated flight missions with temperatures ranging from -320 F to 350 F and a maximum mechanical load of 71,300 lb. in tension.
Author

Cyclic Loads; Storage Tanks; Propellant Tanks; Load Tests; Load Testing Machines; Flight Simulation

19990081112 NASA Glenn Research Center, Cleveland, OH USA
Analysis of a New Rocket-Based Combined-Cycle Engine Concept at Low Speed
Yungster, S.; Trefny, C. J.; September 1999; In English; 35th Joint Propulsion, 20-24 Jun. 1999, Los Angeles, CA, USA
Contract(s)/Grant(s): NCC3-542; RTOP 505-90-5K
Report No.(s): NASA/TM-1999-209393; NAS 1.15:209393; E-11824; AIAA Paper 99-2393; ICOMP-99-05; No Copyright; Avail: CASI; A03, Hardcopy

An analysis of the Independent Ramjet Stream (IRS) cycle is presented. The IRS cycle is a variation of the conventional ejector-Ramjet, and is used at low speed in a rocket-based combined-cycle (RBCC) propulsion system. In this new cycle, complete mixing between the rocket and ramjet streams is not required, and a single rocket chamber can be used without a long mixing duct. Furthermore, this concept allows flexibility in controlling the thermal choke process. The resulting propulsion system is intended to be simpler, more robust, and lighter than an ejector-ramjet. The performance characteristics of the IRS cycle are analyzed for a new single-stage-to-orbit (SSTO) launch vehicle concept, known as 'Trailblazer.' The study is based on a quasi-one-dimensional model of the rocket and air streams at speeds ranging from lift-off to Mach 3. The numerical formulation is described in detail. A performance comparison between the IRS and ejector-ramjet cycles is also presented.
Author

Ramjet Engines; Rocket Engines; Single Stage to Orbit Vehicles; Propulsion System Performance; Propulsion System Configurations; Spacecraft Configurations

19990062664 NASA Glenn Research Center, Cleveland, OH USA

Inlet Development for a Rocket Based Combined Cycle, Single Stage to Orbit Vehicle Using Computational Fluid Dynamics

DeBonis, J. R.; Trefny, C. J.; Steffen, C. J., Jr.; June 1999; In English; 35th Joint Propulsion, 20-24 Jun. 1999, Los Angeles, CA, USA

Contract(s)/Grant(s): RTOP 523-61-13

Report No.(s): NASA/TM-1999-209279; NAS 1.15:209279; AIAA Paper 99-2239; E-11744; No Copyright; Avail: CASI; A03, Hardcopy

Design and analysis of the inlet for a rocket based combined cycle engine is discussed. Computational fluid dynamics was used in both the design and subsequent analysis. Reynolds averaged Navier-Stokes simulations were performed using both perfect gas and real gas assumptions. An inlet design that operates over the required Mach number range from 0 to 12 was produced. Performance data for cycle analysis was post processed using a stream thrust averaging technique. A detailed performance database for cycle analysis is presented. The effect ot vehicle forebody compression on air capture is also examined.

Author

Inlet Flow; Data Bases; Reliability Analysis; Reynolds Averaging; Simulation; Orbital Maneuvers; Design Analysis

19990062176 NASA Dryden Flight Research Center, Edwards,CA USA

Reconfigurable Flight Control Designs With Application to the X-33 Vehicle

Burken, John J.; Lu, Ping; Wu, Zhenglu; August 1999; In English; Guidance Navigation and Control, 9-11 Aug. 1999, Portland, OR, USA

Contract(s)/Grant(s): RTOP 242-33-02-00-23

Report No.(s): NASA/TM-1999-206582; H-2345; NAS 1.15:206582; AIAA Paper 99-4134; No Copyright; Avail: CASI; A03, Hardcopy

Two methods for control system reconfiguration have been investigated. The first method is a robust servomechanism control approach (optimal tracking problem) that is a generalization of the classical proportional-plus-integral control to multiple input-multiple output systems. The second method is a control-allocation approach based on a quadratic programming formulation. A globally convergent fixed-point iteration algorithm has been developed to make onboard implementation of this method feasible. These methods have been applied to reconfigurable entry flight control design for the X-33 vehicle. Examples presented demonstrate simultaneous tracking of angle-of-attack and roll angle commands during failures of the right body flap actuator. Although simulations demonstrate success of the first method in most cases, the control-allocation method appears to provide uniformly better performance in all cases.

Author

Control Systems Design; Mimo (Control Systems); X-33 Reusable Launch Vehicle; Servomechanisms; Proportional Control; Flight Control; Launch Vehicle Configurations

19990053005 NASA Glenn Research Center, Cleveland, OH USA

NASA Glenn Research Center's Hypersonic Propulsion Program

Palac, Donald T.; May 1999; In English; Airbreathing Engines, 5-10 Sep. 1999, Florence, Italy

Contract(s)/Grant(s): RTOP 523-61-13

Report No.(s): NASA/TM-1999-209185; E-11696; NAS 1.15:209185; No Copyright; Avail: CASI; A02, Hardcopy

NASA Glenn Research Center (GRC), as NASA's lead center for aeropropulsion, is responding to the challenge of reducing the cost of space transportation through the integration of air-breathing propulsion into launch vehicles. Air-breathing launch vehicle (ABLV) propulsion requires a marked departure from traditional propulsion applications. and stretches the technology of both rocket and air-breathing propulsion. In addition, the demands of the space launch mission require an unprecedented level of integration of propulsion and vehicle systems. GRC is responding with a program with rocket-based combined cycle (RBCC) propulsion technology as its main focus. RBCC offers the potential for simplicity, robustness, and performance that may enable low-cost single-stage-to-orbit (SSTO) transportation. Other technologies, notably turbine-based combined cycle (TBCC) propulsion, offer benefits such as increased robustness and greater mission flexibility, and are being advanced, at a slower pace, as part of GRC's program in hypersonics.

Author

Air Breathing Boosters; Air Breathing Engines; Launch Vehicles; Spacecraft Launching

19990036776 NASA Langley Research Center, Hampton,VA USA

X-33 (Rev-F) Aeroheating Results of Test 6770 in NASA Langley 20-Inch Mach 6 Air Tunnel

Berry, Scott A.; Horvath, Thomas J.; Kowalkowski, Matthew K.; Liechty, Derek S.; March 1999; In English; Original contains color illustrations

Contract(s)/Grant(s): RTOP 242-80-01-01

Report No.(s): NASA/TM-1999-209122; L-17824; NAS 1.15:209122; No Copyright; Avail: CASI; A05, Hardcopy

Aeroheating characteristics of the X-33 Rev-F configuration have been experimentally examined in the Langley 20-Inch Mach 6 Air Tunnel (Test 6770). Global surface heat transfer distributions, surface streamline patterns, and shock shapes were measured on a 0.013-scale model at Mach 6 in air. Parametric variations include angles-of-attack of 20-deg, 30-deg, and 40-deg; Reynolds numbers based on model length of 0.9 to 4.9 million; and body-flap deflections of 10-deg and 20-deg. The effects of discrete roughness elements on boundary layer transition, which included trip height, size, and location, both on and off the windward centerline, were investigated. This document is intended to serve as a quick release of preliminary data to the X-33 program; analysis is limited to observations of the experimental trends in order to expedite dissemination.

Author

X-33 Reusable Launch Vehicle; Aerodynamic Heating; Aerodynamic Configurations; Wind Tunnel Tests; Hypersonic Speed; Heat Transfer; Boundary Layer Transition; Aerothermodynamics

19990032557 NASA Marshall Space Flight Center, Huntsville, AL USA

General Public Space Travel and Tourism, Volume 1, Executive Summary

ONeil, Daniel, Compiler; Bekey, Ivan; Mankins, John; Rogers, Thomas F.; Stallmer, Eric W.; March 1998; In English; Summary of a Space Act Agreement Study, 19-21 Feb. 1997, Washington, DC, USA

Report No.(s): NASA/NP-1998-03-11-MSFC/VOL1; NAS 1.83:03-11-MSFC/VOL1; No Copyright; Avail: CASI; A03, Hardcopy

Travel and tourism is one of the world's largest businesses. Its gross revenues exceed $400 billion per year in the U.S. alone, and it is our second largest employer. U.S. private sector business revenues in the space information area now approximate $10 billion per year, and are increasing rapidly. Not so in the human spaceflight area. After spending $100s of billions (1998 dollars) in public funds thereon, and continuing to spend over $5 billion per year, the government is still the only customer for human spaceflight goods and services. Serious and detailed consideration was first given to the possibility of space being opened up to trips by the general public three decades ago, and some initial attempts to do so were made a dozen years ago. But the difficulties were great and the Challenger disaster put an end to them. In recent years professional space tourism studies have been conducted in the UK, Germany and, especially, Japan. In the U.S., technological progress has been pronounced; we have had nearly a decade's experience in seeing our astronauts travel to-from low Earth orbit (LEO) safely, and we expect to commence assembly of a LEO space station housing a half-dozen people this year. Too, NASA and our space industry now have new and promising space transportation development programs underway, especially the X-33 and X-34 programs, and some related, further generation, basic technology development programs. And five private companies are also working on the design of new surface - LEO vehicles. The first professional space tourism market studies have been conducted in several countries in the past few years, especially in Japan and here. The U.S. study makes it clear that, conceptually, tens of millions of us would like to take a trip to space if we could do so with reasonable safety, comfort and reliability, and at an acceptable price. Initial businesses will address the desires of those willing to pay a greater price and accept a greater risk. A two-year cooperative Space Act agreement study has been conducted by our National Aeronautics and Space Administration and the Space Transportation Association. It was conducted by NASA and STA study leaders drawing upon the competence, experience and hard-nosed imagination of a national Steering Group and scores of attendees at a multi-day Workshop. The study has involved scores of professionals and business people from various areas: astronauts; space booster technology and operations professionals; a hotel architect and a hotel operator; an airline planner; insurance underwriters; space sickness experts; space theme park designers; space and travel and tourism association and business executives; a space-related financier; university tourism and space policy experts; present and former space-responsible government officials; space entrepreneurs; space writers; This study concludes that serious national attention should now be given to activities that would enable the expansion of today's terrestrial space tourism businesses, and the creation of in-space travel and tourism businesses. Indeed, it concludes that, in time, it should become a very important part of our Country's overall commercial and civil space business-program structure.

Author

Aerospace Engineering; Low Earth Orbits; NASA Programs; Space Flight; Space Stations; Space Transportation; Travel; Space Habitats

19990028385 NASA Marshall Space Flight Center, Huntsville, AL USA

A Strategy for Integrating a Large Finite Element Model Using MSC NASTRAN/PATRAN: X-33 Lessons Learned

McGhee, D. S.; March 1999; In English

Report No.(s): NASA/TM-1999-209201; NAS 1.15:209201; M-917; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 vehicle is an advanced technology demonstrator sponsored by NASA. For the past 3 years the Structural Dynamics and Loads Branch of NASA's Marshall Space Flight Center has had the task of integrating the X-33 vehicle structural finite element model. In that time, five versions of the integrated vehicle model have been produced and a strategy has evolved that would benefit anyone given the task of integrating structural finite element models that have been generated by various modelers and companies. The strategy that has been presented here consists of six decisions that need to be made: purpose of models, units, common materials list, model numbering, interface control, and archive format. This strategy has been proven and expanded from experience on the X-33 vehicle.

Author

X-33 Reusable Launch Vehicle; Finite Element Method; Models; Nastran

19990026605 NASA Dryden Flight Research Center, Edwards,CA USA

A Base Drag Reduction Experiment on the X-33 Linear Aerospike SR-71 Experiment (LASRE) Flight Program

Whitmore, Stephen A.; Moes, Timothy R.; Mar. 1999; In English; 37th Aerospace Sciences, 11-14 Jan. 1999, Reno, NV, USA

Contract(s)/Grant(s): RTOP 242-33-02

Report No.(s): NASA/TM-1999-206575; H-2333; NAS 1.15:206575; AIAA Paper 99-0277; No Copyright; Avail: CASI; A03, Hardcopy

Drag reduction tests were conducted on the LASRE/X-33 flight experiment. The LASRE experiment is a flight test of a roughly 20% scale model of an X-33 forebody with a single aerospike engine at the rear. The experiment apparatus is mounted on top of an SR-71 aircraft. This paper suggests a method for reducing base drag by adding surface roughness along the forebody. Calculations show a potential for base drag reductions of 8-14%. Flight results corroborate the base drag reduction, with actual reductions of 15% in the high-subsonic flight regime. An unexpected result of this experiment is that drag benefits were shown to persist well into the supersonic flight regime. Flight results show no overall net drag reduction. Applied surface roughness causes forebody pressures to rise and offset base drag reductions. Apparently the grit displaced streamlines outward, causing forebody compression. Results of the LASRE drag experiments are inconclusive and more work is needed. Clearly, however, the forebody grit application works as a viable drag reduction tool.

Author

Aerodynamic Drag; Drag Reduction; Flight Tests; Drag Measurement

19990025759 NASA Langley Research Center, Hampton, VA USA

Effective Thermal Conductivity of High Temperature Insulations for Reusable Launch Vehicles

Daryabeigi, Kamran; February 1999; In English

Contract(s)/Grant(s): RTOP 242-33-03-29

Report No.(s): NASA/TM-1999-208972; NAS 1.15:208972; L-17808; No Copyright; Avail: CASI; A03, Hardcopy

An experimental apparatus was designed to measure the effective thermal conductivity of various high temperature insulations subject to large temperature gradients representative of typical launch vehicle re-entry aerodynamic heating conditions. The insulation sample cold side was maintained around room temperature, while the hot side was heated to temperatures as high as 1800 degrees Fahrenheit. The environmental pressure was varied from 0.0001 to 760 torr. All the measurements were performed in a dry gaseous nitrogen environment. The effective thermal conductivity of Saffil, Q-Fiber felt, Cerachrome, and three multi-layer insulation configurations were measured.

Author

Thermal Conductivity; Multilayer Insulation; Reusable Launch Vehicles; Reentry Effects; Thermal Insulation

19990018443 NASA Langley Research Center, Hampton,VA USA

Aeronautical Engineering: A Continuing Bibliography with Indexes

Feb. 05, 1999; In English

Report No.(s): NASA/SP-1999-7037/SUPPL393; NAS 1.21:7037/SUPPL393; No Copyright; Avail: CASI; A04, Hardcopy

This supplemental issue of Aeronautical Engineering: A Continuing Bibliography with Indexes lists reports, articles, and other documents recently announced in the NASA STI Database. The coverage includes documents on the engineering and theoretical aspects of design, construction, evaluation, testing, operation, and performance of aircraft (including aircraft

engines) and associated components, equipment, and systems. It also includes research and development in aerodynamics, aeronautics, and ground support equipment for aeronautical vehicles. Each entry in the publication consists of a standard bibliographic citation accompanied, in most cases, by an abstract. The NASA CASI price code table, addresses of organizations, and document availability information are included before the abstract section. Two indexes-subject and author are included after the abstract section.
Derived from text
Bibliographies; Aeronautical Engineering

19990009863 NASA Lewis Research Center, Cleveland,OH USA
A Hazardous Gas Detection System for Aerospace and Commercial Applications
Hunter, G. W.; Neudeck, P. G.; Chen, L. - Y.; Makel, D. B.; Liu, C. C.; Wu, Q. H.; Knight, D.; Nov. 1998; In English; 34th Joint Propulsion, 12-15 Jul. 1998, Cleveland, OH, USA; Original contains color illustrations
Contract(s)/Grant(s): RTOP 254-02-0A
Report No.(s): NASA/TM-1998-208817; E-11419; NAS 1.15:208817; AIAA Paper 98-3614; No Copyright; Avail: CASI; A03, Hardcopy

The detection of explosive conditions in aerospace propulsion applications is important for safety and economic reasons. Microfabricated hydrogen, oxygen, and hydrocarbon sensors as well as the accompanying hardware and software are being developed for a range of aerospace safety applications. The development of these sensors is being done using MEMS (Micro ElectroMechanical Systems) based technology and SiC-based semiconductor technology. The hardware and software allows control and interrogation of each sensor head and reduces accompanying cabling through multiplexing. These systems are being applied on the X-33 and on an upcoming STS-95 Shuttle mission. A number of commercial applications are also being pursued. It is concluded that this MEMS-based technology has significant potential to reduce costs and increase safety in a variety of aerospace applications.
Author
Aerospace Engineering; Gas Detectors; Aerospace Safety; Space Transportation System; Space Transportation System Flights

19990008856 NASA Lewis Research Center, Cleveland, OH USA
A Hazardous Gas Detection System for Aerospace and Commercial Applications
Hunter, G. W.; Neudeck, P. G.; Chen, L.-Y.; Makel, D. B.; Liu, C. C.; Wu, Q. H.; Knight, D.; Nov. 1998; In English; 34th Propulsion, 12-15 Jul. 1998, Cleveland, OH, USA
Contract(s)/Grant(s): RTOP 254-02-0A
Report No.(s): NASA/TM-1998-208817; E-11419; NAS 1.15:208817; AIAA Paper 98-3614; No Copyright; Avail: CASI; A03, Hardcopy

The detection of explosive conditions in aerospace propulsion applications is important for safety and economic reasons. Microfabricated hydrogen, oxygen, and hydrocarbon sensors as well as the accompanying hardware and software are being, developed for a range of aerospace safety applications. The development of these sensors is being done using MEMS (Micro ElectroMechanical Systems) based technology and SiC-based semiconductor technology. The hardware and software allows control and interrocation of each sensor head and reduces accompanying cabling through multiplexing. These systems are being, applied on the X-33 and on an upcoming STS-95 Shuttle mission. A number of commercial applications are also being pursued. It is concluded that this MEMS-based technology has significant potential to reduce costs and increase safety in a variety of aerospace applications.
Author
Gas Detectors; Hazards; Computer Programs; Hardware; Aerospace Safety

19990008538 NASA Lewis Research Center, Cleveland, OH USA
Real-Time Sensor Validation, Signal Reconstruction, and Feature Detection for an RLV Propulsion Testbed
Jankovsky, Amy L.; Fulton, Christopher E.; Binder, Michael P.; Maul, William A., III; Meyer, Claudia M.; Oct. 1998; In English
Contract(s)/Grant(s): RTOP 242-33-01
Report No.(s): NASA/TM-1998-208807; NAS 1.15:208807; E-11392; No Copyright; Avail: CASI; A03, Hardcopy

A real-time system for validating sensor health has been developed in support of the reusable launch vehicle program. This system was designed for use in a propulsion testbed as part of an overall effort to improve the safety, diagnostic capability, and cost of operation of the testbed. The sensor validation system was designed and developed at the NASA Lewis Research

Center and integrated into a propulsion checkout and control system as part of an industry-NASA partnership, led by Rockwell International for the Marshall Space Flight Center. The system includes modules for sensor validation, signal reconstruction, and feature detection and was designed to maximize portability to other applications. Review of test data from initial integration testing verified real-time operation and showed the system to perform correctly on both hard and soft sensor failure test cases. This paper discusses the design of the sensor validation and supporting modules developed at LeRC and reviews results obtained from initial test cases.
Author
Systems Health Monitoring; Real Time Operation; Signal Processing; Checkout; Expert Systems; Sensors

19990008476 Computer Sciences Corp., Huntsville, AL USA
A Compendium of Wind Statistics and Models for the NASA Space Shuttle and Other Aerospace Vehicle Programs
Smith, O. E.; Adelfang, S. I.; Oct. 1998; In English
Contract(s)/Grant(s): NAS8-60000
Report No.(s): NASA/CR-1998-208859; NAS 1.26:208859; M-899; No Copyright; Avail: CASI; A06, Hardcopy

The wind profile with all of its variations with respect to altitude has been, is now, and will continue to be important for aerospace vehicle design and operations. Wind profile databases and models are used for the vehicle ascent flight design for structural wind loading, flight control systems, performance analysis, and launch operations. This report presents the evolution of wind statistics and wind models from the empirical scalar wind profile model established for the Saturn Program through the development of the vector wind profile model used for the Space Shuttle design to the variations of this wind modeling concept for the X-33 program. Because wind is a vector quantity, the vector wind models use the rigorous mathematical probability properties of the multivariate normal probability distribution. When the vehicle ascent steering commands (ascent guidance) are wind biased to the wind profile measured on the day-of-launch, ascent structural wind loads are reduced and launch probability is increased. This wind load alleviation technique is recommended in the initial phase of vehicle development. The vehicle must fly through the largest load allowable versus altitude to achieve its mission. The Gumbel extreme value probability distribution is used to obtain the probability of exceeding (or not exceeding) the load allowable. The time conditional probability function is derived from the Gumbel bivariate extreme value distribution. This time conditional function is used for calculation of wind loads persistence increments using 3.5-hour Jimsphere wind pairs. These increments are used to protect the commit-to-launch decision. Other topics presented include the Shuttle Shuttle load-response to smoothed wind profiles, a new gust model, and advancements in wind profile measuring systems. From the lessons learned and knowledge gained from past vehicle programs, the development of future launch vehicles can be accelerated. However, new vehicle programs by their very nature will require specialized support for new databases and analyses for wind, atmospheric parameters (pressure, temperature, and density versus altitude), and weather. It is for this reason that project managers are encouraged to collaborate with natural environment specialists early in the conceptual design phase. Such action will give the lead time necessary to meet the natural environment design and operational requirements, and thus, reduce development costs.
Author
Wind Profiles; Wind Effects; Mathematical Models; Wind Velocity Measurement; Gust Loads; Climbing Flight

19980236873 NASA Dryden Flight Research Center, Edwards, CA USA
The X-33 Extended Flight Test Range
Mackall, Dale A.; Sakahara, Robert; Kremer, Steven E.; Oct. 1998; In English; 39th International Telemetry Conference, 26-29 Oct. 1980, San Diego, CA, USA
Contract(s)/Grant(s): RTOP 242-33-02-00-23
Report No.(s): NASA/TM-1998-206557; NAS 1.15:206557; H-2261; No Copyright; Avail: CASI; A03, Hardcopy

Development of an extended test range, with range instrumentation providing continuous vehicle communications, is required to flight-test the X-33, a scaled version of a reusable launch vehicle. The extended test range provides vehicle communications coverage from California to landing at Montana or Utah. This paper provides an overview of the approaches used to meet X-33 program requirements, including using multiple ground stations, and methods to reduce problems caused by reentry plasma radio frequency blackout. The advances used to develop the extended test range show other hypersonic and access-to-space programs can benefit from the development of the extended test range.
Author
X-33 Reusable Launch Vehicle; Reusable Launch Vehicles; Hypersonics; Flight Tests; Recoverable Launch Vehicles; Plasma Frequencies; Communication

19980236871 NASA Ames Research Center, Moffett Field, CA USA, Boeing Co., Long Beach, CA USA

Results of Two-Stage Light-Gas Gun Development Efforts and Hypervelocity Impact Tests of Advanced Thermal Protection Materials

Cornelison, C. J.; Watts, Eric T.; Oct. 1998; In English; Original contains color illustrations

Contract(s)/Grant(s): RTOP 242-80-01

Report No.(s): NASA/TM-1998-112234; NAs 1.15:112234; A-9811760; No Copyright; Avail: CASI; A03, Hardcopy

Gun development efforts to increase the launching capabilities of the NASA Ames 0.5-inch two-stage light-gas gun have been investigated. A gun performance simulation code was used to guide initial parametric variations and hardware modifications, in order to increase the projectile impact velocity capability to 8 km/s, while maintaining acceptable levels of gun barrel erosion and gun component stresses. Concurrent with this facility development effort, a hypervelocity impact testing series in support of the X-33/RLV program was performed in collaboration with Rockwell International. Specifically, advanced thermal protection system materials were impacted with aluminum spheres to simulate impacts with on-orbit space debris. Materials tested included AETB-8, AETB-12, AETB-20, and SIRCA-25 tiles, tailorable advanced blanket insulation (TABI), and high temperature AFRSI (HTA). The ballistic limit for several Thermal Protection System (TPS) configurations was investigated to determine particle sizes which cause threshold TPS/structure penetration. Crater depth in tiles was measured as a function of impact particle size. The relationship between coating type and crater morphology was also explored. Data obtained during this test series was used to perform a preliminary analysis of the risks to a typical orbital vehicle from the meteoroid and space debris environment.

Author

Hypervelocity Impact; Thermal Protection; Space Debris; Meteoroid Concentration; Impact Velocity; Impact Tests; Computerized Simulation

19980236696 NASA Dryden Flight Research Center, Edwards, CA USA

X-33 Integrated Test Facility Extended Range Simulation

Sharma, Ashley; Oct. 1998; In English; 34th International Telemetry Conference, 26-29 Oct. 1998, San Diego, CA, USA

Contract(s)/Grant(s): RTOP 242-33-02-00-23

Report No.(s): NASA/TM-1998-206564; NAS 1.15:206564; H-2273; No Copyright; Avail: CASI; A03, Hardcopy

In support of the X-33 single-stage-to-orbit program, NASA Dryden Flight Research Center was selected to provide continuous range communications of the X-33 vehicle from launch at Edwards Air Force Base, California, through landing at Malmstrom Air Force Base Montana, or at Michael Army Air Field, Utah. An extensive real-time range simulation capability is being developed to ensure successful communications with the autonomous X-33 vehicle. This paper provides an overview of various levels of simulation, integration, and test being developed to support the X-33 extended range subsystems. These subsystems include the flight termination system, L-band command uplink subsystem, and S-band telemetry downlink subsystem.

Author

X-33 Reusable Launch Vehicle; Ground Support Systems; Military Air Facilities; Test Facilities; Uplinking; Radio Telemetry; Downlinking; Telecommunication

19980223961 NASA Dryden Flight Research Center, Edwards, CA USA

Flight Testing the Linear Aerospike SR-71 Experiment (LASRE)

Corda, Stephen; Neal, Bradford A.; Moes, Timothy R.; Cox, Timothy H.; Monaghan, Richard C.; Voelker, Leonard S.; Corpening, Griffin P.; Larson, Richard R.; Powers, Bruce G.; Sep. 1998; In English; 30th, 15-17 Sep. 1998, Reno, NV, USA

Contract(s)/Grant(s): RTOP 242-33-02-00-23

Report No.(s): NASA/TM-1998-206567; H-2280; NAS 1.15:206567; No Copyright; Avail: CASI; A03, Hardcopy

The design of the next generation of space access vehicles has led to a unique flight test that blends the space and flight research worlds. The new space vehicle designs, such as the X-33 vehicle and Reusable Launch Vehicle (RLV), are powered by linear aerospike rocket engines. Conceived of in the 1960's, these aerospike engines have yet to be flown, and many questions remain regarding aerospike engine performance and efficiency in flight. To provide some of these data before flying on the X-33 vehicle and the RLV, a spacecraft rocket engine has been flight-tested atop the NASA SR-71 aircraft as the Linear Aerospike SR-71 Experiment (LASRE). A 20 percent-scale, semispan model of the X-33 vehicle, the aerospike engine, and all the required fuel and oxidizer tanks and propellant feed systems have been mounted atop the SR-71 airplane for this experiment. A major technical objective of the LASRE flight test is to obtain installed-engine performance flight data for comparison to wind-tunnel results and for the development of computational fluid dynamics-based design methodologies. The ultimate goal of firing the aerospike rocket engine in flight is still forthcoming. An extensive design and development phase

of the experiment hardware has been completed, including approximately 40 ground tests. Five flights of the LASRE and firing the rocket engine using inert liquid nitrogen and helium in place of liquid oxygen and hydrogen have been successfully completed.
Author
SR-71 Aircraft; X-33 Reusable Launch Vehicle; Reusable Launch Vehicles; Flight Tests; Computational Fluid Dynamics; Aerospike Engines; Aerodynamic Characteristics

19980218686 NASA Marshall Space Flight Center, Huntsville, AL USA
Reusable Rocket Engine Operability Modeling and Analysis
Christenson, R. L.; Komar, D. R.; Jul. 1998; In English
Report No.(s): NASA/TP-1998-208530; M-879; NAS 1.60:208530; No Copyright; Avail: CASI; A05, Hardcopy

This paper describes the methodology, model, input data, and analysis results of a reusable launch vehicle engine operability study conducted with the goal of supporting design from an operations perspective. Paralleling performance analyses in schedule and method, this requires the use of metrics in a validated operations model useful for design, sensitivity, and trade studies. Operations analysis in this view is one of several design functions. An operations concept was developed given an engine concept and the predicted operations and maintenance processes incorporated into simulation models. Historical operations data at a level of detail suitable to model objectives were collected, analyzed, and formatted for use with the models, the simulations were run, and results collected and presented. The input data used included scheduled and unscheduled timeline and resource information collected into a Space Transportation System (STS) Space Shuttle Main Engine (SSME) historical launch operations database. Results reflect upon the importance not only of reliable hardware but upon operations and corrective maintenance process improvements.
Author
Reusable Rocket Engines; Models; Procedures; Data Bases; Design Analysis

19980218658 California Univ., Santa Barbara, CA USA
An Investigation of the Effect of Surface Impurities on the Adsorption Kinetics of Hydrogen Chemisorbed onto Iron
Shanabarger, M. R.; 1997; In English
Contract(s)/Grant(s): NCC2-63
Report No.(s): NASA/CR-1998-208255; NAS 1.26:208255; No Copyright; Avail: CASI; A03, Hardcopy

The original goal of this program was to investigate the effect surface impurities have on the heterogeneous kinetic processes of those molecular species which produce gaseous hydrogen degradation of the mechanical properties of metallic structural materials. However, shortly after the initiation of the original program, the program's NASA Technical Monitor, Dr. Howard Nelson, requested that the effort supported by this Co-operative Agreement be redirected to study more pressing materials issues associated to the development of the National Aero-Space Plane (NASP). The results of these efforts are outlined in this report. Detailed discussions of specific work, including experimental techniques and procedures, will be found in the publications listed with the subsection discussing that specific work as well and in Section 5. No inventions were generated or disclosed within this Agreement.
Author
Research; Surface Defects; Impurities; Adsorption; Hydrogen; Iron; Reaction Kinetics

19980217098 NASA Dryden Flight Research Center, Edwards, CA USA
Flight Stability and Control and Performance Results from the Linear Aerospike SR-71 Experiment (LASRE)
Moes, Timothy R.; Cobleigh, Brent R.; Cox, Timothy H.; Conners, Timothy R.; Iliff, Kenneth W.; Powers, Bruce G.; Aug. 1998; In English; Atmosphere Flight Mechanics, 10-12 Aug. 1998, Boston, MA, USA
Contract(s)/Grant(s): RTOP 244-33-02
Report No.(s): NASA/TM-1998-206565; H-2276; NAS 1.15:206565; No Copyright; Avail: CASI; A03, Hardcopy

The Linear Aerospike SR-71 Experiment (LASRE) is presently being conducted to test a 20-percent-scale version of the Linear Aerospike rocket engine. This rocket engine has been chosen to power the X-33 Single Stage to Orbit Technology Demonstrator Vehicle. The rocket engine was integrated into a lifting body configuration and mounted to the upper surface of an SR-71 aircraft. This paper presents stability and control results and performance results from the envelope expansion flight tests of the LASRE configuration up to Mach 1.8 and compares the results with wind tunnel predictions. Longitudinal stability and elevator control effectiveness were well-predicted from wind tunnel tests. Zero-lift pitching moment was mispredicted transonically. Directional stability, dihedral stability, and rudder effectiveness were overpredicted. The SR-71

handling qualities were never significantly impacted as a result of the missed predictions. Performance results confirmed the large amount of wind-tunnel-predicted transonic drag for the LASRE configuration. This drag increase made the performance of the vehicle so poor that acceleration through transonic Mach numbers could not be achieved on a hot day without depleting the available fuel.
Author
Flight Tests; Controllability; Aerospike Engines; Aerodynamic Stability; SR-71 Aircraft; Wind Tunnel Tests; Aircraft Performance

19980215576 NASA Langley Research Center, Hampton, VA USA
Thrust Enhancement in Hypervelocity Nozzles by Chemical Catalysis
Singh, D. J.; Carpenter, Mark H.; Drummond, J. P.; Journal of Propulsion and Power; Aug. 1997; Volume 13, No. 4; In English
Report No.(s): NASA/TM-1997-208084; NAS 1.15:208084; No Copyright; Avail: CASI; A01, Hardcopy

In the hypersonic flight regime, the air-breathing supersonic combustion ramjet (scramjet) has been shown to be a viable propulsion system. The current designs of scramjet engines provide performance benefits only up to a Mach number of 14. Performance losses increase rapidly as the Mach number increases. To extend the applicability of scram'jets beyond Mach 14, research is being conducted in the area of inlet and wave drag reduction, skin-friction and heat-transfer reduction, nozzle loss minimization, low-loss mixing, and combustion enhancement. For high Mach number applications, hydrogen is the obvious fuel choice because of its high energy content per unit mass in comparison with conventional fuels. These flight conditions require engines to operate at supersonic internal velocities, high combustor temperatures, and low static pressures. The high static temperature condition enhances the production of radicals such as H and OH, and the low-pressure condition slows the reaction rates, particularly the recombination reactions. High-temperature and low-pressure constraints, in combination with a small residence time, result in a radical-rich exhaust gas mixture exiting the combustor. At high Mach number conditions (due to low residence time), H and OH do not have enough time to recombine ; thus, a significant amount of energy is lost as these high-energy free radical are exhausted. The objective of the present study is to conduct a flowfield analysis for a typical nozzle geometry for NASP-type vehicle to assess for thrust enhancement in hypervelocity nozzles by substituting small amount of phosphine for hydrogen.
Derived from text
Hypervelocity Flow; Nozzle Geometry; Catalysis; Flow Distribution; Hypersonic Flight; Intake Systems; Thrust

19980210501 NASA Dryden Flight Research Center, Edwards, CA USA
Linear Aerospike SR-71 Experiment (LASRE): Aerospace Propulsion Hazard Mitigation Systems
Mizukami, Masashi; Corpening, Griffin P.; Ray, Ronald J.; Hass, Neal; Ennix, Kimberly A.; Lazaroff, Scott M.; Jul. 1998; In English; 34th Joint Propulsion, 13-15 Jul. 1998, Cleveland, OH, USA
Contract(s)/Grant(s): RTOP 242-33-02-00-33
Report No.(s): NASA/TM-1998-206561; H-2268; NAS 1.15:206561; No Copyright; Avail: CASI; A01, Hardcopy

A major hazard posed by the propulsion system of hypersonic and space vehicles is the possibility of fire or explosion in the vehicle environment. The hazard is mitigated by minimizing or detecting, in the vehicle environment, the three ingredients essential to producing fire: fuel, oxidizer, and an ignition source. The Linear Aerospike SR-71 Experiment (LASRE) consisted of a linear aerospike rocket engine integrated into one-half of an X-33-like lifting body shape, carried on top of an SR-71 aircraft. Gaseous hydrogen and liquid oxygen were used as propellants. Although LASRE is a one-of-a-kind experimental system, it must be rated for piloted flight, so this test presented a unique challenge. To help meet safety requirements, the following propulsion hazard mitigation systems were incorporated into the experiment: pod inert purge, oxygen sensors, a hydrogen leak detection algorithm, hydrogen sensors, fire detection and pod temperature thermocouples, water misting, and control room displays. These systems are described, and their development discussed. Analyses, ground test, and flight test results are presented, as are findings and lessons learned.
Author
Aerospike Engines; SR-71 Aircraft; Spacecraft Propulsion; Propulsion System Performance; Propulsion System Configurations; Liquid Oxygen; Lifting Bodies; Hypersonic Vehicles; Flight Tests

19980206153 Alabama Univ., Tuscaloosa, AL USA, Alabama Univ., Huntsville, AL USA
Research Reports: 1996 NASA/ASEE Summer Faculty Fellowship Program
Freeman, M., Editor; Chappell, C. R., Editor; Six, F., Editor; Karr, G. R., Editor; Oct. 1996; In English; 32nd, 28 May - 2 Aug. 1996, Huntsville, AL, USA
Contract(s)/Grant(s): NGT8-52819
Report No.(s): NASA-CR-205205; NAS 1.26:205205; No Copyright; Avail: CASI; A18, Hardcopy

For the 32nd consecutive year, a NASA/ASEE Summer Faculty Fellowship Program was conducted at the Marshall Space Flight Center (MSFC). The program was conducted by the University of Alabama and MSFC during the period May 28, 1996 through August 2, 1996. Operated under the auspices of the American Society for Engineering Education, the MSFC program, as well as those at other NASA centers, was sponsored by the Higher Education Branch, Education Division, NASA Headquarters, Washington, D.C. The basic objectives of the programs, which are in the 33rd year of operation nationally, are (1) to further the professional knowledge of qualified engineering and science faculty members; (2) to stimulate an exchange of ideas between participants and NASA; (3) to enrich and refresh the research and teaching activities of the participants' institutions; and (4) to contribute to the research objectives of the NASA centers. The Faculty Fellows spent 10 weeks at MSFC engaged in a research project compatible with their interests and background and worked in collaboration with a NASA/MSFC colleague. This document is a compilation of Fellows' reports on their research during the summer of 1996. The University of Alabama presents the Co-Directors' report on the administrative operations of the program. Further information can be obtained by contacting any of the editors.
Author
NASA Programs; Education; University Program; Astrionics; Materials Science; Structural Engineering; Systems Integration; Systems Analysis; Propulsion; Technology Transfer; Aerospace Sciences; Neural Nets; Systems Health Monitoring; Welding; Space Processing

19980202962 Georgia Inst. of Tech., Atlanta, GA USA
Highly Reusable Space Transportation System Concept Evaluation (The Argus Launch Vehicle)
Olds, John R.; Bellini, Peter X.; Apr. 30, 1998; In English
Contract(s)/Grant(s): NCC8-108
Report No.(s): NASA/CR-1998-208093; NAS 1.26:208093; No Copyright; Avail: CASI; A03, Hardcopy

This paper summarizes the results of a conceptual design study that was performed in support of NASA's recent Highly Reusable Space Transportation study. The Argus concept uses a Maglifter magnetic-levitation sled launch assist system to accelerate it to a takeoff ground speed of 800 fps on its way to delivering a payload of 20,000 lb. to low earth orbit. Main propulsion is provided by two supercharged ejector rocket engines. The vehicle is autonomous and is fully reusable. A conceptual design exercise determined the vehicle gross weight to be approximately 597,250 lb. and the dry weight to be 75,500 lb. Aggressive weight and operations cost assumptions were used throughout the design process consistent with a second-generation reusable system that might be deployed in 10-15 years. Drawings, geometry, and weight of the concept are included. Preliminary development, production, and operations costs along with a business scenario assuming a price-elastic payload market are also included. A fleet of three Argus launch vehicles flying a total of 149 flights per year is shown to have a financial internal rate of return of 28%. At $169/lb., the recurring cost of Argus is shown to meet the study goal of $100/lb.-$200/lb., but optimum market price results in only a factor of two to five reduction compared to today's launch systems.
Author
Space Transportation; Argus Project; Design Analysis; Reusable Launch Vehicles; Spacecraft Design

19980201356 Pennsylvania State Univ., University Park, PA USA
An Experimental Study of Characteristic Combustion-Driven Flow for CFD Validation
Santoro, Robert J.; Oct. 07, 1997; In English
Contract(s)/Grant(s): NAS8-38862
Report No.(s): NASA/CR-97-208258; NAS 1.26:208258; NAS8-38862-2; No Copyright; Avail: CASI; A07, Hardcopy

A series of uni-element rocket injector studies were completed to provide benchmark quality data needed to validate computational fluid dynamic models. A shear coaxial injector geometry was selected as the primary injector for study using gaseous hydrogen/oxygen and gaseous hydrogen/liquid oxygen propellants. Emphasis was placed on the use of non-intrusive diagnostic techniques to characterize the flowfields inside an optically-accessible rocket chamber. Measurements of the velocity and species fields were obtained using laser velocimetry and Raman spectroscopy, respectively Qualitative flame shape information was also obtained using laser-induced fluorescence excited from OH radicals and laser light scattering studies of aluminum oxide particle seeded combusting flows. The gaseous hydrogen/liquid oxygen propellant studies for the shear coaxial injector focused on breakup mechanisms associated with the liquid oxygen jet under sub-critical pressure conditions. Laser sheet illumination techniques were used to visualize the core region of the jet and a Phase Doppler Particle Analyzer was utilized for drop velocity, size and size distribution characterization. The results of these studies indicated that the shear coaxial geometry configuration was a relatively poor injector in terms of mixing. The oxygen core was observed to extend well downstream of the injector and a significant fraction of the mixing occurred in the near nozzle region where

measurements were not possible to obtain Detailed velocity and species measurements were obtained to allow CFD model validation and this set of benchmark data represents the most comprehensive data set available to date As an extension of the investigation, a series of gas/gas injector studies were conducted in support of the X-33 Reusable Launch Vehicle program. A Gas/Gas Injector Technology team was formed consisting of the Marshall Space Flight Center, the NASA Lewis Research Center, Rocketdyne and Penn State. Injector geometries studied under this task included shear and swirl coaxial configurations as well as an impinging jet injector.
Author
Computational Fluid Dynamics; Gas-Gas Interactions; Dynamic Models; Flow Distribution; Injectors; Laser Doppler Velocimeters; Laser Induced Fluorescence; Nonintrusive Measurement; Liquid Oxygen; Raman Spectroscopy; X-33 Reusable Launch Vehicle; Thrust Chambers

19980200836 Rohr Corp., Chula Vista, CA USA
Thermal Management Design for the X-33 Lifting Body
Bouslog, S.; Mammano, J.; Strauss, B.; 1998; In English; 3rd European Workshop on Thermal Protection Systems, 25-27 Mar. 1998, Netherlands
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA/CR-1998-208247; NAS 1.26:208247; No Copyright; Avail: CASI; A03, Hardcopy

The X-33 Advantage Technology Demonstrator offers a rare and exciting opportunity in Thermal Protection System development. The experimental program incorporates the latest design innovation in re-useable, low life cycle cost, and highly dependable Thermal Protection materials and constructions into both ground based and flight test vehicle validations. The unique attributes of the X-33 demonstrator for design application validation for the full scale Reusable Launch Vehicle, (RLV), are represented by both the configuration of the stand-off aeroshell, and the extreme exposures of sub-orbital hypersonic re-entry simulation. There are several challenges of producing a sub-orbital prototype demonstrator of Single Stage to Orbit/Reusable Launch Vehicle (SSTO/RLV) operations. An aggressive schedule with budgetary constraints precludes the opportunity for an extensive verification and qualification program of vehicle flight hardware. However, taking advantage of off the shelf components with proven technologies reduces some of the requirements for additional testing. The effects of scale on thermal heating rates must also be taken into account during trajectory design and analysis. Described in this document are the unique Thermal Protection System (TPS) design opportunities that are available with the lifting body configuration of the X-33. The two principal objectives for the TPS are to shield the primary airframe structure from excessive thermal loads and to provide an aerodynamic mold line surface. With the relatively benign aeroheating capability of the lifting body, an integrated stand-off aeroshell design with minimal weight and reduced procurement and operational costs is allowed. This paper summarizes the design objectives of the X-33 TPS, the flight test requirements driven configuration, and design benefits. Comparisons are made of the X-33 flight profiles and Space Shuttle Orbiter, and lifting body Reusable Launch Vehicle aerothermal environments. The X-33 TPS is based on a design to cost configuration concept. Only RLV critical technologies are verified to conform to cost and schedule restrictions. The one-off prototype vehicle configuration has evolved to minimize the tooling costs by reducing the number of unique components. Low cost approaches such as a composite/blanket leeward aeroshell and the use of Shuttle technology are implemented where applicable. The success of the X-33 will overcome the ballistic re-entry TPS mindset. The X-33 TPS is tailored to an aircraft type mission while maintaining sufficient operational margins. The flight test program for the X-33 will demonstrate that TPS for the RLV is not simply a surface insulation but rather an integrated aeroshell system.
Author
Thermal Protection; Temperature Control; X-33 Reusable Launch Vehicle; Design to Cost; Lifting Reentry Vehicles; Structural Design; Structural Design Criteria; Airframes; Aircraft Construction Materials

19980174935 NASA Lewis Research Center, Cleveland, OH USA
Preliminary Assessment of Using Gelled and Hybrid Propellant Propulsion for VTOL/SSTO Launch Systems
Palaszewski, Bryan; OLeary, Robert; Pelaccio, Dennis G.; Feb. 1998; In English; 33rd Joint Propulsion Conference, 7-9 Jul. 1997, Seattle, WA, USA
Contract(s)/Grant(s): RTOP 260-98-09
Report No.(s): NASA/TM-1998-206306; NAS 1.15:206306; AIAA Paper 97-3216; E-11004; No Copyright; Avail: CASI; A03, Hardcopy

A novel, reusable, Vertical-Takeoff-and-Vertical-Takeoff-and-Landing, Single-Stage-to-Orbit (VTOL/SSTO) launch system concept, named AUGMENT-SSTO, is presented in this paper to help quantify the advantages of employing gelled and hybrid propellant propulsion system options for such applications. The launch vehicle system concept considered uses a highly

coupled, main high performance liquid oxygen/liquid hydrogen (LO2/LH2) propulsion system, that is used only for launch, while a gelled or hybrid propellant propulsion system auxiliary propulsion system is used during final orbit insertion, major orbit maneuvering, and landing propulsive burn phases of flight. Using a gelled or hybrid propellant propulsion system for major orbit maneuver burns and landing has many advantages over conventional VTOL/SSTO concepts that use LO2/LH2 propulsion system(s) burns for all phases of flight. The applicability of three gelled propellant systems, O2/H2/Al, O2/RP-1/Al, and NTO/MMH/Al, and a state-of-the-art (SOA) hybrid propulsion system are examined in this study. Additionally, this paper addresses the applicability of a high performance gelled O2/H2 propulsion system to perform the primary, as well as the auxiliary propulsion system functions of the vehicle.
Author
Gelled Rocket Propellants; Cryogenic Rocket Propellants; Hypergolic Rocket Propellants; Vertical Takeoff Aircraft; Single Stage to Orbit Vehicles; Propulsion; Liquid Oxygen; Liquid Hydrogen; Hybrid Propulsion

19980174934 NASA Marshall Space Flight Center, Huntsville, AL USA
The Control System for the X-33 Linear Aerospike Engine
Jackson, Jerry E.; Espenschied, Erich; Klop, Jeffrey; 1998; In English
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA/CR-1998-207923; NAS 1.26:207923; No Copyright; Avail: CASI; A03, Hardcopy

The linear aerospike engine is being developed for single-stage -to-orbit (SSTO) applications. The primary advantages of a linear aerospike engine over a conventional bell nozzle engine include altitude compensation, which provides enhanced performance, and lower vehicle weight resulting from the integration of the engine into the vehicle structure. A feature of this integration is the ability to provide thrust vector control (TVC) by differential throttling of the engine combustion elements, rather than the more conventional approach of gimballing the entire engine. An analysis of the X-33 flight trajectories has shown that it is necessary to provide +/- 15% roll, pitch and yaw TVC authority with an optional capability of +/- 30% pitch at select times during the mission. The TVC performance requirements for X-33 engine became a major driver in the design of the engine control system. The thrust level of the X-33 engine as well as the amount of TVC are managed by a control system which consists of electronic, instrumentation, propellant valves, electro-mechanical actuators, spark igniters, and harnesses. The engine control system is responsible for the thrust control, mixture ratio control, thrust vector control, engine health monitoring, and communication to the vehicle during all operational modes of the engine (checkout, pre-start, start, main-stage, shutdown and post shutdown). The methodology for thrust vector control, the health monitoring approach which includes failure detection, isolation, and response, and the basic control system design are the topic of this paper. As an additional point of interest a brief description of the X-33 engine system will be included in this paper.
Author
X-33 Reusable Launch Vehicle; Control Systems Design; Aerospike Engines; Engine Control; Single Stage to Orbit Vehicles; Throttling; Thrust Control; Thrust Vector Control

19980174930 Lockheed Martin Corp., Palmdale, CA USA, Allied-Signal Aerospace Co., Torrance, CA USA, Rockwell International Corp., Canoga Park, CA USA, Goodrich (B. F.) Aerospace, Chula Vista, CA USA, Sverdrup Technology, Inc., Saint Louis, MO USA
X-33, Phase 2
1998; In English; Original contains color illustrations
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA/CR-1998-208192; NAS 1.26:208192; No Copyright; Avail: CASI; A07, Hardcopy

In response to the Cooperative Agreement, Lockheed Martin Skunk Works has compiled an Annual Performance Report of the X-33/RLV Program. This report consists of individual reports from all industry team members, as well as NASA team centers. The first milestone was hand delivered to NASA MSFC. The second year has been one of significant accomplishment in which team members have demonstrated their ability to meet vital benchmarks while continuing on the technical adventure of the 20th century.
Author (revised)
X-33 Reusable Launch Vehicle; Reusable Spacecraft

19980169231 NASA Dryden Flight Research Center, Edwards, CA USA
Wingless Flight: The Lifting Body Story
Reed, R. Dale; Lister, Darlene, Editor; Huntley, J. D., Editor; 1997; In English
Report No.(s): NASA/SP-4220; NAS 1.21:4220; No Copyright; Avail: CASI; A12, Hardcopy

Wingless Flight tells the story of the most unusual flying machines ever flown, the lifting bodies. It is my story about my friends and colleagues who committed a significant part of their lives in the 1960s and 1970s to prove that the concept was a viable one for use in spacecraft of the future. This story, filled with drama and adventure, is about the twelve-year period from 1963 to 1975 in which eight different lifting-body configurations flew. It is appropriate for me to write the story, since I was the engineer who first presented the idea of flight-testing the concept to others at the NASA Flight Research Center. Over those twelve years, I experienced the story as it unfolded day by day at that remote NASA facility northeast of los Angeles in the bleak Mojave Desert. Benefits from this effort immediately influenced the design and operational concepts of the winged NASA Shuttle Orbiter. However, the full benefits would not be realized until the 1990s when new spacecraft such as the X-33 and X-38 would fully employ the lifting-body concept. A lifting body is basically a wingless vehicle that flies due to the lift generated by the shape of its fuselage. Although both a lifting reentry vehicle and a ballistic capsule had been considered as options during the early stages of NASA's space program, NASA initially opted to go with the capsule. A number of individuals were not content to close the book on the lifting-body concept. Researchers including Alfred Eggers at the NASA Ames Research Center conducted early wind-tunnel experiments, finding that half of a rounded nose-cone shape that was flat on top and rounded on the bottom could generate a lift-to-drag ratio of about 1.5 to 1. Eggers' preliminary design sketch later resembled the basic M2 lifting-body design. At the NASA Langley Research Center, other researchers toyed with their own lifting-body shapes. Meanwhile, some of us aircraft-oriented researchers at the, NASA Flight Research Center at Edwards Air Force Base (AFB) in California were experiencing our own fascination with the lifting-body concept. A model-aircraft builder and private pilot on my own time, I found the lifting-body idea intriguing. I built a model based on Eggers' design, tested it repeatedly, made modifications in its control and balance characteristics along the way, then eventually presented the concept to others at the Center, using a film of its flights that my wife, Donna and I had made with our 8-mm home camera.
Author
Aircraft Models; Flight Tests; Shapes; Nose Cones; Lifting Bodies; Wings

19980151078 NASA Lewis Research Center, Cleveland, OH USA
Neural Network and Regression Approximations in High Speed Civil Transport Aircraft Design Optimization
Patniak, Surya N.; Guptill, James D.; Hopkins, Dale A.; Lavelle, Thomas M.; Apr. 1998; In English
Contract(s)/Grant(s): RTOP 523-22-13
Report No.(s): NASA/TM-1998-206316; NAS 1.15:206316; E-10872; No Copyright; Avail: CASI; A03, Hardcopy

Nonlinear mathematical-programming-based design optimization can be an elegant method. However, the calculations required to generate the merit function, constraints, and their gradients, which are frequently required, can make the process computational intensive. The computational burden can be greatly reduced by using approximating analyzers derived from an original analyzer utilizing neural networks and linear regression methods. The experience gained from using both of these approximation methods in the design optimization of a high speed civil transport aircraft is the subject of this paper. The Langley Research Center's Flight Optimization System was selected for the aircraft analysis. This software was exercised to generate a set of training data with which a neural network and a regression method were trained, thereby producing the two approximating analyzers. The derived analyzers were coupled to the Lewis Research Center's CometBoards test bed to provide the optimization capability. With the combined software, both approximation methods were examined for use in aircraft design optimization, and both performed satisfactorily. The CPU time for solution of the problem, which had been measured in hours, was reduced to minutes with the neural network approximation and to seconds with the regression method. Instability encountered in the aircraft analysis software at certain design points was also eliminated. On the other hand, there were costs and difficulties associated with training the approximating analyzers. The CPU time required to generate the input-output pairs and to train the approximating analyzers was seven times that required for solution of the problem.
Author
Nonlinear Programming; Neural Nets; Aircraft Design; National Aerospace Plane Program; Civil Aviation; Supersonic Transports

19980151073 Alabama Univ., Huntsville, AL USA
Affordable In-Space Transportation, Phase 2, An Advanced Concepts Project
Nov. 26, 1996; In English; Technical Interchange Meeting, 16-17 Oct. 1996, Huntsville, AL, USA
Contract(s)/Grant(s): NAS8-38609
Report No.(s): NASA/CR-97-205203; NAS 1.26:205203; UAH-5-25620; No Copyright; Avail: CASI; A04, Hardcopy

The Affordable In-Space Transportation (AIST) program was established by the NASA Office of Space Access to improve transportation and lower the costs from Low Earth Orbit (LEO) to Geostationary Earth Orbit (GEO) and beyond (to Lunar orbit, Mars orbit, inner solar system missions, and return to LEO). A goal was established to identify and develop radically

innovative concepts for new upper stages for Reusable Launch Vehicles (RLV) and Highly Reusable Space Transportation (HRST) systems. New architectures and technologies are being identified which have the potential to meet a cost goal of $1,000 to $2,000 per pound for transportation to GEO and beyond for overall mission cost (including the cost to LEO). A Technical Interchange Meeting (ITM) was held on October 16 and 17, 1996 in Huntsville, Alabama to review previous studies, present advanced concepts and review technologies that could be used to meet the stated goals. The TIM was managed by NASA-Mar-shaU Space Flight Center (MSFC) Advanced Concepts Office with Mr. Alan Adams providing TIM coordination. Mr. John C. Manidns of NASA Headquarters provided overall sponsorship. The University of Alabama in Huntsville (UAH) Propulsion Research Center hosted the TM at the UAH Research Center. Dr. Clark Hawk, Center Director, was the principal investigator. Technical support was provided by Christensen Associates. Approximately 70 attendees were present at the meeting. This Executive Summary provides a record of the key discussions and results of the TIM in a summary format. It incorporates the response to the following basic issues of the TPA, which addressed the following questions: 1. What are the cost drivers and how can they be reduced? 2. What are the operational issues and their impact on cost? What is the current Technology Readiness Level (TRL) and what will it take to reach TRL 6? 4. What are the key enabling technologies and sequence for their accomplishment? 5. What is the proposed implementation time frame? See Appendix A for the TIM Agenda and Appendix C for the AIST Program Terms of Reference.
Author
Space Transportation System; Cost Reduction; Coordination; Reusable Launch Vehicles

19980107885 NASA Langley Research Center, Hampton, VA USA
Thermal Structures Technology Development for Reusable Launch Vehicle Cryogenic Propellant Tanks
Johnson, Theodore F.; Natividad, Roderick; Rivers, H. Kevin; Smith, Russell; 1998; In English; 3rd Space Technology and Application International Forum: Conference on Next Generation Launch Systems, 25-29 Jan. 1998, Albuquerque, NM, USA
Report No.(s): NASA/TM-1998-207375; NAS 1.15:207375; No Copyright; Avail: CASI; A03, Hardcopy

Analytical and experimental studies conducted at the NASA Langley Research Center for investigating integrated cryogenic propellant tank systems for a Reusable Launch Vehicle are described. The cryogenic tanks are investigated as an integrated tank system. An integrated tank system includes the tank wall, cryogenic insulation, Thermal Protection System (TPS) attachment sub-structure, and TPS. Analysis codes are used to size the thicknesses of cryogenic insulation and TPS insulation for thermal loads, and to predict tank buckling strengths at various ring frame spacings. The unique test facilities developed for the testing of cryogenic tank components are described. Testing at cryogenic and high-temperatures verifies the integrity of materials, design concepts, manufacturing processes, and thermal/structural analyses. Test specimens ranging from the element level to the subcomponent level are subjected to projected vehicle operational mechanical loads and temperatures. The analytical and experimental studies described in this paper provide a portion of the basic information required for the development of light-weight reusable cryogenic propellant tanks.
Author
Reusable Launch Vehicles; Cryogenic Fluid Storage; Structural Engineering; Structural Analysis; Thermal Protection; Propellant Tanks

19980073196 Lockheed Sanders, Inc., Nashua, NH USA
X-33/RLV System Health Management/ Vehicle Health Management
Garbos, Raymond J.; Mouyos, William; 1998; In English; Structures, Structural Dynnamics, and Materials Conference and Adaptive Structures Forum, 20-23 Apr. 1998, Long Beach, CA, USA
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA/CR-1998-208136; NAS 1.26:208136; AIAA Paper 98-1928; No Copyright; Avail: CASI; A02, Hardcopy

To reduce operations cost, the RLV must include the following elements: highly reliable, robust subsystems designed for simple repair access with a simplified servicing infrastructure and incorporating expedited decision making about faults and anomalies. A key component for the Single Stage to Orbit (SSTO) RLV System used to meet these objectives is System Health Management (SHM). SHM deals with the vehicle component- Vehicle Health Management (VHM), the ground processing associated with the fleet (GVHM) and the Ground Infrastructure Health Management (GIHM). The objective is to provide an automated collection and paperless health decision, maintenance and logistics system. Many critical technologies are necessary to make the SHM (and more specifically VHM) practical, reliable and cost effective. Sanders is leading the design, development and integration of the SHM system for RLV and X-33 SHM (a sub-scale, sub-orbit Advanced Technology

Demonstrator). This paper will present the X-33 SHM design which forms the baseline for RLV SHM. This paper will also discuss other applications of these technologies.
Author
X-33 Reusable Launch Vehicle; Decision Making; Smart Structures; Cost Reduction; Management Systems

19980069722 Lockheed Sanders, Inc., Nashua, NH USA
Reusable Cryogenic Tank VHM Using Fiber Optic Distributed Sensing Technology
Bodan-Sanders, Patricia; Bouvier, Carl; 1998; In English; Structures, Structural Dynamics and Materials Conference and Adaptive Structures Forum, 20-23 Apr. 1998, Long Beach, CA, USA
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA/CR-1998-208132; NAS 1.26:208132; AIAA Paper 98-1929; No Copyright; Avail: CASI; A02, Hardcopy

The reusable oxygen and hydrogen tanks are key systems for both the X-33 (sub-scale, sub-orbital technology demonstrator) and the commercial Reusable Launch Vehicle (RLV). The backbone of the X-33 Reusable Cryogenic Tank Vehicle Health Management (VHM) system lies in the optical network of distributed strain temperature and hydrogen sensors. This network of fiber sensors will create a global strain and temperature map for monitoring the health of the tank structure, cryogenic insulation, and Thermal Protection System. Lockheed Martin (Sanders and LMMSS) and NASA Langley have developed this sensor technology for the X-33 and have addressed several technical issues such as fiber bonding and laser performance in this harsh environment.
Author
X-33 Reusable Launch Vehicle; Cryogenic Fluid Storage; Fuel Tanks; Temperature Sensors; Fluid Management; Thermal Insulation; Fiber Optics; Cryogenic Rocket Propellants

19980055201 Boeing Commercial Airplane Co., Seattle, WA USA
Aircraft Emission Scenarios Projected in Year 2015 for the NASA Technology Concept Aircraft (TCA) High Speed Civil Transport
Baughcum, Steven L.; Henderson, Stephen C.; Mar. 1998; In English
Contract(s)/Grant(s): NAS1-20220; RTOP 537-09-23-02
Report No.(s): NASA/CR-1998-207635; NAS 1.26:207635; No Copyright; Avail: CASI; A03, Hardcopy

This report describes the development of a three-dimensional database of aircraft fuel burn and emissions (fuel burned, NOx, CO, and hydrocarbons) from projected fleets of high speed civil transports (HSCTs) on a universal airline network. Inventories for 500 and 1000 HSCT fleets, as well as the concurrent subsonic fleets, were calculated. The HSCT scenarios are calculated using the NASA technology concept airplane (TCA) and update an earlier report. These emissions inventories are available for use by atmospheric scientists conducting the Atmospheric Effects of Stratospheric Aircraft (AESA) modeling studies. Fuel burned and emissions of nitrogen oxides (NOx as NO2), carbon monoxide, and hydrocarbons have been calculated on a 1 degree latitude x 1 degree longitude x 1 kilometer pressure altitude grid and delivered to NASA as electronic files.
Author
Aircraft Fuels; Emission; Supersonic Transports; Atmospheric Effects; Airline Operations; Ozonosphere; X-30 Vehicle

19980055123 NASA Dryden Flight Research Center, Edwards, CA USA
Development of the X-33 Aerodynamic Uncertainty Model
Cobleigh, Brent R.; Apr. 1998; In English
Contract(s)/Grant(s): RTOP 242-33-02-00-23
Report No.(s): NASA/TP-1998-206544; NAS 1.60:206544; H-2221; No Copyright; Avail: CASI; A03, Hardcopy

An aerodynamic uncertainty model for the X-33 single-stage-to-orbit demonstrator aircraft has been developed at NASA Dryden Flight Research Center. The model is based on comparisons of historical flight test estimates to preflight wind-tunnel and analysis code predictions of vehicle aerodynamics documented during six lifting-body aircraft and the Space Shuttle Orbiter flight programs. The lifting-body and Orbiter data were used to define an appropriate uncertainty magnitude in the subsonic and supersonic flight regions, and the Orbiter data were used to extend the database to hypersonic Mach numbers. The uncertainty data consist of increments or percentage variations in the important aerodynamic coefficients and derivatives as a function of Mach number along a nominal trajectory. The uncertainty models will be used to perform linear analysis of the X-33 flight control system and Monte Carlo mission simulation studies. Because the X-33 aerodynamic uncertainty model

was developed exclusively using historical data rather than X-33 specific characteristics, the model may be useful for other lifting-body studies.
Author
Aerodynamic Characteristics; X-33 Reusable Launch Vehicle; Monte Carlo Method; Lifting Bodies; Hypersonics; Flight Control; Space Shuttle Orbiters

19980053187 NASA Marshall Space Flight Center, Huntsville, AL USA
Analysis of Aerospike Plume Induced Base-Heating Environment
Wang, Ten-See; 1998; In English; 7th Joint Thermophysics and Heat Transfer Conference, 15-18 Jun. 1998, Albuquerque, NM, USA
Report No.(s): NASA/TM-1998-208139; NAS 1.15:208139; AIAA Paper 98-2469; No Copyright; Avail: CASI; A02, Hardcopy

Computational analysis is conducted to study the effect of an aerospike engine plume on X-33 base-heating environment during ascent flight. To properly account for the effect of forebody and aftbody flowfield such as shocks and to allow for potential plume-induced flow-separation, thermo-flowfield of trajectory points is computed. The computational methodology is based on a three-dimensional finite-difference, viscous flow, chemically reacting, pressure-base computational fluid dynamics formulation, and a three-dimensional, finite-volume, spectral-line based weighted-sum-of-gray-gases radiation absorption model computational heat transfer formulation. The predicted convective and radiative base-heat fluxes are presented.
Author
Aerospike Engines; Plumes; Base Heating; Boundary Layer Separation; Convective Heat Transfer; Gray Gas; Heat Transfer; Separated Flow; Three Dimensional Flow; X-33 Reusable Launch Vehicle

19980048417 NASA Marshall Space Flight Center, Huntsville, AL USA
Tether Transportation System Study
Bangham, M. E.; Lorenzini, E.; Vestal, L.; Mar. 1998; In English
Contract(s)/Grant(s): NAS8-39400
Report No.(s): NASA/TP-1998-206959; NAS 1.60:206959; M-853; No Copyright; Avail: CASI; A05, Hardcopy

The projected traffic to geostationary earth orbit (GEO) is expected to increase over the next few decades. At the same time, the cost of delivering payloads from the Earth's surface to low earth orbit (LEO) is projected to decrease, thanks in part to the Reusable Launch Vehicle (RLV). A comparable reduction in the cost of delivering payloads from LEO to GEO is sought. The use of in-space tethers, eliminating the requirement for traditional chemical upper stages and thereby reducing the launch mass, has been identified as such an alternative. Spinning tethers are excellent kinetic energy storage devices for providing the large delta vee's required for LEO to GEO transfer. A single-stage system for transferring payloads from LEO to GEO was proposed some years ago. The study results presented here contain the first detailed analyses of this proposal, its extension to a two-stage system, and the likely implementation of the operational system.
Author
Low Earth Orbits; Geosynchronous Orbits; Tethering; Reusable Launch Vehicles; Payloads; Costs

19980047114 NASA Langley Research Center, Hampton, VA USA
Hyper-X Wind Tunnel Program
McClinton, C. R.; Holland, S. D.; Rock, K. E.; Engelund, W. C.; Voland, R. T.; Huebner, L. D.; Roger, R. C.; 1998; In English; Aerospace Sciences Meeting and Exhibit, 12-15 Jan. 1998, Reno, NV, USA
Report No.(s): NASA/TM-1998-207317; NAS 1.15:207317; AIAA Paper 98-0553; No Copyright; Avail: CASI; A03, Hardcopy

This paper provides an overview of NASA's focused hypersonic technology program, called the Hyper-X Program. The Hyper-X Program, a joint NASA Langley and Dryden program, is designed to move hypersonic, air breathing vehicle technology from the laboratory environment to the flight environment, the last stage preceding prototype development. The Hyper-X research vehicle will provide the first ever opportunity to obtain data on an airframe integrated scramjet (supersonic combustion ramjet) propulsion system at true flight conditions and the first opportunity for flight validation of experimental wind tunnel, numerical and analytical methods used for design of these vehicles. A substantial portion of the program is experimentally based, both for database development and performance validation. The program is now concentrating on Mach 7 vehicle development, verification and validation and flight test risk reduction. This paper concentrates on the aerodynamic

and propulsion experimental programs. Wind tunnel testing of the flight engine and complete airframe integrated scramjet configuration flow-path is expected in 1998 and 1999, respectively, and flight test is planned for 2000.
Author
Supersonic Combustion Ramjet Engines; Research Vehicles; Hypersonic Vehicles; Flight Tests; Transatmospheric Vehicles

19980046639 NASA Langley Research Center, Hampton, VA USA
Computational/Experimental Aeroheating Predictions for X-33 Phase 2 Vehicle
Hamilton, H. Harris,II; Weilmuenster, K. James; Horvath, Thomas J.; Berry, Scott A.; 1998; In English; Aerospace Sciences Meeting and Exhibit, 12-15 Jan. 1998, Reno, NV, USA
Report No.(s): NASA/TM-1998-207290; NAS 1.15:207290; AIAA Paper 98-0869; No Copyright; Avail: CASI; A03, Hardcopy

Laminar and turbulent heating-rate calculations from an 'engineering' code and laminar calculations from a 'benchmark' Navier-Stokes code are compared with experimental wind-tunnel data obtained on several candidate configurations for the X-33 Phase 2 flight vehicle. The experimental data were obtained at a Mach number of 6 and a freestream Reynolds number ranging from 1 to 8 x 10(exp 6)/ft. Comparisons are presented along the windward symmetry plane and in a circumferential direction around the body at several axial stations at angles of attack from 20 to 40 deg. The experimental results include both laminar and turbulent How. For the highest angle of attack some of the measured heating data exhibited a 'non-laminar' behavior which caused the heating to increase above the laminar level long before 'classical' transition to turbulent flow was observed. This trend was not observed at the lower angles of attack. When the flow was laminar, both codes predicted the heating along the windward symmetry plane reasonably well but under-predicted the heating in the chine region. When the flow was turbulent the LATCH code accurately predicted the measured heating rates. Both codes were used to calculate heating rates over the X-33 vehicle at the peak heating point on the design trajectory and they were found to be in very good agreement over most of the vehicle windward surface.
Author
Computational Fluid Dynamics; Navier-Stokes Equation; Numerical Flow Visualization; Direct Numerical Simulation; X-33 Reusable Launch Vehicle; Reynolds Number; Transition Flow; Turbulent Flow; Hypersonic Boundary Layer; Hypersonic Vehicles

19980041407 Rockwell International Corp., Canoga Park, CA USA
Operationally efficient propulsion system study (OEPSS) data book, Volume 10, Air Augmented Rocket Afterburning
Farhangi, Shahram; Trent, Donnie, Editor; Oct. 30, 1992; In English
Contract(s)/Grant(s): NAS10-11568
Report No.(s): NASA/CR-92-207458; NAS 1.26:207458; RI/RD90-149-10-VOL-10; CDR-91-099; No Copyright; Avail: CASI; A03, Hardcopy

A study was directed towards assessing viability and effectiveness of an air augmented ejector/rocket. Successful thrust augmentation could potentially reduce a multi-stage vehicle to a single stage-to-orbit vehicle (SSTO) and, thereby, eliminate the associated ground support facility infrastructure and ground processing required by the eliminated stage. The results of this preliminary study indicate that an air augmented ejector/rocket propulsion system is viable. However, uncertainties resulting from simplified approach and assumptions must be resolved by further investigations.
Author
Thrust Augmentation; Rocket Engines; Advanced Launch System (Sts); Propulsion System Performance; Mach Number; Ejectors

19980041384 Rockwell International Corp., Canoga Park, CA USA
Operationally Efficient Propulsion System Study (OEPSS): OEPSS Video Script
Wong, George S.; Waldrop, Glen S.; Trent, Donnie, Editor; Sep. 30, 1992; In English
Contract(s)/Grant(s): NAS10-11568
Report No.(s): NASA/CR-92-207441; NAS 1.26:207441; No Copyright; Avail: CASI; A05, Hardcopy

The OEPSS video film, along with the OEPSS Databooks, provides a data base of current launch experience that will be useful for design of future expendable and reusable launch systems. The focus is on the launch processing of propulsion systems. A brief 15-minute overview of the OEPSS study results is found at the beginning of the film. The remainder of the film discusses in more detail: current ground operations at the Kennedy Space Center; typical operations issues and problems; critical operations technologies; and efficiency of booster and space propulsion systems. The impact of system architecture on

the launch site and its facility infrastucture is emphasized. Finally, a particularly valuable analytical tool, developed during the OEPSS study, that will provide for the 'first time' a quantitative measure of operations efficiency for a propulsion system is described.
Author
Propulsion System Performance; Video Tapes; Spacecraft Launching

19980037670 NASA Langley Research Center, Hampton, VA USA
Hypersonic Boundary-Layer Transition for X-33 Phase 2 Vehicle
Thompson, Richard A.; Hamilton, Harris H., II; Berry, Scott A.; Horvath, Thomas J.; Nowak, Robert J.; 1998; In English; Aerospace Sciences Meeting and Exhibit, 12-15 Jan. 1998, Reno, NV, USA; Original contains color illustrations
Report No.(s): NASA/TM-1998-207316; NAS 1.15:207316; AIAA Paper 98-0867; No Copyright; Avail: CASI; A03, Hardcopy

A status review of the experimental and computational work performed to support the X-33 program in the area of hypersonic boundary-layer transition is presented. Global transition fronts are visualized using thermographic phosphor measurements. Results are used to derive transition correlations for 'smooth body' and discrete roughness data and a computational tool is developed to predict transition onset for X-33 using these results. The X-33 thermal protection system appears to be conservatively designed for transition effects based on these studies. Additional study is needed to address concerns related to surface waviness. A discussion of future test plans is included.
Author
Boundary Layer Transition; Hypersonics; Flow Distribution; Wind Tunnel Tests; Finite Volume Method; Flow Visualization

19980036345 Alabama Univ., Huntsville, AL USA
Study Acoustic Emissions from Composites
Walker, James; Workman,Gary; Jan. 1998; In English; Original contains color illustrations
Contract(s)/Grant(s): NAS8-38609
Report No.(s): NASA/CR-1998-207483; NAS 1.26:207483; No Copyright; Avail: CASI; A06, Hardcopy

The purpose of this work will be to develop techniques for monitoring the acoustic emissions from carbon epoxy composite structures at cryogenic temperatures. Performance of transducers at temperatures ranging from ambient to cryogenic and the characteristics of acoustic emission from composite structures will be studied and documented. This entire effort is directed towards characterization of structures used in NASA propulsion programs such as the X-33.
Author
Epoxy Resins; Acoustic Emission; Cryogenic Temperature; Composite Structures

19980020931 Albany Univ., Albany, NY USA
Advanced High Brilliance X-Ray Source
Gibson, Walter M.; 1998; In English
Contract(s)/Grant(s): NAS8-39926
Report No.(s): NASA/CR-1998-207486; NAS 1.26:207486; No Copyright; Avail: CASI; A03, Hardcopy

The possibility to dramatically increase the efficiency of laboratory based protein structure measurements through the use of polycapillary X-ray optics was investigated. This project initiated April 1, 1993 and concluded December 31, 1996 (including a no cost extension from June 31, 1996). This is a final report of the project. The basis for the project is the ability to collect X-rays from divergent electron bombardment laboratory X-ray sources and redirect them into quasiparallel or convergent (focused) beams. For example, a 0.1 radian (approx. 6 deg) portion of a divergent beam collected by a polycapillary collimator and transformed into a quasiparallel beam of 3 millradian (0.2 deg) could give a gain of 6(exp 2)/0.2(exp 2) x T for the intensity of a diffracted beam from a crystal with a 0.2 deg diffraction width. T is the transmission efficiency of the polycapillary diffraction optic, and for T=0.5, the gain would be 36/0.04 x O.5=45. In practice, the effective collection angle will depend on the source spot size, the input focal length of the optic (usually limited by the source spot-to-window distance on the x-ray tube) and the size of the crystal relative to the output diameter of the optic. The transmission efficiency, T, depends on the characteristics (fractional open area, surface roughness, shape and channel diameter) of the polycapillary optic and is typically in the range 0.2-0.4. These effects could substantially reduce the expected efficiency gain. During the course of this study, the possibility to use a weakly focused beam (0.5 deg convergence) was suggested which could give an additional 10-20 X efficiency gain for small samples . Weakly focused beams from double focusing mirrors are frequently used for macromolecular crystallography studies. Furthermore the crystals are typically oscillated by as much as

2 deg during each X-ray exposure in order to increase the reciprocal space (number of crystal planes) sampled and use of a slightly convergent beam could, in principle, provide a similar sampling benefit without oscillation. Although more problematic, because of complications in analyzing the diffraction patterns, it was also suggested that even more extreme beam convergence might be used to give another order of magnitude intensity gain and even smaller focused spot size which could make it possible to study smaller protein crystals than can be studied using standard laboratory based X-ray diffraction systems. This project represents the first systematic investigation of these possibilities. As initially proposed, the contract included requirements for design, purchase, evaluation and delivery of three polycapillary lenses to the Laboratory for Structural Biology at MSFC and demonstration of such optics at MSFC for selected protein crystal diffraction applications.
Author
X Ray Sources; Efficiency; Proteins; Chemical Analysis

19980019978 San Jose State Univ., CA USA
Development of Processing Techniques for Advanced Thermal Protection Materials
Selvaduray, Guna; Cox, Michael; Srinivasan, Vijayakumar; Nov. 30, 1997; In English
Contract(s)/Grant(s): NAG2-848
Report No.(s): NASA/CR-1998-207440; NAS 1.26:207440; No Copyright; Avail: CASI; A02, Hardcopy

Thermal Protection Materials Branch (TPMB) has been involved in various research programs to improve the properties and structural integrity of the existing aerospace high temperature materials. Specimens from various research programs were brought into the analytical laboratory for the purpose of obtaining and refining the material characterization. The analytical laboratory in TPMB has many different instruments which were utilized to determine the physical and chemical characteristics of materials. Some of the instruments that were utilized by the SJSU students are: Scanning Electron Microscopy (SEM), Energy Dispersive X-ray analysis (EDX), X-ray Diffraction Spectrometer (XRD), Fourier Transform-Infrared Spectroscopy (FTIR), Ultra Violet Spectroscopy/Visible Spectroscopy (UV/VIS), Particle Size Analyzer (PSA), and Inductively Coupled Plasma Atomic Emission Spectrometer (ICP-AES). The above mentioned analytical instruments were utilized in the material characterization process of the specimens from research programs such as: aerogel ceramics (I) and (II), X-33 Blankets, ARC-Jet specimens, QUICFIX specimens and gas permeability of lightweight ceramic ablators. In addition to analytical instruments in the analytical laboratory at TPMB, there are several on-going experiments. One particular experiment allows the measurement of permeability of ceramic ablators. From these measurements, physical characteristics of the ceramic ablators can be derived.
Derived from text
Thermal Protection; Infrared Spectroscopy; Refractory Materials; Aircraft Construction Materials; Ablative Materials; Infrared Spectra; Plasma Radiation; X Ray Analysis

19980019278 NASA Langley Research Center, Hampton, VA USA
Computational/Experimental Aeroheating Predictions for X-33, Phase 2, Vehicle
Hamilton, H. Harris, II; Weilmuenster, K. James; Horvath, Thomas J.; Berry, Scott A.; 1998; In English; 36th Aerospace Sciences Meeting and Exhibit, 12-15 Jan. 1998, Reno, NV, USA
Report No.(s): NASA/TM-1998-207315; NAS 1.15:207315; AIAA Paper 98-0869; No Copyright; Avail: CASI; A03, Hardcopy

Laminar and turbulent heating-rate calculations from an 'engineering' code and laminar calculations from a 'benchmark' Navier-Stokes code are compared with experimental wind-tunnel data obtained on several candidate configurations for the X-33 Phase 2 flight vehicle. The experimental data were obtained at a Mach number of 6 and a freestream Reynolds number ranging from 1 to 8 x 10(exp 6)/ft. Comparisons are presented along the windward symmetry plane and in a circumferential direction around the body at several axial stations at angles of attack from 20 to 40 deg. The experimental results include both laminar and turbulent flow. For the highest angle of attack some of the measured heating data exhibited a 'non-laminar' behavior which caused the heating to increase above the laminar level long before 'classical' transition to turbulent flow was observed. This trend was not observed at the lower angles of attack. When the flow was laminar, both codes predicted the heating along the windward symmetry plane reasonably well but under-predicted the heating in the chine region. When the flow was turbulent the LATCH code accurately predicted the measured heating rates. Both codes were used to calculate heating rates over the X-33 vehicle at the peak heating point on the design trajectory and they were found to be in very good agreement over most of the vehicle windward surface.
Author
Aerodynamic Heating; X-33 Reusable Launch Vehicle; Wind Tunnel Tests; Angle of Attack; Turbulent Flow; Laminar Flow; Navier-Stokes Equation; Computational Fluid Dynamics

19980015220 NERAC, Inc., Tolland, CT USA, NASA, Washington, DC USA
National Aerospace Plane Thermal Development. (Latest Citations from the Aerospace Database)
Mar. 1996; In English; Page count unavailable.
Report No.(s): NASA/TM-96-206845; NAS 1.15:206845; PB96-864210; Copyright; Avail: National Technical Information Service (NTIS)

The bibliography contains citations concerning thermal properties of the National Aerospace Plane (NASP). Analysis of thermal stress, and methods for determining thermal effects on the plane's supersonic structure are discussed. The citations also review temperature extremes that the vehicle is likely to encounter. (Contains 50-250 citations and includes a subject term index and title list.)
NTIS
Bibliographies; Thermal Stresses; Stress Analysis; Aerospace Planes

19980010422 NASA, Washington, DC USA
X-33 Development History
Butrica, Andrew J.; Dec. 15, 1997; In English
Contract(s)/Grant(s): NASw-97005
Report No.(s): NASA/CR-97-206438; NAS 1.26:206438; No Copyright; Avail: CASI; A02, Hardcopy

The problem of dealing with various types of proprietary documents, whether from the Lockheed Martin, the Skunk Works, McDonnell Douglas, Rockwell, and other corporations extant or extinct, remains unresolved. The computerized archive finding aid has over 100 records at present. These records consist of X-33 photographs, press releases, media clippings, and the small number of X-33 project records collected to date.
Author
Computer Storage Devices; Computer Systems Programs; Data Processing

19980008580 NASA Dryden Flight Research Center, Edwards, CA USA
Design and Calibration of the X-33 Flush Airdata Sensing (FADS) System
Whitmore, Stephen A.; Cobleigh, Brent R.; Haering, Edward A.; Jan. 1998; In English; AIAA Aerospace Sciences Meeting and Exhibit, 12-15 Jan. 1998, Reno, NV, USA
Contract(s)/Grant(s): RTOP 242-33-02-00-23
Report No.(s): NASA/TM-98-206540; H-2219; NAS 1.15:206540; AIAA Paper 98-0201; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents the design of the X-33 Flush Airdata Sensing (FADS) system. The X-33 FADS uses a matrix of pressure orifices on the vehicle nose to estimate airdata parameters. The system is designed with dual-redundant measurement hardware, which produces two independent measurement paths. Airdata parameters that correspond to the measurement path with the minimum fit error are selected as the output values. This method enables a single sensor failure to occur with minimal degrading of the system performance. The paper shows the X-33 FADS architecture, derives the estimating algorithms, and demonstrates a mathematical analysis of the FADS system stability. Preliminary aerodynamic calibrations are also presented here. The calibration parameters, the position error coefficient (epsilon), and flow correction terms for the angle of attack (delta alpha), and angle of sideslip (delta beta) are derived from wind tunnel data. Statistical accuracy of' the calibration is evaluated by comparing the wind tunnel reference conditions to the airdata parameters estimated. This comparison is accomplished by applying the calibrated FADS algorithm to the sensed wind tunnel pressures. When the resulting accuracy estimates are compared to accuracy requirements for the X-33 airdata, the FADS system meets these requirements.
Author
X-33 Reusable Launch Vehicle; Data Systems; Systems Integration; Calibrating; Remote Sensing; Systems Engineering

19980003836 Alabama Univ., Huntsville, AL USA
Affordable In-Space Transportation Phase 2: An Advanced Concepts Project
1996; In English; Technical Interchange Meeting, 16-17 Oct. 1996, USA
Contract(s)/Grant(s): NAS8-38609
Report No.(s): NASA/CR-97-206469; NAS 1.26:206469; No Copyright; Avail: CASI; A04, Hardcopy

The Affordable In-Space Transportation (AIST) program was established by the NASA Office of Space Access to improve transportation and lower the costs from Low Earth Orbit (LEO) to Geostationary Earth Orbit (GEO) and beyond (to Lunar orbit, Mars orbit, inner solar system missions, and return to LEO). A goal was established to identify and develop radically

innovative concepts for new upper stages for Reusable Launch Vehicles (RLV) and Highly Reusable Space Transportation (HRST) systems. New architectures and technologies are being identified which have the potential to meet a cost goal of $1,000 to $2,000 per pound for transportation to GEO and beyond for overall mission cost (including the cost to LEO). A Technical Interchange Meeting (TTM) was held on October 16 and 17, 1996 in Huntsville, Alabama to review previous studies, present advanced concepts and review technologies that could be used to meet the stated goals. The TIN4 was managed by NASA-Marshall Space Flight Center (MSFC) Advanced Concepts Office with Mr. Alan Adams providing TIM coordination. Mr. John C. Mankins of NASA Headquarters provided overall sponsorship. The University of Alabama in Huntsville (UAH) Propulsion Research Center hosted the TIM at the UAH Research Center. Dr. Clark Hawk, Center Director, was the principal investigator. Technical support was provided by Christensen Associates. Approximately 70 attendees were present at the meeting. This Executive Summary provides a record of the key discussions and results of the TIN4 in a summary for-mat. It incorporates the response to the following basic issues of the TDVL which addressed the following questions: 1. What are the cost drivers and how can they be reduced? 2. What are the operational issues and their impact on cost? 3. What is the current technology readiness level (TRL) and what will it take to reach TRL 6? 4. What are the key enabling technologies and sequence for their accomplishment? 5 . What is the proposed implementation time frame? See Appendix A for the TIM Agenda and Appendix C for the AIST Program Terms of Reference.
Derived from text
Space Transportation System; Low Earth Orbits; Cost Reduction; Geosynchronous Orbits; Reusable Launch Vehicles; Technology Assessment; Lunar Orbits; Costs

19980000968 NASA, Washington, DC USA
National Aerospace Plane Thermal Development. (Latest citations from the Aerospace Database)
Jun. 1997; In English
Report No.(s): NASA/TM-97-113072; NAS 1.15:113072; PB97-860860; No Copyright; Avail: National Technical Information Service (NTIS)

The bibliography contains citations concerning thermal properties of the National Aerospace Plane (NASP). Analysis of thermal stress, and methods for determining thermal effects on the plane's supersonic structure are discussed. The citations also review temperature extremes that the vehicle is likely to encounter.
NTIS
National Aerospace Plane Program; Thermodynamic Properties; Aerospace Planes

19970040845 State of New Mexico, Las Cruces, NM USA
Business Plan for the Southwest Regional Spaceport: Executive Summary
Jul. 1997; In English; Original contains color illustrations
Contract(s)/Grant(s): NAGw-5131
Report No.(s): NASA/CR-97-206023; NAS 1.26:206023; No Copyright; Avail: CASI; A03, Hardcopy

A proposal for a commercial, full-service launch, tracking, and recovery complex for Reusable Launch Vehicles in New Mexico is presented. Vision, mission, business definition, competitive advantages, and business approach are formulated. Management plan and team structure are detailed, and anticipated market is described. Finance and marketing plans are presented. Financial analysis is performed.
Derived from text
Launching Bases; Proposals

19970040679 Space America, Inc., Huntsville, AL USA
Highly Reusable Space Transportation (HRST) Baseline Concepts and Analysis: Rocket/RBCC Options, Part 2, A Comparative Study
Woodcock, Gordon; Oct. 29, 1997; In English
Contract(s)/Grant(s): W-7-PP-03050(1F); NASA Order H-28681-D
Report No.(s): NASA/CR-97-112565; NAS 1.26:112565; No Copyright; Avail: CASI; A06, Hardcopy

This study is an extension of a previous effort by the Principal Investigator to develop baseline data to support comparative analysis of Highly Reusable Space Transportation (HRST) concepts. The analyses presented herin develop baseline data bases for two two-stage-to-orbit (TSTO) concepts: (1) Assisted horizontal take-off all rocket (assisted HTOHL); and (2) Assisted vertical take-off rocket based combined cycle (RBCC). The study objectives were to: (1) Provide configuration definitions and illustrations for assisted HTOHL and assisted RBCC; (2) Develop a rationalization approach and

compare these concepts with the HRST reference; and (3) Analyze TSTO configurations which try to maintain SSTO benefits while reducing inert weight sensitivity.
Author
Space Transportation; Reusable Spacecraft; Single Stage to Orbit Vehicles; Data Bases; Vertical Takeoff

19970040461 Lockheed Martin Corp., Palmdale, CA USA
X-33 Phase 2
1997; In English; Original contains color illustrations
Contract(s)/Grant(s): NCC8-115
Report No.(s): NASA-CR-205695; NAS 1.26:205695; No Copyright; Avail: CASI; A06, Hardcopy

In response to Clause 17 of the Cooperative Agreement NCC8-115, Lockheed Martin Skunk Works has compiled an Annual Performance Report of the X-33/RLV Program. This report consists of individual reports from all industry team members, as well as NASA team centers. Contract award was announced on July 2, 1996 and the first milestone was hand delivered to NASA MSFC on July 17, 1996. The first year has been one of growth and progress as all team members staffed up and embarked on the technical adventure of the 20th century... the ultimate goal . . a Single Stage to Orbit (SSTO) Reuseable Launch Vehicle (RLV).
Derived from text
X-33 Reusable Launch Vehicle; Single Stage to Orbit Vehicles; Launch Vehicles; Reusable Spacecraft

19970040167 NASA Ames Research Center, Moffett Field, CA USA
Near-Optimal Re-Entry Trajectories for Reusable Launch Vehicles
Chou, H.-C.; Ardema, M. D.; Bowles, J. V.; American Inst. of Aeronautics and Astronautics Journal; 1997; In English
Contract(s)/Grant(s): NCC2-5217; NCC2-5069
Report No.(s): NASA/CR-97-112611; NAS 1.26:112611; AIAA Paper 97-3582; Copyright; Avail: CASI; A03, Hardcopy

A near-optimal guidance law for the descent trajectory for earth orbit re-entry of a fully reusable single-stage-to-orbit pure rocket launch vehicle is derived. A methodology is developed to investigate using both bank angle and altitude as control variables and selecting parameters that maximize various performance functions. The method is based on the energy-state model of the aircraft equations of motion. The major task of this paper is to obtain optimal re-entry trajectories under a variety of performance goals: minimum time, minimum surface temperature, minimum heating, and maximum heading change; four classes of trajectories were investigated: no banking, optimal left turn banking, optimal right turn banking, and optimal bank chattering. The cost function is in general a weighted sum of all performance goals. In particular, the trade-off between minimizing heat load into the vehicle and maximizing cross range distance is investigated. The results show that the optimization methodology can be used to derive a wide variety of near-optimal trajectories.
Author
Reusable Launch Vehicles; Reentry Trajectories; Descent Trajectories; Equations of Motion; Heating; Time Temperature Parameter; Surface Temperature

19970037803 Pennsylvania State Univ., University Park, PA USA
An Experimental Study of Characteristic Combustion-Driven Flow for CFD Validation
Santoro, Robert J.; Oct. 1997; In English
Contract(s)/Grant(s): NAS8-38862
Report No.(s): NASA/CR-97-205735; NAS 1.26:205735; PSU-NAS8-38862-2; No Copyright; Avail: CASI; A07, Hardcopy

A series of uni-element rocket injector studies were completed to provide benchmark quality data needed to validate computational fluid dynamic models. A shear coaxial injector geometry was selected as the primary injector for study using gaseous hydrogen/oxygen and gaseous hydrogen/liquid oxygen propellants. Emphasis was placed on the use of nonintrusive diagnostic techniques to characterize the flowfields inside an optically-accessible rocket chamber. Measurements of the velocity and species fields were obtained using laser velocimetry and Raman spectroscopy, respectively. Qualitative flame shape information was also obtained using laser-induced fluorescence excited from OH radicals and laser light scattering studies of aluminum oxide particle seeded combusting flows. The gaseous hydrogen/liquid oxygen propellant studies for the shear coaxial injector focused on breakup mechanisms associated with the liquid oxygen jet under subcritical pressure conditions. Laser sheet illumination techniques were used to visualize the core region of the jet and a Phase Doppler Particle Analyzer was utilized for drop velocity, size and size distribution characterization. The results of these studies indicated that the shear coaxial geometry configuration was a relatively poor injector in terms of mixing. The oxygen core was observed to

extend well downstream of the injector and a significant fraction of the mixing occurred in the near nozzle region where measurements were not possible to obtain. Detailed velocity and species measurements were obtained to allow CFD model validation and this set of benchmark data represents the most comprehensive data set available to date. As an extension of the investigation, a series of gas/gas injector studies were conducted in support of the X-33 Reusable Launch Vehicle program. A Gas/Gas Injector Technology team was formed consisting of the Marshall Space Flight Center, the NASA Lewis Research Center, Rocketdyne and Penn State. Injector geometries studied under this task included shear and swirl coaxial configurations as well as an impinging jet injector.
Author
Optical Measurement; Flow Distribution; Nonintrusive Measurement; Thrust Chambers; Velocity Measurement; Injectors; Combustible Flow

19970029012 Analytic Services and Materials, Inc., Hampton, VA USA
Bonding and Sealing Evaluations for Cryogenic Tanks
Glass, David E.; Aug. 1997; In English
Contract(s)/Grant(s): NAS1-96014; RTOP 242-33-01
Report No.(s): NASA-CR-201734; NAS 1.26:201734; ASM-LS05-97-01; No Copyright; Avail: CASI; A03, Hardcopy

Several different cryogenic tank concepts are being considered for reusable launch vehicles (RLV'S) . Though different tank concepts are being considered, many will require that the cryogenic insulation be evacuated and be bonded to a structure. In this work, an attempt was made to evaluate the effectiveness of maintaining a vacuum on a specimen where foam or honeycomb core was encased within Gr/Ep. In addition to these tests, flatwise adhesion pull off tests were performed at room temperature with PR 1664, EA 9394, FM-300, Crest 3170, and HT 435 adhesives. The materials bonded included Gr/Ep, Gr/BMI, Al, and stainless steel facesheets, and Ti honeycomb, Hexcel honeycomb, and Rohacell foam core materials.
Author
Bonding; Sealing; Cryogenic Fluid Storage; Adhesion Tests; Honeycomb Cores; Reusable Launch Vehicles; Bonded Joints

19970028584 Princeton Synergetics, Inc., NJ USA
STARS: The Space Transportation Architecture Risk System
Greenberg, Joel S.; Apr. 1997; In English
Contract(s)/Grant(s): NASA Order H-27284D
Report No.(s): NASA-CR-2051;87; NAS 1.26:205187; No Copyright; Avail: CASI; A04, Hardcopy

Because of the need to perform comparisons between transportation systems that are likely to have significantly different levels of risk, both because of differing degrees of freedom in achieving desired performance levels and their different states of development and utilization, an approach has been developed for performing early comparisons of transportation architectures explicitly taking into account quantitative measures of uncertainty and resulting risk. The approach considers the uncertainty associated with the achievement of technology goals, the effect that the achieved level of technology will have on transportation system performance and the relationship between transportation system performance/capability and the ability to accommodate variations in payload mass. The consequences of system performance are developed in terms of expected values and associated standard deviations of nonrecurring, recurring and the present value of transportation system life cycle cost. Typical results are presented to illustrate the application of the methodology.
Author
Space Transportation System; Risk; Life Cycle Costs; Monte Carlo Method

19970028362 Boeing Defense and Space Group, Canoga Park, CA USA
Space Shuttle Main Engine Start with Off-Nominal Propellant Inlet Pressures
Bradley, Michael; 1997; In English; 33rd Joint Propulsion, 6-9 Jul. 1997, Seattle, WA, USA
Contract(s)/Grant(s): NCC8-45
Report No.(s): NASA-CR-205359; NAS 1.26:205359; AIAA Paper 97-2687; No Copyright; Avail: CASI; A03, Hardcopy

This paper describes Rocketdyne's successful analysis and demonstration of the Space Shuttle Main Engine (SSME) operation at off-nominal propellant inlet conditions during the Reusable Launch Vehicle (RLV) evaluation tests. The nominal inlet condition range is: 103 to 111 psia and 170.5 to 178 deg. R for the oxidizer and 43 to 47 psia and 37 to 40 deg. R for the fuel. The SSME start was successfully demonstrated with engine inlet pressures of 50 psia liquid oxygen (LOX) with

subcooled LOX at 160 deg R and 38 psia fuel at 38 deg. R. Four tests were used to incrementally modify the start sequence to demonstrate the final goal.
Author
Space Shuttle Main Engine; Liquid Oxygen; Engine Inlets; Oxidizers; Mixing Ratios; Liquid Flow

19970028361 Boeing Defense and Space Group, Canoga Park, CA USA
Space Shuttle Main Engine Off-Nominal Low Power Level Operation
Bradley, Michael; 1997; In English; 33rd Joint Propulsion, 6-9 Jul. 1997, Seattle, WA, USA
Contract(s)/Grant(s): NCC8-45
Report No.(s): NASA-CR-205360; NAS 1.26:205360; AIAA Paper 97-2685; Copyright; Avail: CASI; A03, Hardcopy

This paper describes Rocketdyne's successful analysis and demonstration of the Space Shuttle Main Engine (SSME) operation at off-nominal power levels during Reusable Launch Vehicle (RLV) evaluation tests. The nominal power level range for the SSME is from 65% rated power level (RPL) to 109% RPL. Off-nominal power levels incrementally demonstrated were: 17% RPL, 22% RPL, 27% RPL, 40% RPL, 45% RPL, and 50% RPL. Additional achievements during low power operation included: use of a hydrostatic bearing High Pressure Oxidizer Turbopump (HPOTP), nominal High Pressure Fuel Turbopump (HPFTP) first rotor critical speed operation, combustion stability at low power levels, and refined definition of nozzle flow separation heat loads.
Author
Space Shuttle Main Engine; Turbine Pumps; Separated Flow; Oxidizers; Nozzle Flow; Hydrostatic Pressure; Combustion Stability

19970028022 NASA Langley Research Center, Hampton, VA USA
Systems Challenges for Hypersonic Vehicles
Hunt, James L.; Laruelle, Gerard; Wagner, Alain; 1997; In English; Future Aerospace Technology in the Service of the Alliance, 14-16 Apr. 1997, Palaiseau, France
Report No.(s): NASA-TM-112908; NAS 1.15:112908; AGARD-Paper-C37; No Copyright; Avail: CASI; A03, Hardcopy

This paper examines the system challenges posed by fully reusable hypersonic cruise airplanes and access to space vehicles. Hydrocarbon and hydrogen fueled airplanes are considered with cruise speeds of Mach 5 and 10, respectively. The access to space matrix is examined. Airbreathing and rocket powered, single- and two-stage vehicles are considered. Reference vehicle architectures are presented. Major systems/subsystems challenges are described. Advanced, enhancing systems concepts as well as common system technologies are discussed.
Author
Hypersonic Vehicles; Hypersonics; Hydrogen Fuels; Hydrocarbon Fuels; Air Breathing Engines; Rocket Engines

19970027074 NASA Ames Research Center, Moffett Field, CA USA
Evaluation of Thermal Control Coatings for Flexible Ceramic Thermal Protection Systems
Kourtides, Demetrius; Carroll, Carol; Smith, Dane; Guzinski, Mike; Marschall, Jochen; Pallix, Joan; Ridge, Jerry; Tran, Duoc; Jul. 1997; In English
Contract(s)/Grant(s): RTOP 242-20-01
Report No.(s): NASA-TM-112199; NAS 1.15:112199; A-976757; No Copyright; Avail: CASI; A03, Hardcopy

This report summarizes the evaluation and testing of high emissivity protective coatings applied to flexible insulations for the Reusable Launch Vehicle technology program. Ceramic coatings were evaluated for their thermal properties, durability, and potential for reuse. One of the major goals was to determine the mechanism by which these coated blanket surfaces become brittle and try to modify the coatings to reduce or eliminate embrittlement. Coatings were prepared from colloidal silica with a small percentage of either SiC or SiB6 as the emissivity agent. These coatings are referred to as gray C-9 and protective ceramic coating (PCC), respectively. The colloidal solutions were either brushed or sprayed onto advanced flexible reusable surface insulation blankets. The blankets were instrumented with thermocouples and exposed to reentry heating conditions in the Ames Aeroheating Arc Jet Facility. Post-test samples were then characterized through impact testing, emissivity measurements, chemical analysis, and observation of changes in surface morphology. The results show that both coatings performed well in arc jet tests with backface temperatures slightly lower for the PCC coating than with gray C-9. Impact testing showed that the least extensive surface destruction was experienced on blankets with lower areal density coatings.
Author
Thermal Control Coatings; Thermal Protection; Reusable Launch Vehicles; Reentry Effects; Protective Coatings; Durability; Chemical Analysis; Ceramic Coatings; Brittleness; Aerodynamic Heating

19970026958 NASA Ames Research Center, Moffett Field, CA USA

Surface Catalysis and Characterization of Proposed Candidate TPS for Access-to-Space Vehicles

Stewart, David A.; Jul. 1997; In English; Original contains color illustrations

Contract(s)/Grant(s): RTOP 242-30-01

Report No.(s): NASA-TM-112206; NAS 1.15:112206; A-976955; No Copyright; Avail: CASI; A04, Hardcopy

Surface properties have been obtained on several classes of thermal protection systems (TPS) using data from both side-arm-reactor and arc-jet facilities. Thermochemical stability, optical properties, and coefficients for atom recombination were determined for candidate TPS proposed for single-stage-to-orbit vehicles. The systems included rigid fibrous insulations, blankets, reinforced carbon carbon, and metals. Test techniques, theories used to define arc-jet and side-arm-reactor flow, and material surface properties are described. Total hemispherical emittance and atom recombination coefficients for each candidate TPS are summarized in the form of polynomial and Arrhenius expressions.

Author

Thermal Protection; Catalysis; Surface Properties; Thermochemical Properties; Emittance; Atomic Recombination; Recombination Coefficient

19970024873 NASA Lewis Research Center, Cleveland, OH USA

Comparison of High Aspect Ratio Cooling Channel Designs for a Rocket Combustion Chamber

Wadel, Mary F.; Jun. 1997; In English; 33rd Joint Propulsion Conference and Exhibit, 6-9 Jul. 1997, Seattle, WA, USA

Contract(s)/Grant(s): RTOP 242-20-06

Report No.(s): NASA-TM-107473; NAS 1.15:107473; E-10762; AIAA Paper 97-2913; No Copyright; Avail: CASI; A03, Hardcopy

An analytical investigation on the effect of high aspect ratio (height/width) cooling channels, considering different coolant channel designs, on hot-gas-side wall temperature and coolant pressure drop for a liquid hydrogen cooled rocket combustion chamber, was performed. Coolant channel design elements considered were: length of combustion chamber in which high aspect ratio cooling was applied, number of coolant channels, and coolant channel shape. Seven coolant channel designs were investigated using a coupling of the Rocket Thermal Evaluation code and the Two-Dimensional Kinetics code. Initially, each coolant channel design was developed, without consideration for fabrication, to reduce the hot-gas-side wall temperature from a given conventional cooling channel baseline. These designs produced hot-gas-side wall temperature reductions up to 22 percent, with coolant pressure drop increases as low as 7.5 percent from the baseline. Fabrication constraints for milled channels were applied to the seven designs. These produced hot-gas-side wall temperature reductions of up to 20 percent, with coolant pressure drop increases as low as 2 percent. Using high aspect ratio cooling channels for the entire length of the combustion chamber had no additional benefit on hot-gas-side wall temperature over using high aspect ratio cooling channels only in the throat region, but increased coolant pressure drop 33 percent. Independent of coolant channel shape, high aspect ratio cooling was able to reduce the hot-gas-side wall temperature by at least 8 percent, with as low as a 2 percent increase in coolant pressure drop. The design with the highest overall benefit to hot-gas-side wall temperature and minimal coolant pressure drop cooling can now be done in relatively short periods of time with multiple iterations.

Author

Combustion Chambers; High Aspect Ratio; Cooling; Channels; Computer Programs; Design

19970024861 NASA Lewis Research Center, Cleveland, OH USA

Performance Tests of a Liquid Hydrogen Propellant Densification Ground System for the X33/RLV

Tomsik, Thomas M.; Jun. 1997; In English; 33rd Joint Propulsion Conference and Exhibit, 6-9 Jul. 1997, Seattle, WA, USA

Contract(s)/Grant(s): RTOP 242-33-01

Report No.(s): NASA-TM-107469; NAS 1.15:107469; AIAA Paper 97-2976; E-10737; No Copyright; Avail: CASI; A03, Hardcopy

A concept for improving the performance of propulsion systems in expendable and single-stage-to-orbit (SSTO) launch vehicles much like the X33/RLV has been identified. The approach is to utilize densified cryogenic liquid hydrogen (LH2) and liquid oxygen (LOX) propellants to fuel the propulsion stage. The primary benefit for using this relatively high specific impulse densified propellant mixture is the subsequent reduction of the launch vehicle gross lift-off weight. Production of densified propellants however requires specialized equipment to actively subcool both the liquid oxygen and liquid hydrogen to temperatures below their normal boiling point. A propellant densification unit based on an external thermodynamic vent principle which operates at subatmospheric pressure and supercold temperatures provides a means for the LH2 and LOX densification process to occur. To demonstrate the production concept for the densification of the liquid hydrogen propellant, a system comprised of a multistage gaseous hydrogen compressor, LH2 recirculation pumps and a cryogenic LH2 heat

exchanger was designed, built and tested at the NASA Lewis Research Center (LeRC). This paper presents the design configuration of the LH2 propellant densification production hardware, analytical details and results of performance testing conducted with the hydrogen densifier Ground Support Equipment (GSE).
Author
Propulsion System Performance; Liquid Hydrogen; Densification; Cryogenic Rocket Propellants; Performance Tests; Design Analysis

19970023998 GPS Solutions, Inc., Carson City, NV USA
Reusable Launch Vehicle Research
Brinker, Stan; Crisalli, Dave; French, Jim; Henry, John; Quisenberry, Bob; Scottoline, Charles; Shell, Dale; Sullivan, Greg; May 31, 1997; In English
Contract(s)/Grant(s): NAS8-97212
Report No.(s): NASA-CR-204526; NAS 1.26:204526; No Copyright; Avail: CASI; A24, Hardcopy

NASA is engaged in the development of technologies and experimental Reusable Launch Vehicles (RLV) for the purpose of significantly reducing the cost of space access through the commercialization of space launch. This involves the unprecedented integration of often conflicting business and technical requirements and activities. The following efforts shall enhance NASA's ability to successfully enable low cost space access while meeting National needs.
Author
Reusable Launch Vehicles; Cost Reduction; Spacecraft Launching

19970017261 Santa Clara Univ., CA USA
Minimum Heating Re-Entry Trajectories for Advanced Hypersonic Launch Vehicles
Windhorst, Robert; Feb. 05, 1997; In English
Contract(s)/Grant(s): NCC2-5165
Report No.(s): NASA-CR-204339; NAS 1.26:204339; No Copyright; Avail: CASI; A04, Hardcopy

Optimal re-entry trajectories are generated for reusable launch vehicles which minimize: (1) the heat absorbed at the vehicle surface, (2) the lower surface temperature, and (3) the heat absorbed by the internal structure. The approach uses the energy state approximation technique and a finite control volume heat transfer code coupled to a flight path integration code. These trajectories are compared to the optimal re-entry trajectory minimizing the integrated convective heat rate to determine which trajectory produces the minimum internal structural temperatures for a given thermal protection system. Three different thermal protection systems are considered: tile, blanket, and metallic.
Author
Reentry Trajectories; Reusable Launch Vehicles; Hypersonic Vehicles; Convective Heat Transfer

19970016839 NASA Ames Research Center, Moffett Field, CA USA
Near-Optimal Operation of Dual-Fuel Launch Vehicles
Ardema, M. D.; Chou, H. C.; Bowles, J. V.; 1996; In English; Atmospheric Flight Mechanics, 29-31 Jul. 1996, San Diego, CA, USA
Contract(s)/Grant(s): NCC2-5165
Report No.(s): NASA-CR-204358; NAS 1.26:204358; AIAA Paper 96-3397; Copyright; Avail: CASI; A03, Hardcopy

A near-optimal guidance law for the ascent trajectory from earth surface to earth orbit of a fully reusable single-stage-to-orbit pure rocket launch vehicle is derived. Of interest are both the optimal operation of the propulsion system and the optimal flight path. A methodology is developed to investigate the optimal throttle switching of dual-fuel engines. The method is based on selecting propulsion system modes and parameters that maximize a certain performance function. This function is derived from consideration of the energy-state model of the aircraft equations of motion. Because the density of liquid hydrogen is relatively low, the sensitivity of perturbations in volume need to be taken into consideration as well as weight sensitivity. The cost functional is a weighted sum of fuel mass and volume; the weighting factor is chosen to minimize vehicle empty weight for a given payload mass and volume in orbit.
Author
Ascent Trajectories; Single Stage to Orbit Vehicles; Propulsion System Performance; Flight Paths; Guidance (Motion); Algorithms; Trajectory Optimization; Mach Number

19970015639 NASA Ames Research Center, Moffett Field, CA USA
Director's Discretionary Fund Report for Fiscal Year 1996
Mar. 1997; In English
Contract(s)/Grant(s): RTOP 274-53-71
Report No.(s): NASA-TM-110445; NAS 1.15:110445; A-975819; No Copyright; Avail: CASI; A07, Hardcopy

Topics covered include: Waterproofing the Space Shuttle tiles, thermal protection system for Reusable Launch Vehicles, computer modeling of the thermal conductivity of cometary ice, effects of ozone depletion and ultraviolet radiation on plants, a novel telemetric biosensor to monitor blood pH on-line, ion mobility in polymer electrolytes for lithium-polymer batteries, a microwave-pumped far infrared photoconductor, and a new method for measuring cloud liquid vapor using near infrared remote sensing. Also included: laser-spectroscopic instrument for turbulence measurement, remote sensing of aircraft contrails using a field portable imaging interferometer, development of a silicon-micromachined gas chromatography system for determination of planetary surface composition, planar Doppler velocimetry, chaos in interstellar chemistry, and a limited pressure cycle engine for high-speed output.

Derived from text

Thermal Protection; Ozone Depletion; Remote Sensing; Climate Change; Photoconductors; Comets; Turbulent Flow; Semiconductor Lasers; Gas Chromatography; Interstellar Chemistry; Tilt Rotor Aircraft

19970014941 NASA Langley Research Center, Hampton, VA USA
Multidisciplinary Approach to Aerospike Nozzle Design
Korte, J. J.; Salas, A. O.; Dunn, H. J.; Alexandrov, N. M.; Follett, W. W.; Orient, G. E.; Hadid, A. H.; Feb. 1997; In English
Contract(s)/Grant(s): RTOP 522-31-41-02
Report No.(s): NASA-TM-110326; NAS 1.15:110326; No Copyright; Avail: CASI; A03, Hardcopy

A model of a linear aerospike rocket nozzle that consists of coupled aerodynamic and structural analyses has been developed. A nonlinear computational fluid dynamics code is used to calculate the aerodynamic thrust, and a three-dimensional finite-element model is used to determine the structural response and weight. The model will be used to demonstrate multidisciplinary design optimization (MDO) capabilities for relevant engine concepts, assess performance of various MDO approaches, and provide a guide for future application development. In this study, the MDO problem is formulated using the multidisciplinary feasible (MDF) strategy. The results for the MDF formulation are presented with comparisons against separate aerodynamic and structural optimized designs. Significant improvements are demonstrated by using a multidisciplinary approach in comparison with the single-discipline design strategy.

Author

Nozzle Design; Engine Design; Multidisciplinary Design Optimization; Finite Element Method; Rocket Nozzles

19970014807 Georgia Inst. of Tech., Atlanta, GA USA
Supercharged Ejector Ramjet with Maglifter Launch Assist (for the Argus Concept)
Olds, John R.; Apr. 08, 1997; In English
Contract(s)/Grant(s): NCC8-108
Report No.(s): NASA-CR-204070; NAS 1.26:204070; No Copyright; Avail: CASI; A03, Hardcopy

Main propulsion system consists of two 330 klb-class LOX/LH2 Supercharged Ejector RamJet (SERJ) RBCC engines. Engines are capable of multi-mode operation including supercharged ejector, fan-ramjet, ramjet, and pure rocket modes. A low thrust 'fan-only' mode is available for powered landing, go-around, taxi, etc. (5 minutes at full throttle). Baseline Maglifter launch assist system consists of a magnetically levitated sled and track system that accelerates the vehicle to approximately 800 fps. At an average acceleration of 1-g, the Maglifter track will be approximately 10,000 ft. long. The baseline system assumes an exposed, horizontal track constructed at a sea-level launch site. Power for the Maglifter is drawn from a large, nearby energy storage system (capacitor-like). The Argus launch vehicle is a moderate lift-to-drag ratio fuselage (circular cross section), highly reusable launch vehicle. It operates autonomously (or remotely). There is no crew cabin. It is capable of delivering and returning a 20 klb payload or 6 passengers to a 100 nmi. circular low earth orbit, due east from KSC (about 10 klb to Space Station Alpha). The fan-derived propulsion modes allow improved aborts and landing operations. Without a payload, the vehicle is capable of transcontinental self-ferry.

Derived from text

Propulsion; Reusable Launch Vehicles; Ramjet Engines; Ejectors

19970012889 NASA Langley Research Center, Hampton, VA USA

Effects of Stiffening and Mechanical Load on Thermal Buckling of Stiffened Cylindrical Shells

Johnson, Theodore F.; Card, Michael F.; Apr. 12, 1995; In English; 36th Structures, Structural Dynamics and Materials Conference, 10-12 Apr. 1995, New Orleans, LA, USA

Report No.(s): NASA-TM-112006; NAS 1.15:112006; AIAA Paper 95-1317; No Copyright; Avail: CASI; A02, Hardcopy

A study of thermal buckling of stiffened cylindrical shells with the proportions of a preliminary supersonic transport fuselage design (1970) is presented. The buckling analysis is performed using an axisymmetric shell-of-revolution code, BOSOR4. The effects of combined mechanical (axial loading) and thermal loading (heated skins) are investigated. Results indicate that the location of longitudinal eccentric stiffening has a very large effect on the thermal buckling strength of longitudinally stiffened shells, and on longitudinally stiffened shells with rings.

Author

Thermal Buckling; Cylindrical Shells; Stiffening; Mechanical Properties; Applications Programs (Computers); Axial Loads; Transatmospheric Vehicles; Digital Simulation; Computerized Simulation

19970012468 NASA Ames Research Center, Moffett Field, CA USA

Materials Testing on the DC-X and DC-XA

Smith, Dane; Carroll, Carol; Marschall, Jochen; Pallix, Joan; Jan. 1997; In English; Original contains color illustrations

Contract(s)/Grant(s): RTOP 242-30-01

Report No.(s): NASA-TM-110430; NAS 1.15:110430; A-975493; No Copyright; Avail: CASI; A03, Hardcopy

Flight testing of thermal protection materials has been carried out over a two year period on the base heat shield of the Delta Clipper (DC-X and DC-XA), as well on a body flap. The purpose was to use the vehicle as a test bed for materials and more efficient repair or maintenance processes which would be potentially useful for application on new entry vehicles (i.e., X-33, RLV, planetary probes), as well as on the existing space shuttle orbiters. Panels containing Thermal Protection Systems (TPS) and/or structural materials were constructed either at NASA Ames Research Center or at McDonnell Douglas Aerospace (MDA) and attached between two of the four thrusters in the base heat shield of the DC-X or DC-XA. Three different panels were flown on DC-X flights 6, 7, and 8. A total of 7 panels were flown on DC-XA flights 1, 2, and 3. The panels constructed at Ames contained a variety of ceramic TPS including flexible blankets, tiles with high emissivity coatings, lightweight ceramic ablators and other ceramic composites. The MDS test panels consisted primarily of a variety of metallic composites. This report focuses on the ceramic TPS test results.

Author

Thermal Protection; Delta Clipper; Heat Shielding; Ceramic Matrix Composites; Ablative Materials

19970012335 Georgia Inst. of Tech., Atlanta, GA USA

Technology and Risk Assessment Using a Robust Design Simulation Methodology, Phase 1, Definition and Initial Implementation of RDS

Mavris, Dimitri N.; 1997; In English

Report No.(s): NASA-CR-203308; NAS 1.26:203308; No Copyright; Avail: CASI; A02, Hardcopy

This year we were able to implement the Robust Design Methodology developed at Aerospace System Design Laboratory (ASDL) into our aerospace engineering graduate design curriculum by having this year's design team execute the methodology for their team project. The project was focused on the preliminary design of an High Speed Civil Transport (HSCT) concept, and more specifically, the development of a design configuration that is insensitive to variability induced by the economic environment it is operated in. This work is summarized in the proceedings of this year's ASDL external advisory board symposium held at Georgia Tech in May 1996 (proceedings have already been mailed to sponsor) and a 300 page final report submitted by the team which will be mailed.

Derived from text

Aerospace Engineering; Aircraft Design; Aerospace Planes; Design to Cost

19970011757 Society of Automotive Engineers, Inc., Warrendale, PA USA

Results of a Rocket-Based Combined-Cycle SSTO Design Using Parametric MDO Methods

Olds, John R.; 1994; ISSN 0148-7191; In English; Aerospace Atlantic, 18-22 Apr. 1994, Dayton, OH, USA

Contract(s)/Grant(s): NCC1-168; NAGw-1331

Report No.(s): NASA-CR-203224; NAS 1.26:203224; SAE-TP-941165; Copyright; Avail: CASI; A03, Hardcopy

This paper reports the results of the second phase of a research project to characterize and optimize the design of an

advanced launch vehicle for human access to low earth orbit. The vehicle makes use of Rocket-Based Combined-Cycle (RBCC) propulsion - a concept combining operating modes of an ejector, ramjet, scramjet, and rocket in a single engine. This research builds on previous work focused on advanced multiple mode propulsion concepts and advanced conical acceleration-class Single-Stage-To-Orbit (SSTO) launch vehicles. Three systems level design variables of interest were optimized using Multidisciplinary Design Optimization (MDO) techniques. Specifically, Taguchi's method of robust design was used to identify a combination of variables that minimize the vehicle sensitivity to unpredictable changes in engine weights and performance. In addition, a second-order Response Surface Method (RSM) was used to approximate the design space and predict the minimum dry weight vehicle. The optimized vehicle results (weights, dimensions, performance) are favorably compared with other SSTO designs including rocket and airbreathing concepts.
Author
Design Analysis; Launch Vehicles; Low Earth Orbits

19970009763 NASA Johnson Space Center, Houston, TX USA
A Collection of Technical Papers
A Collection of Technical Papers from the 6th Space Logistics Symposium; 1995; In English; 6th Space Logistics Symposium, 22-24 Feb. 1995, Houston, TX, USA
Report No.(s): NASA-TM-112136; NAS 1.15:112136; Copyright; Avail: CASI; A10, Hardcopy

Papers presented at the 6th Space Logistics Symposium covered such areas as: The International Space Station; The Hubble Space Telescope; Launch site computer simulation; Integrated logistics support; The Baikonur Cosmodrome; Probabalistic tools for high confidence repair; A simple space station rescue vehicle; Integrated Traffic Model for the International Space Station; Packaging the maintenance shop; Leading edge software support; Storage information management system; Consolidated maintenance inventory logistics planning; Operation concepts for a single stage to orbit vehicle; Mission architecture for human lunar exploration; Logistics of a lunar based solar power satellite scenario; Just in time in space; NASA acquisitions/logistics; Effective transition management; Shuttle logistics; and Revitalized space operations through total quality control management.
Derived from text
International Space Station; Single Stage to Orbit Vehicles; Solar Power Satellites; Space Logistics; Logistics Management; Rescue Operations; Support Systems; Total Quality Management

19970009604 NASA Langley Research Center, Hampton, VA USA
Aerodynamic Characteristics of a Vertical Takeoff Vertical Landing (VTVL) Single Stage to Orbit Vehicle from M(sub infinity) = 0.1 to 10
Woods, W. C.; Merski, N. R.; 1995; In English; 13th Applied Aerodynamics Conference, 19-22 Jun. 1995, San Diego, CA, USA
Report No.(s): NASA-TM-112160; NAS 1.15:112160; AIAA Paper 95-1828; Copyright; Avail: CASI; A03, Hardcopy

Future access to space studies have considered a wide variety of follow on options to augment/replace the Space Shuttle. One option is a vertical takeoff and landing single stage to orbit vehicle. Experimental aerodynamic characteristics have been obtained on a generic design representative of such a vehicle; i.e., spherical nose blunting, a forebody of revolution and flat surfaces on the afterbody to accommodate control surfaces. Data has been obtained at subsonic, supersonic, and hypersonic Mach numbers to aid in determining cross range capability and performance, stability and control characteristics. The baseline configuration is longitudinally and laterally unstable, but appears to have the control authority to provide trim and stability augmentation. At subsonic speeds and angles of attack above 30 deg large lateral forces occur at zero sideslip and the lateral/directional characteristics are not well behaved.
Author
Aerodynamic Characteristics; Single Stage to Orbit Vehicles; Afterbodies; Angle of Attack; Control Surfaces; Noses (Forebodies); Vertical Landing; Vertical Takeoff; Mach Number

19970008365 NASA Marshall Space Flight Center, Huntsville, AL USA
Computational Aerodynamics Analysis of Future Launch Vehicle Configurations
Vu, B.; McConnaughey, P.; Huddleston, D.; 1994; In English; 12th Applied Aerodynamics, 20-22 Jun. 1994, Colorado Springs, CO, USA
Report No.(s): NASA-TM-112141; NAS 1.15:112141; AIAA Paper 94-1936; No Copyright; Avail: CASI; A02, Hardcopy

An implicit finite volume high resolution scheme is applied for predicting three-dimensional, inviscid flow field over

several launch vehicles including the National Launch System (NLS), the Single-Stage-To-Orbit (SSTO) lifting bodies, and wing-body vehicles. Simulations about NLS configurations were used to help benchmark Computational Fluid Dynamics (CFD) capabilities at NASA/MSFC. Likewise, simulations about SSTO were used to provide aerodynamic data to structure group to calculate structural wing loading for preliminary conceptual designs.
Author
Computational Fluid Dynamics; Computerized Simulation; Digital Simulation; Single Stage to Orbit Vehicles; Three Dimensional Flow; Lifting Bodies; Launch Vehicles; Launch Vehicle Configurations; Finite Volume Method

19970006985 NASA Marshall Space Flight Center, Huntsville, AL USA
Impact Characteristics of Candidate Materials for Single-Stage-to-Orbit (SSTO) Technology
Nettles, Alan; 1995; In English; 40th International SAMPE Symposium and Exhibition, 8-11 May 1995, Anaheim, CA, USA
Report No.(s): NASA-TM-111794; NAS 1.15:111794; No Copyright; Avail: CASI; A03, Hardcopy

Four fiber/resin systems were compared for resistance to damage and damage tolerance. One toughened epoxy and three toughened bismaleimide (BMI) resins were used., all with IM7 carbon fiber reinforcement. A statistical design of experiments technique was used to evaluate the effects of impact energy, specimen thickness and tup diameter on the damage area and residual compression-after-impact (CAI) strength. Results showed that two of the BMI systems sustained relatively large damage areas yet had an excellent retention of CAI strength.
Author
Impact Strength; Impact Resistance; Polymer Matrix Composites; Bismaleimide; Polyimide Resins; Epoxy Resins; Impact Tests; Nondestructive Tests; Spacecraft Construction Materials

19970005470 NASA Langley Research Center, Hampton, VA USA
Comparison of Two Multidisciplinary Optimization Strategies for Launch-Vehicle Design
Braun, R. D.; Powell, R. W.; Lepsch, R. A.; Stanley, D. O.; Kroo, I. M.; Journal of Spacecraft and Rockets; Jun. 1995; Volume 33, No. 3; In English
Report No.(s): NASA-TM-111822; NAS 1.15:111822; No Copyright; Avail: CASI; A02, Hardcopy

The investigation focuses on development of a rapid multidisciplinary analysis and optimization capability for launch-vehicle design. Two multidisciplinary optimization strategies in which the analyses are integrated in different manners are implemented and evaluated for solution of a single-stage-to-orbit launch-vehicle design problem. Weights and sizing, propulsion, and trajectory issues are directly addressed in each optimization process. Additionally, the need to maintain a consistent vehicle model across the disciplines is discussed. Both solution strategies were shown to obtain similar solutions from two different starting points. These solutions suggests that a dual-fuel, single-stage-to-orbit vehicle with a dry weight of approximately 1.927 x 10(exp 5)lb, gross liftoff weight of 2.165 x 10(exp 6)lb, and length of 181 ft is attainable. A comparison of the two approaches demonstrates that treatment or disciplinary coupling has a direct effect on optimization convergence and the required computational effort. In comparison with the first solution strategy, which is of the general form typically used within the launch vehicle design community at present, the second optimization approach is shown to he 3-4 times more computationally efficient.
Author
Design Analysis; Optimization; Launch Vehicles; Convergence; Single Stage to Orbit Vehicles

19970005454 NASA Marshall Space Flight Center, Huntsville, AL USA
Advanced Low-Cost O2/H2 Engines for the SSTO Application
Goracke, B. David; Levack, Daniel J. H.; Nixon, Robert F.; 1994; In English; 30th Joint Propulsion, 27-29 Jun. 1994, Indianapolis, IN, USA
Report No.(s): NASA-TM-111897; NAS 1.15:111897; AIAA Paper 94-3317; Copyright; Avail: CASI; A02, Hardcopy

The recent NASA Access to Space study examined future Earth to orbit (ETO) transportation needs and fleets out to 2030. The baseline in the option 3 assessment was a single stage to orbit (SSTO) vehicle. A study was conducted to assess the use of new advanced low cost O2/H2 engines for this SSTO application. The study defined baseline configurations and ground rules and defined six engine cycles to explore engine performance. The cycles included an open cycle, and a series of closed cycles with varying abilities to extract energy from the propellants to power he turbomachinery. The cycles thus varied in the maximum chamber pressure they could reach and in their weights at any given chamber pressure. The weight of each cycle was calculated for two technology levels versus chamber pressure up to the power limit of the cycle. The performance in the SSTO mission was then modeled using the resulting engine weights and specific impulse performance using the Access to

Space option 3 vehicle. The results showed that new O2/H2 engines are viable and competitive candidates for the SSTO application using chamber pressures of 4,000 psi.
Author
Single Stage to Orbit Vehicles; Thermodynamic Cycles; Specific Impulse; Engine Design; Propulsion System Configurations; Closed Cycles; Liquid Propellant Rocket Engines

19970005361 National Aeronautics and Space Administration. Langley Research Center, Hampton, VA USA
Development of Metallic Thermal Protection Systems for the Reusable Launch Vehicle
Blosser, Max L.; Oct. 1996; In English; Space Technology and Applications International Forum, 26-30 Jan. 1997, Albuquerque, NM, USA
Contract(s)/Grant(s): RTOP 242-20-02-01
Report No.(s): NASA-TM-110296; NAS 1.15:110296; No Copyright; Avail: CASI; A03, Hardcopy

A reusable Thermal Protection System (TPS) that is not only lightweight, but durable, operable and cost effective is one of the technologies required by the Reusable Launch Vehicle (RLV) to achieve the goal of drastically reducing the cost of delivering payload to orbit. Metallic TPS is one of the systems being developed to meet this challenge. Current efforts involve improving the superalloy honeycomb TPS concept, which consists of a foil-gage metallic box encapsulating a low density fibrous insulation, and evaluating it for RLV requirements. The superalloy honeycomb TPS concept is mechanically attached to the vehicle structure. Improvements include more efficient internal insulation, a simpler, lighter weight configuration, and a quick-release fastener system for easier installation and removal. Evaluation includes thermal and structural analysis, fabrication and testing of both coupons and TPS panels under conditions simulating RLV environments. Coupons of metallic honeycomb sandwich, representative of the outer TPS surface, were subjected to low speed impact, hypervelocity impact, and rain erosion testing as well as subsequent arcjet exposure. Arrays of TPS panels have been subjected to radiant heating in a thermal/vacuum facility, aerodynamic heating in an arcjet facility and acoustic loading.
Author
Thermal Protection; Structural Analysis; Reusable Launch Vehicles; Measuring Instruments; Payloads; Loads (Forces); Hypervelocity Impact; Honeycomb Structures; Heat Resistant Alloys; Aerodynamic Heating; Acoustic Fatigue

19970005131 NASA Langley Research Center, Hampton, VA USA
Airbreathing/Rocket Single-Stage-to-Orbit Design Matrix
Hunt, James L.; 1995; In English; 6th International Aerospace Planes and Hypersonics Technologies, 3-7 Apr. 1995, Chattanooga, TN, USA
Report No.(s): NASA-TM-112140; NAS 1.15:112140; AIAA Paper 95-6011; Copyright; Avail: CASI; A03, Hardcopy

A definitive design/performance study was performed on a single-stage-to-orbit (SSTO) airbreathing propelled orbital vehicle with rocket propulsion augmentation in the Access to Space activities during 1993. A credible reference design was established, but by no means an optimum. The results supported the viability of SSTO airbreathing/rocket vehicles for operational scenarios and indicated compelling reasons to continue to explore the design matrix. This paper will (1) summarize the Access to Space design activity from the SSTO airbreathing/rocket perspective, (2) present an airbreathing/rocket SSTO design matrix established for continued optimization of the design space, and (3) focus on the compelling reasons for airbreathing vehicles in Access to Space scenarios.
Author
Single Stage to Orbit Vehicles; Air Breathing Engines; Systems Integration; System Effectiveness

19970005122 NASA Langley Research Center, Hampton, VA USA
Subsonic-to-Hypersonic Aerodynamic Characteristics for a Winged, Circular-Body, Single-Stage-to-Orbit Spacecraft Configuration
Phillips, W. P.; Engelund, W. C.; 1995; In English; Applied Aerodynamics, 19-22 Jun. 1995, San Diego, CA, USA
Report No.(s): NASA-TM-111926; NAS 1.15:111926; AIAA Paper 95-1848; Copyright; Avail: CASI; A03, Hardcopy

Experimental aerodynamic characteristics were obtained for a generic, winged, circular-body, single-stage-to-orbit spacecraft configuration. The baseline configuration was longitudinally stable and trimmable at almost all Mach numbers from 0.15 to 10.0--with the exception occurring at low supersonic speeds. Landing speed and subsonic-to-hypersonic longitudinal stability and control appear to be within design guidelines. Lateral-directional instabilities found over the entire speed range, however, create a problem area for this configuration. Longitudinal aerodynamic predictions made utilizing the Aerodynamic

Preliminary Analysis System (APAS) were in qualitative, often quantitative agreement with experimental values.
Author
Aerodynamic Characteristics; Spacecraft Configurations; Subsonic Speed; Supersonic Speed; Hypersonics; Single Stage to Orbit Vehicles; Longitudinal Stability; Controllability

19970005058 NASA Langley Research Center, Hampton, VA USA
Dual-Fuel Propulsion in Single-Stage Advanced Manned Launch System Vehicle
Lepsch, Roger A., Jr.; Stanley, Douglas O.; Unal, Resit; Journal of Spacecraft and Rockets; Jun. 1995; Volume 32, Number 3; In English; 29th Joint Propulsion Conference, 28-30 Jun. 1993, Monterey, CA, USA
Report No.(s): NASA-TM-111903; NAS 1.15:111903; AIAA Paper 93-2275; Copyright; Avail: CASI; A03, Hardcopy

As part of the USA Advanced Manned Launch System study to determine a follow-on, or complement, to the Space Shuttle, a reusable single-stage-to-orbit concept utilizing dual-fuel rocket propulsion has been examined. Several dual-fuel propulsion concepts were investigated. These include: a separate-engine concept combining Russian RD-170 kerosene-fueled engines with space shuttle main engine-derivative engines: the kerosene- and hydrogen-fueled Russian RD-701 engine; and a dual-fuel, dual-expander engine. Analysis to determine vehicle weight and size characteristics was performed using conceptual-level design techniques. A response-surface methodology for multidisciplinary design was utilized to optimize the dual-fuel vehicles with respect to several important propulsion-system and vehicle design parameters, in order to achieve minimum empty weight. The tools and methods employed in the analysis process are also summarized. In comparison with a reference hydrogen- fueled single-stage vehicle, results showed that the dual-fuel vehicles were from 10 to 30% lower in empty weight for the same payload capability, with the dual-expander engine types showing the greatest potential.
Author
Engine Design; Design Analysis; Space Shuttle Main Engine; Propulsion System Configurations; Single Stage to Orbit Vehicles; Weight Reduction

19970003025 Maryland Univ., College Park, MD USA
Effect of Aeropropulsive Interactions and Design Sensitivities on Optimal Hypersonic Ascent Trajectories
Lovell, T. Alan; Schmidt, David K.; [1994]; In English
Contract(s)/Grant(s): NAG1-1540
Report No.(s): NASA-CR-202601; NAS 1.26:202601; AIAA Paper 94-3524; Copyright; Avail: CASI; A03, Hardcopy

In this paper, trajectory optimization is used as a tool to better understand the performance characteristics of hypersonic Single-Stage-To-Orbit(SSTO) vehicle exhibiting significant aeropropulsive interactions. The energy state approximation is used to determine a straightforward method of determining the scramjet-powered phase of the mission. The energy state method is then used to generate fuel-optimal trajectory results over this portion of the mission for a vehicle configuration of this class. The fuel-optimal unconstrained trajectory is marked by low-altitude acceleration, while a dynamic pressure-constrained trajectory is seen to ride the dynamic pressure constraint for the entire scramjet mission phase. The significance of aeropropulsive interactions in affecting vehicle performance is also investigated.
Author
Ascent Trajectories; Trajectory Optimization; Single Stage to Orbit Vehicles; Supersonic Combustion Ramjet Engines; Propulsion System Performance; Fuel Consumption

19970001742 North Carolina State Univ., Raleigh, NC USA
System Sensitivity Analysis Applied to the Conceptual Design of a Dual-Fuel Rocket SSTO
Olds, John R.; 1994; In English; 5th Symposium on Multidisciplinary Analysis and Optimization, 7-9 Sep. 1994, Panama City Beach, Fl, USA
Contract(s)/Grant(s): NCC1-185
Report No.(s): NASA-CR-202599; NAS 1.26:202599; AIAA Paper 94-4339; Copyright; Avail: CASI; A03, Hardcopy

This paper reports the results of initial efforts to apply the System Sensitivity Analysis (SSA) optimization method to the conceptual design of a single-stage-to-orbit (SSTO) launch vehicle. SSA is an efficient, calculus-based MDO technique for generating sensitivity derivatives in a highly multidisciplinary design environment. The method has been successfully applied to conceptual aircraft design and has been proven to have advantages over traditional direct optimization methods. The method is applied to the optimization of an advanced, piloted SSTO design similar to vehicles currently being analyzed by NASA as possible replacements for the Space Shuttle. Powered by a derivative of the Russian RD-701 rocket engine, the vehicle employs a combination of hydrocarbon, hydrogen, and oxygen propellants. Three primary disciplines are included in the

design - propulsion, performance, and weights & sizing. A complete, converged vehicle analysis depends on the use of three standalone conceptual analysis computer codes. Efforts to minimize vehicle dry (empty) weight are reported in this paper. The problem consists of six system-level design variables and one system-level constraint. Using SSA in a 'manual' fashion to generate gradient information, six system-level iterations were performed from each of two different starting points. The results showed a good pattern of convergence for both starting points. A discussion of the advantages and disadvantages of the method, possible areas of improvement, and future work is included.
Author
Single Stage to Orbit Vehicles; Multidisciplinary Design Optimization; Systems Analysis; Design Analysis

19970001416 Lockheed Martin Engineering and Sciences Co., Hampton, VA USA
Computational Study of a McDonnell Douglas Single-Stage-to-Orbit Vehicle Concept for Aerodynamic Analysis
Prabhu, Ramadas K.; Sep. 1996; In English
Contract(s)/Grant(s): NAS1-19000; RTOP 242-20-80-02
Report No.(s): NASA-CR-201606; NAS 1.26:201606; No Copyright; Avail: CASI; A03, Hardcopy

This paper presents the results of a computational flow analysis of the McDonnell Douglas single-stage-to-orbit vehicle concept designated as the 24U. This study was made to determine the aerodynamic characteristics of the vehicle with and without body flaps over an angle of attack range of 20-40 deg. Computations were made at a flight Mach number of 20 at 200,000 ft. altitude with equilibrium air, and a Mach number of 6 with CF4 gas. The software package FELISA (Finite Element Langley imperial College Sawansea Ames) was used for all the computations. The FELISA software consists of unstructured surface and volume grid generators, and inviscid flow solvers with (1) perfect gas option for subsonic, transonic, and low supersonic speeds, and (2) perfect gas, equilibrium air, and CF4 options for hypersonic speeds. The hypersonic flow solvers with equilibrium air and CF4 options were used in the present studies. Results are compared with other computational results and hypersonic CF4 tunnel test data.
Author
Unstructured Grids (Mathematics); Single Stage to Orbit Vehicles; Inviscid Flow; Hypersonic Speed; Hypersonic Flow; Finite Element Method; Computational Grids; Applications Programs (Computers); Aerodynamic Characteristics

19970001260 NASA, Huntsville, AL USA
Acoustic Emission Monitoring of the DC-XA Composite Liquid Hydrogen Tank During Structural Testing
Wilkerson, C.; Oct. 1996; In English
Report No.(s): NASA-TM-108520; NAS 1.15:108520; No Copyright; Avail: CASI; A03, Hardcopy

The results of acoustic emission (AE) monitoring of the DC-XA composite liquid hydrogen tank are presented in this report. The tank was subjected to pressurization, tensile, and compressive loads at ambient temperatures and also while full of liquid nitrogen. The tank was also pressurized with liquid hydrogen. AE was used to monitor the tank for signs of structural defects developing during the test.
Author
Acoustic Emission; Liquid Hydrogen; Fuel Tanks; Nondestructive Tests; Temperature Effects; Structural Members

19960055049 NASA Langley Research Center, Hampton, VA USA
The 1995 NASA High-Speed Research Program Sonic Boom Workshop, Volume 1
Baize, Daniel G., Editor; 1995 NASA High-Speed Research Program Sonic Boom Workshop. Volume 1; Jul. 1996; In English, 12-13 Sep. 1995, Hampton, VA, USA
Contract(s)/Grant(s): RTOP 537-07-21-21
Report No.(s): NASA-CP-3335-Vol-1; L-17572-Vol-1; NAS 1.55:3335-Vol-1; No Copyright; Avail: CASI; A17, Hardcopy

The High-Speed Research Program and NASA Langley Research Center sponsored the NASA High-Speed Research Program Sonic Boom Workshop on September 12-13, 1995. The workshop was designed to bring together NASAs scientists and engineers and their counterparts in industry, other Government agencies, and academia working together in the sonic boom element of NASAs High-Speed Research Program. Specific objectives of this workshop were to (1) report the progress and status of research in sonic boom propagation, acceptability, and design; (2) promote and disseminate this technology within the appropriate technical communities; (3) help promote synergy among the scientists working in the Program; and (4) identify technology pacing the development of viable reduced-boom High-Speed Civil Transport concepts. The Workshop included these sessions: Session 1 - Sonic Boom Propagation (Theoretical); Session 2 - Sonic Boom Propagation

(Experimental); and Session 3 - Acceptability Studies - Human and Animal.
Author
Aerospace Planes; Sonic Booms; High Speed; Sound Propagation

19960053992 NASA Langley Research Center, Hampton, VA USA
Reusable Launch Vehicle Technology Program
Freeman, Delma C., Jr.; Talay, Theodore A.; Austin, R. Eugene; 1996; In English; 47th International Astronautical Congress, 7-11 Oct. 1996, Beijing, China
Report No.(s): NASA-TM-110473; NAS 1.15:110473; IAF-96-V.4.01; Copyright; Avail: CASI; A03, Hardcopy

Industry/NASA Reusable Launch Vehicle (RLV) Technology Program efforts are underway to design, test, and develop technologies and concepts for viable commercial launch systems that also satisfy national needs at acceptable recurring costs. Significant progress has been made in understanding the technical challenges of fully reusable launch systems and the accompanying management and operational approaches for achieving a low-cost program. This paper reviews the current status of the Reusable Launch Vehicle Technology Program including the DC-XA, X-33 and X-34 flight systems and associated technology programs. It addresses the specific technologies being tested that address the technical and operability challenges of reusable launch systems including reusable cryogenic propellant tanks, composite structures, thermal protection systems, improved propulsion, and subsystem operability enhancements. The recently concluded DC-XA test program demonstrated some of these technologies in ground and flight tests. Contracts were awarded recently for both the X-33 and X-34 flight demonstrator systems. The Orbital Sciences Corporation X-34 flight test vehicle will demonstrate an air-launched reusable vehicle capable of flight to speeds of Mach 8. The Lockheed-Martin X-33 flight test vehicle will expand the test envelope for critical technologies to flight speeds of Mach 15. A propulsion program to test the X-33 linear aerospike rocket engine using a NASA SR-71 high speed aircraft as a test bed is also discussed. The paper also describes the management and operational approaches that address the challenge of new cost-effective, reusable launch vehicle systems.
Author
Propulsion System Performance; Technologies; Single Stage to Orbit Vehicles; Composite Structures; Cryogenic Fluid Storage; Propellant Tanks

19960049683 Lockheed Martin Corp., Hampton, VA USA
A study of facilities and fixtures for testing of a high speed civil transport wing component
Cerro, J. A.; Vause, R. F.; Bowman, L. M.; Jensen, J. K.; Martin, C. J., Jr.; Stockwell, A. E.; Waters, W. A., Jr.; Jul. 1996; In English
Contract(s)/Grant(s): NAS1-19000; RTOP 537-06-34
Report No.(s): NASA-CR-198352; NAS 1.26:198352; No Copyright; Avail: CASI; A04, Hardcopy

A study was performed to determine the feasibility of testing a large-scale High Speed Civil Transport wing component in the Structures and Materials Testing Laboratory in Building 1148 at NASA Langley Research Center. The report includes a survey of the electrical and hydraulic resources and identifies the backing structure and floor hard points which would be available for reacting the test loads. The backing structure analysis uses a new finite element model of the floor and backstop support system in the Structures Laboratory. Information on the data acquisition system and the thermal power requirements is also presented. The study identified the hardware that would be required to test a typical component, including the number and arrangement of hydraulic actuators required to simulate expected flight loads. Load introduction and reaction structure concepts were analyzed to investigate the effects of experimentally induced boundary conditions.
Author
X-30 Vehicle; Wings; Aerodynamic Loads; Test Chambers; Test Facilities; Finite Element Method; Loads (Forces); Supersonic Transports; Support Systems

19960047476 Boeing Commercial Airplane Co., Seattle, WA USA
Materials Research for High Speed Civil Transport and Generic Hypersonics-Metals Durability
Schulz, Paul; Hoffman, Daniel; Apr. 1996; In English
Contract(s)/Grant(s): NAS1-20013; RTOP 537-06-20-05
Report No.(s): NASA-CR-198326; NAS 1.26:198326; No Copyright; Avail: CASI; A05, Hardcopy

This report covers a portion of an ongoing investigation of the durability of titanium alloys for the High Speed Civil Transport (HSCT). Candidate alloys need to possess an acceptable combination of properties including strength and toughness as well as fatigue and corrosion resistance when subjected to the HSCT operational environment. These materials must also

be capable of being processed into required product forms while maintaining their properties. Processing operations being considered for this airplane include forming, welding, adhesive bonding, and superplastic forming with or without diffusion bonding. This program was designed to develop the material properties database required to lower the risk of using advanced titanium alloys on the HSCT.
Author
Aerospace Planes; Hypersonic Aircraft; Titanium Alloys; Supersonic Transports; Metal Bonding; Fatigue (Materials); Fracture Strength; Mechanical Properties; Durability

19960047140 NASA Langley Research Center, Hampton, VA USA
Subsonic Aerodynamic Characteristics of a Circular Body Earth-to-Orbit Vehicle
Lepsch, Roger A., Jr.; Ware, George M.; MacConochie, Ian O.; Jul. 1996; In English
Contract(s)/Grant(s): RTOP 242-20-08-01
Report No.(s): NASA-TM-4726; L-17429; NAS 1.15:4726; No Copyright; Avail: CASI; A03, Hardcopy

A test of a generic reusable earth-to-orbit transport was conducted in the 7- by 10-Foot high-speed tunnel at the Langley Research Center at Mach number 0.3. The model had a body with a circular cross section and a thick clipped delta wing as the major lifting surface. For directional control, three different vertical fin arrangements were investigated: a conventional aft-mounted center vertical fin, wingtip fins, and a nose-mounted vertical fin. The configuration was longitudinally stable about the estimated center-of-gravity position of 0.72 body length and had sufficient pitch-control authority for stable trim over a wide range of angle of attack, regardless of fin arrangement. The maximum trimmed lift/drag ratio for the aft center-fin configuration was less than 5, whereas the other configurations had values of above 6. The aft center-fin configuration was directionally stable for all angles of attack tested. The wingtip and nose fins were not intended to produce directional stability but to be active controllers for artificial stabilization. Small rolling-moment values resulted from yaw control of the nose fin. Large adverse rolling-moment increments resulted from tip-fin controller deflection above 13 deg angle of attack. Flow visualization indicated that the adverse rolling-moment increments were probably caused by the influence of the deflected tip-fin controller on wing flow separation.
Author
Wind Tunnel Tests; Delta Wings; Angle of Attack; Rolling Moments; Flow Visualization; Subsonic Flow; Wind Tunnel Models; Fins; Satellite Control; Directional Control; Lift Drag Ratio

19960047075 Vanderbilt Univ., Nashville, TN USA
Dynamics and Control of Orbiting Space Structures NASA Advanced Design Program (ADP)
Cruse, T. A.; Aug. 20, 1996; In English
Contract(s)/Grant(s): NAGw-4252
Report No.(s): NASA-CR-202061; NAS 1.26:202061; Rept-96-08-02; No Copyright; Avail: CASI; A03, Hardcopy

The report summarizes the advanced design program in the mechanical engineering department at Vanderbilt University for the academic years 1994-1995 and 1995-1996. Approximately 100 students participated in the two years of the subject grant funding. The NASA-oriented design projects that were selected included lightweight hydrogen propellant tank for the reusable launch vehicle, a thermal barrier coating test facility, a piezoelectric motor for space antenna control, and a lightweight satellite for automated materials processing. The NASA supported advanced design program (ADP) has been a success and a number of graduates are working in aerospace and are doing design.
Author
Spacecraft Structures; Reusable Launch Vehicles; Spacecraft Design

19960045833 NASA Marshall Space Flight Center, Huntsville, AL USA
The Reusable Launch Vehicle Technology Program and the X-33 Advanced Technology Demonstrator
Cook, Stephen A.; Apr. 04, 1995; In English; 6th AIAA Aerospace Plane and Hypersonics Technology Conference, 4 Apr. 1995, Chattanooga, TN, USA
Report No.(s): NASA-TM-111868; NAS 1.15:111868; No Copyright; Avail: CASI; A02, Hardcopy

The goal of the Reusable Launch Vehicle (RLV) technology program is formulated, and the primary objectives of RLV are listed. RLV technology program implementation phases are outlined. X-33 advanced technology demonstrator is described. Program management is addressed.
CASI
Technologies; Single Stage to Orbit Vehicles; X-33 Reusable Launch Vehicle; Advanced Launch System (Sts)

19960045205 NASA Langley Research Center, Hampton, VA USA

Design and Analysis of a Two-Stage-to-Orbit Airbreathing Hypersonic Vehicle Concept

Lockwood, Mary Kae; Hunt, James L.; Kabis, Hanee; Moses, Paul; Pao, Jenn-Louh; Yarrington, Phillip; Collier, Craig; Jul. 03, 1996; In English; 32nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 1-3 Jul. 1996, Lake Buena Vista, FL, USA; Original contains color illustrations

Report No.(s): NASA-CR-201784; NAS 1.26:201784; AIAA Paper 96-2890; Copyright; Avail: CASI; A03, Hardcopy

An initial design of a second generation two-stage-to-orbit vehicle, with an airbreathing first stage, capable of delivering a 2000 lb payload to orbit has been completed. Two low speed propulsion systems were considered for the first stage vehicle for Mach 0 to 3 operation, an ejector ramjet and an advanced air turbo-ramjet engine. A dual mode ramjet/scramjet was used above Mach 3. The airframe structure is composed of cold integral-tank graphite-epoxy stiffened panels with an adhesively-bonded thermal protection system. Active cooling through aluminum heat exchanger panels is used in the engine. The near optimal staging Mach number to minimize vehicle dry weight and take-off gross weight for this mission was determined to be Mach 8. Dry weights range from 67,000 to 69,000 lbs and take-off gross weights range from 119,000 to 131,000 lbs, depending on the low speed propulsion system. Three-stage-to-orbit configurations were also considered, with the first stage being a platform for a Mach .8 launch, and the second and third stages being similar to the first and second stages of the two-stage-to-orbit vehicle. A 10,000 lb reduction in TOGW resulted for the combined 2nd and 3rd stages of the three-stage-to-orbit vehicle, as compared to the two-stage-to-orbit vehicle. This relatively small reduction probably does not warrant the added complexity for the air launch in this case.

Author

Reusable Launch Vehicles; Air Breathing Engines; Graphite-Epoxy Composites; Hypersonic Vehicles; Propulsion System Configurations; Supersonic Combustion Ramjet Engines

19960042873 Lockheed Martin Missiles and Space, Huntsville, AL USA

Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract, Volume 2, Technical Results

Jul. 1995; In English

Contract(s)/Grant(s): NAS8-39208

Report No.(s): NASA-CR-201127-Vol-2; NAS 1.26:201127-Vol-2; LMSC-P038190-Vol-2; No Copyright; Avail: CASI; A99, Hardcopy

The sections in this report include: Single Stage to Orbit (SSTO) Design Ground-rules; Operations Issues and Lessons Learned; Vertical-Takeoff/Landing Versus Vertical-Takeoff/Horizontal-Landing; SSTO Design Results; SSTO Simulation Results; SSTO Assessment Results; SSTO Sizing Tool User's Guide; SSto Turnaround Assessment Report; Ground Operations Assessment First Year Executive Summary; Health Management System Definition Study; Major TA-2 Presentations; First Lunar Outpost Heavy Lift Launch Vehicle Design and Assessment; and the section, Russian Propulsion Technology Assessment Reports.

CASI

Heavy Lift Launch Vehicles; Single Stage to Orbit Vehicles; Propulsion; Vertical Takeoff; Vertical Landing; Technology Assessment

19960041001 Lockheed Martin Astronautics, Huntsville, AL USA

Advanced transportation system studies. Technical area 2: Heavy lift launch vehicle development, Volume 2, Technical Results

Jul. 1995; In English

Contract(s)/Grant(s): NAS8-39208

Report No.(s): NASA-CR-201126; NAS 1.26:201126; LMSC-P038190; No Copyright; Avail: CASI; A17, Hardcopy

Sections 10 to 13 of the Advanced Transportation System Studies final report are included in this volume. Section 10 contains a copy of an executive summary that was prepared by Lockheed Space Operations Company (LSOC) to document their support to the TA-2 contract during the first-year period of performance of the contract, May 1992 through May 1993. LSOC participated on the TA-2 contract as part of the concurrent engineering launch system definition team, and provided outstanding heavy lift launch vehicle (HLLV) ground operations requirements and concept assessments for Lockheed Missiles and Space Company (LMSC) through an intercompany work transfer as well as providing specific HLLV ground operations assessments at the direction of NASA KSC through KSC funding that was routed to the TA-2 contract. Section 11 contains a copy of a vehicle-independent, launch system health management requirements assessment. The purpose of the assessment was to define both health management requirements and the associated interfaces between a generic advanced transportation

system launch vehicle and all related elements of the entire transportation system, including the ground segment. Section 12 presents the major TA-2 presentations provided to summarize the significant results and conclusions that were developed over the course of the contract. Finally, Section 13 presents the design and assessment report on the first lunar outpost heavy lift launch vehicle.
CASI
Heavy Lift Launch Vehicles; Single Stage to Orbit Vehicles; Launch Vehicle Configurations; Advanced Launch System (Sts); Recoverable Spacecraft; Spacecraft Design

19960040942 Georgia Inst. of Tech., Atlanta, GA USA
Options for flight testing rocket-based combined-cycle (RBCC) engines
Olds, John; Jul. 03, 1996; In English; 32nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 1-3 Jul. 1996, Atlanta, GA, USA
Contract(s)/Grant(s): NAG8-1202
Report No.(s): NASA-CR-201897; NAS 1.26:201897; AIAA Paper 96-2688; Copyright; Avail: CASI; A03, Hardcopy

While NASA's current next-generation launch vehicle research has largely focused on advanced all-rocket single-stage-to-orbit vehicles (i.e. the X-33 and it's RLV operational follow-on), some attention is being given to advanced propulsion concepts suitable for 'next-generation-and-a-half' vehicles. Rocket-based combined-cycle (RBCC) engines combining rocket and airbreathing elements are one candidate concept. Preliminary RBCC engine development was undertaken by the USA in the 1960's. However, additional ground and flight research is required to bring the engine to technological maturity. This paper presents two options for flight testing early versions of the RBCC ejector scramjet engine. The first option mounts a single RBCC engine module to the X-34 air-launched technology testbed for test flights up to about Mach 6.4. The second option links RBCC engine testing to the simultaneous development of a small-payload (220 lb.) two-stage-to-orbit operational vehicle in the Bantam payload class. This launcher/testbed concept has been dubbed the W vehicle. The W vehicle can also serve as an early ejector ramjet RBCC launcher (albeit at a lower payload). To complement current RBCC ground testing efforts, both flight test engines will use earth-storable propellants for their RBCC rocket primaries and hydrocarbon fuel for their airbreathing modes. Performance and vehicle sizing results are presented for both options.
Author
Engine Tests; Flight Tests; Supersonic Combustion Ramjet Engines; Rocket Engines; X-34 Reusable Launch Vehicle

19960028975 Lockheed Martin Missiles and Space, Huntsville, AL USA
Advanced transportation system studies technical area 2 (TA-2): Heavy lift launch vehicle development, volume 3, Program Cost estimates
McCurry, J. B.; Jul. 1995; In English
Contract(s)/Grant(s): NAS8-39208
Report No.(s): NASA-CR-201128; NAS 1.26:201128; LMSC P038190; No Copyright; Avail: CASI; A04, Hardcopy

The purpose of the TA-2 contract was to provide advanced launch vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements. Contracted analysis activities included vehicle sizing and performance analysis, subsystem concept definition, propulsion subsystem definition (foreign and domestic), ground operations and facilities analysis, and life cycle cost estimation. The basic period of performance of the TA-2 contract was from May 1992 through May 1993. No-cost extensions were exercised on the contract from June 1993 through July 1995. This document is part of the final report for the TA-2 contract. The final report consists of three volumes: Volume 1 is the Executive Summary, Volume 2 is Technical Results, and Volume 3 is Program Cost Estimates. The document-at-hand, Volume 3, provides a work breakdown structure dictionary, user's guide for the parametric life cycle cost estimation tool, and final report developed by ECON, Inc., under subcontract to Lockheed Martin on TA-2 for the analysis of heavy lift launch vehicle concepts.
Derived from text
Space Transportation System; Launch Vehicles; Life Cycle Costs; Spacecraft Design; Cost Analysis

19960028972 Lockheed Martin Missiles and Space, Huntsville, AL USA
Advanced transportation system studies technical area 2(TA-2): Heavy lift launch vehicle development, volume 1, Executive summary
McCurry, J.; Jul. 1995; In English
Contract(s)/Grant(s): NAS8-39208
Report No.(s): NASA-CR-201124; NAS 1.26:201124; LMSC PC308190; No Copyright; Avail: CASI; A04, Hardcopy

The purpose of the TA-2 contract was to provide advanced launch vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements. Contracted analysis activities included vehicle sizing and performance analysis, subsystem concept definition, propulsion subsystem definition (foreign and domestic), ground operations and facilities analysis, and life cycle cost estimation. This document is part of the final report for the TA-2 contract. The final report consists of three volumes: Volume 1 is the Executive Summary, Volume 2 is Technical Results, and Volume 3 is Program Cost Estimates. The document-at-hand, Volume 1, provides a summary description of the technical activities that were performed over the entire contract duration, covering three distinct launch vehicle definition activities: heavy-lift (300,000 pounds injected mass to low Earth orbit) launch vehicles for the First Lunar Outpost (FLO), medium-lift (50,000-80,000 pounds injected mass to low Earth orbit) launch vehicles, and single-stage-to-orbit (SSTO) launch vehicles (25,000 pounds injected mass to a Space Station orbit).
Derived from text
Launch Vehicles; Life Cycle Costs; Space Transportation; Low Earth Orbits; Space Stations; Heavy Lift Launch Vehicles

19960027977 Lockheed Martin Missiles and Space, Huntsville, AL USA
Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract, Volume 2, Technical Results
Jul. 1995; In English
Contract(s)/Grant(s): NAS8-39208
Report No.(s): NASA-CR-201125-Vol-2; NAS 1.26:201125-Vol-2; LMSC-P038190; No Copyright; Avail: CASI; A17, Hardcopy

The purpose of the Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development contract was to provide advanced launch vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements. Contracted analysis activities included vehicle sizing and performance analysis, subsystem concept definition, propulsion subsystem definition (foreign and domestic), ground operations and facilities analysis, and life cycle cost estimation. This document is Volume 2 of the final report for the contract. It provides documentation of selected technical results from various TA-2 analysis activities, including a detailed narrative description of the SSTO concept assessment results, a user's guide for the associated SSTO sizing tools, an SSTO turnaround assessment report, an executive summary of the ground operations assessments performed during the first year of the contract, a configuration-independent vehicle health management system requirements report, a copy of all major TA-2 contract presentations, a copy of the FLO launch vehicle final report, and references to Pratt & Whitney's TA-2 sponsored final reports regarding the identification of Russian main propulsion technologies.
Derived from text
Heavy Lift Launch Vehicles; Single Stage to Orbit Vehicles; Space Transportation; Spacecraft Design

19960027549 Lockheed Martin Astronautics, Huntsville, AL USA
Transportation Systems Analyses (TSA): Highly Reusable Space Transportation (HRST). A preliminary programmatic assessment
Duffey, Jack; Lowrey, Alan; May 1996; In English
Contract(s)/Grant(s): NAS8-39209
Report No.(s): NASA-CR-201069; NAS 1.26:201069; No Copyright; Avail: CASI; A03, Hardcopy

This report overviews the strategic implications of the Highly Reusable Space Transportation (HRST) program. The analysis postulates the anticipated HRST market (window is 2006-30, with a 2015 focus). Next the analysis speculates on market 'price of entry' for several potential markets. HRST is envisioned as a NASA overlay to either the STS modernization or the on-going RLV initiative. Three NASA options are reviewed. An example HRST program (MagLifter + RBCC RLV) is assessed in terms of financial/political issues. The merits of HRST-vs-RLV are briefly examined. Finally, a Small Launch Vehicle (SLV) HRST application is reviewed.
Author
Space Transportation System; Market Research; Reusable Launch Vehicles; Reusable Spacecraft

19960021025 NASA Marshall Space Flight Center, Huntsville,AL USA, American Inst. of Aeronautics and Astronautics, Washington, DC USA
Rocket-Induced Magnetohydrodynamic Ejector: A Single-Stage-to-Orbit Advanced Propulsion Concept
Cole, John; Campbell, Jonathan; Robertson, Anthony; Sep. 1995; In English; AIAA 1995 Space Programs and Technologies Conference, 26-28 Sep. 1995, Huntsville, AL, USA
Report No.(s): NASA-TM-111435; NAS 1.15:111435; AIAA Paper 95-4079; Copyright; Avail: CASI; A03, Hardcopy

During the atmospheric boost phase of a rocket trajectory, magnetohydrodynamic (MHD) principles can be utilized to augment the thrust by several hundred percent without the input of additional energy. The concept is an MHD implementation of a thermodynamic ejector. Some ejector history is described and some test data showing the impressive thrust augmentation capabilities of thermodynamic ejectors are provided. A momentum and energy balance is used to derive the equations to predict the MHD ejector performance. Results of these equations are compared with the test data and then applied to a specific performance example. The rocket-induced MHD ejector (RIME) engine is described and a status of the technology and availability of the engine components is provided. A top level vehicle sizing analysis is performed by scaling existing MHD designs to the required flight vehicle levels. The vehicle can achieve orbit using conservative technology. Modest improvements are suggested using recently developed technologies, such as superconducting magnets, which can improve predicted performance well beyond those expected for current single-stage-to-orbit (SSTO) designs.
Author
Magnetohydrodynamics; Single Stage to Orbit Vehicles; Rocket Engines; Ejectors; Thrust Augmentation

19960020534 NASA, Hampton, VA USA
Development of a Pressure Box to Evaluate Reusable-Launch-Vehicle Cryogenic-Tank Panels
Ambur, Damodar R.; Sikora, Joseph; Maguire, James F.; Winn, Peter M.; Apr. 1996; In English; AIAA/ASME/ASCE/AHS/ASC 37th Structures, Structural Dynamics, and Materials Conference, 15-17 Apr. 1996, Salt Lake City, UT, USA
Report No.(s): NASA-TM-111452; NAS 1.15:111452; AIAA Paper 96-1640; Copyright; Avail: CASI; A03, Hardcopy

A cryogenic pressure-box test machine has been designed and is being developed to test full-scale reusable launch vehicle cryogenic-tank panels. This machine is equipped with an internal pressurization system, a cryogenic cooling system, and a heating system to simulate the mechanical and thermal loading conditions that are representative of a reusable launch vehicle mission profile. The cryogenic cooling system uses liquid helium and liquid nitrogen to simulate liquid hydrogen and liquid oxygen tank internal temperatures. A quartz lamp heating system is used for heating the external surface of the test panels to simulate cryogenic-tank external surface temperatures during re-entry of the launch vehicle. The pressurization system uses gaseous helium and is designed to be controlled independently of the cooling system. The tensile loads in the axial direction of the test panel are simulated by means of hydraulic actuators and a load control system. The hoop loads in the test panel are reacted by load-calibrated turnbuckles attached to the skin and frame elements of the test panel. The load distribution in the skin and frames can be adjusted to correspond to the tank structure by using these turnbuckles. The seal between the test panel and the cryogenic pressure box is made from a reinforced Teflon material which can withstand pressures greater than 52 psig at cryogenic temperatures. Analytical results and tests on prototype test components indicate that most of the cryogenic-tank loading conditions that occur in flight can be simulated in the cryogenic pressure-box test machine.
Author
Reusable Launch Vehicles; Cryogenic Fluid Storage; Fuel Tanks; Storage Tanks; Panels; Pressure Vessels; Load Tests; Thermal Stresses; Tensile Stress; Seals (Stoppers)

19960017710 Rockwell International Corp., Canoga Park, CA USA
Alternate Propulsion Subsystem Concepts Tripropellant Comparison Study
Levack, Daniel; Oct. 1995; In English
Contract(s)/Grant(s): NAS8-39210
Report No.(s): NASA-CR-200703; NAS 1.26:200703; 1-1-PP-02147; No Copyright; Avail: CASI; A10, Hardcopy

A study was conducted under MSFC contract NAS8-39210 to compare tripropellant and bipropellant engine configurations for the SSTO mission. The objective was to produce an ‘apples-to-apples’ comparison to isolate the effects of design implementation, designing company, year of design, or technologies included from the basic tripropellant/bipropellant comparison. Consequently, identical technologies were included (e.g., jet pumps) and the same design groundrules and practices were used. Engine power cycles were examined as were turbomachinery/preburner arrangements for each cycle. The bipropellant approach and two tripropellant approaches were separately optimized in terms of operating parameters: exit pressures, mixture ratios, thrust splits, etc. This briefing presents the results of the study including engine weights for both tripropellant and bipropellant engines; dry vehicle weight performance for a range of engine chamber pressures; discusses the basis for the results; examines vehicle performance due to engine cycles and the margin characteristics of various cycles; and identifies technologies with significant payoffs for this application.
Author
Propulsion System Performance; Liquid Propellant Rocket Engines; Propulsion System Configurations; Design Analysis

19960017671 NASA Langley Research Center, Hampton, VA USA, Applied Engineering Technologies Ltd., Westborough, MA USA

Development of a Pressure Box to Evaluate Reusable-Launch-Vehicle Cryogenic-Tank Panels

Ambur, Damodar R.; Sikora, Joseph; Maguire, James F.; Winn, Peter M.; 1996; In English; AIAA/ASME/ASCE/AHS/ASC 37th Structures, Structural Dynamics, and Materials Conference, 15-17 Apr. 1996, Salt Lake City, UT, USA

Report No.(s): NASA-TM-111406; NAS 1.15:111406; AIAA Paper 96-1640; No Copyright; Avail: CASI; A03, Hardcopy

A cryogenic pressure-box test machine has been designed and is being developed to test full-scale reusable-launch-vehicle cryogenic-tank panels. This machine is equipped with an internal pressurization system, a cryogenic cooling system, and a heating system to simulate the mechanical and thermal loading conditions that are representative of a reusable-launch-vehicle mission profile. The cryogenic cooling system uses liquid helium and liquid nitrogen to simulate liquid hydrogen and liquid oxygen tank internal temperatures. A quartz lamp heating system is used for heating the external surface of the test panels to simulate cryogenic-tank external surface temperatures during re-entry of the launch vehicle. The pressurization system uses gaseous helium and is designed to be controlled independently of the cooling system. The tensile loads in the axial direction of the test panel are simulated by means of hydraulic actuators and a load control system. The hoop loads in the test panel are reacted by load-calibrated turnbuckles attached to the skin and frame elements of the test panel. The load distribution in the skin and frames can be adjusted to correspond to the tank structure by using these turnbuckles. The seal between the test panel and the cryogenic pressure box is made from a reinforced Teflon material which can withstand pressures greater than 52 psig at cryogenic temperatures. Analytical results and tests on prototype test components indicate that most of the cryogenic-tank loading conditions that occur in flight can be simulated in the cryogenic pressure-box test machine.

Author (AIAA)

Reusable Launch Vehicles; Storage Tanks; Cryogenic Fluid Storage; Load Tests; Panels; Teflon (Trademark)

19960015940 NASA Ames Research Center, Moffett Field, CA USA

Surface Catalytic Efficiency of Advanced Carbon Carbon Candidate Thermal Protection Materials for SSTO Vehicles

Stewart, David A.; Feb. 01, 1996; In English

Contract(s)/Grant(s): RTOP 242-20-01

Report No.(s): NASA-TM-110383; A-961133; NAS 1.15:110383; No Copyright; Avail: CASI; A03, Hardcopy

The catalytic efficiency (atom recombination coefficients) for advanced ceramic thermal protection systems was calculated using arc-jet data. Coefficients for both oxygen and nitrogen atom recombination on the surfaces of these systems were obtained to temperatures of 1650 K. Optical and chemical stability of the candidate systems to the high energy hypersonic flow was also demonstrated during these tests.

Author

Thermal Protection; Recombination Coefficient; Oxygen Atoms; Nitrogen Atoms; Carbon-Carbon Composites; Surface Properties; Catalytic Activity

19960015880 Smith Advanced Technology, Inc., Huntsville,AL USA

Advanced Propulsion System Test and Tell Capability

Horn, Albert E.; Jan. 1996; In English

Contract(s)/Grant(s): NAS3-27580; RTOP 244-01-00

Report No.(s): NASA-CR-198415; E-9979; NAS 1.26:198415; No Copyright; Avail: CASI; A03, Hardcopy

The feasibility of a 'test and tell' capability that performs real-time (or near real-time) diagnostic and prognostic health monitoring for liquid rocket propulsion systems, utilizing a synthesized mathematical modeling expert system technology, was investigated. Such a system would have the potential for reducing operational costs of a launch vehicle propulsion system by reducing the post-firing analysis time and personnel resources needed to determine component health and subsystem replacement requirements. A candidate architecture was defined consisting of propulsion system and subsystem models executing on a flight-avionics based platform. Related work was analyzed, supporting technologies such as NASA Lewis Research Center's (LeRC's) 'feature extraction' were studied, and the proposed commercial modeling package, Abductive Information Models (AIM), was investigated. The space shuttle main engine (SSME) was selected as a representative propulsion system for development of actual models. Example line replaceable units (LRU's) and their appropriate sensor measurements were defined. A sample SSME subsystem, the high pressure fuel turbopump (HPFTP), was selected, models developed, and abductive polynomials were synthesized with the AIM tool using training data from actual SSME test firings provided by NASA LeRC. The models were then run with additional test data inputs representing both nominal data and off-nominal data. Evaluation of the results led to the conclusion that the 'test and tell' (and eventually 'land and tell' for

reusable launch vehicles) capability was in fact feasible, and could be developed into a commercially-viable product.
Derived from text
Expert Systems; Systems Health Monitoring; Reusable Launch Vehicles; Failure Analysis; Liquid Propellant Rocket Engines; Mathematical Models; Test Firing

19960014875 Dayton Univ., OH USA
O/S analysis of conceptual space vehicles, Part 1
Ebeling, Charles E.; Dec. 31, 1995; In English
Contract(s)/Grant(s): NAG1-1327
Report No.(s): NASA-CR-199465; NAS 1.26:199465; No Copyright; Avail: CASI; A05, Hardcopy

The application of recently developed computer models in determining operational capabilities and support requirements during the conceptual design of proposed space systems is discussed. The models used are the reliability and maintainability (R&M) model, the maintenance simulation model, and the operations and support (O&S) cost model. In the process of applying these models, the R&M and O&S cost models were updated. The more significant enhancements include (1) improved R&M equations for the tank subsystems, (2) the ability to allocate schedule maintenance by subsystem, (3) redefined spares calculations, (4) computing a weighted average of the working days and mission days per month, (5) the use of a position manning factor, and (6) the incorporation into the O&S model of new formulas for computing depot and organizational recurring and nonrecurring training costs and documentation costs, and depot support equipment costs. The case study used is based upon a winged, single-stage, vertical-takeoff vehicle (SSV) designed to deliver to the Space Station Freedom (SSF) a 25,000 lb payload including passengers without a crew.
CASI
Cost Analysis; Maintainability; Spacecraft Reliability; Support Systems; Spacecraft Design; Operations Research; Reliability Analysis; Life Cycle Costs; Spacecraft Maintenance

19960014829 NASA Marshall Space Flight Center, Huntsville, AL USA
Design of launch systems using continuous improvement process
Brown, Richard W.; Sep. 28, 1995
Report No.(s): NASA-TM-111266; NAS 1.15:111266; Copyright; Avail: CASI; A03, Hardcopy

The purpose of this paper is to identify a systematic process for improving ground operations for future launch systems. This approach is based on the Total Quality Management (TQM) continuous improvement process. While the continuous improvement process is normally identified with making incremental changes to an existing system, it can be used on new systems if they use past experience as a knowledge base. In the case of the Reusable Launch Vehicle (RLV), the Space Shuttle operations provide many lessons. The TQM methodology used for this paper will be borrowed from the USA Air Force 'Quality Air Force' Program. There is a general overview of the continuous improvement process, with concentration on the formulation phase. During this phase critical analyses are conducted to determine the strategy and goals for the remaining development process. These analyses include analyzing the mission from the customers point of view, developing an operations concept for the future, assessing current capabilities and determining the gap to be closed between current capabilities and future needs and requirements. A brief analyses of the RLV, relative to the Space Shuttle, will be used to illustrate the concept. Using the continuous improvement design concept has many advantages. These include a customer oriented process which will develop a more marketable product and a better integration of operations and systems during the design phase. But, the use of TQM techniques will require changes, including more discipline in the design process and more emphasis on data gathering for operational systems. The benefits will far outweigh the additional effort.
Author
Total Quality Management; Launching; Ground Operational Support System

19960014828 NASA Langley Research Center, Hampton, VA USA
Optimal technology investment strategies for a reusable launch vehicle
Moore, A. A.; Braun, R. D.; Powell, R. W.; Sep. 28, 1995
Report No.(s): NASA-TM-111267; AIAA Paper 95-3610; NAS 1.15:111267; Copyright; Avail: CASI; A02, Hardcopy

Within the present budgetary environment, developing the technology that leads to an operationally efficient space transportation system with the required performance is a challenge. The present research focuses on a methodology to determine high payoff technology investment strategies. Research has been conducted at Langley Research Center in which design codes for the conceptual analysis of space transportation systems have been integrated in a multidisciplinary design

optimization approach. The current study integrates trajectory, propulsion, weights and sizing, and cost disciplines where the effect of technology maturation on the development cost of a single stage to orbit reusable launch vehicle is examined. Results show that the technology investment prior to full-scale development has a significant economic payoff. The design optimization process is used to determine strategic allocations of limited technology funding to maximize the economic payoff.
Author
Multidisciplinary Design Optimization; Recoverable Launch Vehicles; Single Stage to Orbit Vehicles; Technology Assessment; Technology Utilization; NASA Space Programs; Applications Programs (Computers); Computer Aided Design; Design Analysis

19960013899 National Academy of Sciences - National Research Council, Washington, DC USA
Reusable launch vehicle: Technology development and test program
Jan. 01, 1995
Report No.(s): NASA-CR-200228; LC-95-73106; NAS 1.26:200228; NAS 1.26:200228; Copyright; Avail: CASI; A05, Hardcopy

The National Aeronautics and Space Administration (NASA) requested that the National Research Council (NRC) assess the Reusable Launch Vehicle (RLV) technology development and test programs in the most critical component technologies. At a time when discretionary government spending is under close scrutiny, the RLV program is designed to reduce the cost of access to space through a combination of robust vehicles and a streamlined infrastructure. Routine access to space has obvious benefits for space science, national security, commercial technologies, and the further exploration of space. Because of technological challenges, knowledgeable people disagree about the feasibility of a single-stage-to-orbit (SSTO) vehicle. The purpose of the RLV program proposed by NASA and industry contractors is to investigate the status of existing technology and to identify and advance key technology areas required for development and validation of an SSTO vehicle. This report does not address the feasibility of an SSTO vehicle, nor does it revisit the roles and responsibilities assigned to NASA by the National Transportation Policy. Instead, the report sets forth the NRC committee's findings and recommendations regarding the RLV technology development and test program in the critical areas of propulsion, a reusable cryogenic tank system (RCTS), primary vehicle structure, and a thermal protection system (TPS).
Derived from text
Reusable Launch Vehicles; NASA Programs; Cryogenic Fluid Storage; Thermal Protection; Composite Structures; Propulsion System Configurations; Spacecraft Design; Spacecraft Configurations

19960012497 NASA Langley Research Center, Hampton, VA, USA
Analytical comparison of three stiffened panel concepts
Maloney, Jill M.; Wu, K. Chauncey; Robinson, James C.; Dec 1, 1995; In English
Contract(s)/Grant(s): RTOP 242-10-01-01
Report No.(s): NASA-TM-110165; NAS 1.15:110165; NIPS-96-08490; No Copyright; Avail: CASI; A03, Hardcopy

Three stiffened panel concepts are evaluated to find optimized designs for integral stiffeners in the barrels of Reusable Launch Vehicle fuel tanks. The three panel concepts considered are a T-stiffened panel, a panel with one blade stiffener centered between each pair of T-stiffeners, and a panel with two blade stiffeners equally spaced between each pair of T-stiffeners. The panels are optimized using PASCO for a range of compressive loads, and the computed areal weight for each panel is used to compare the concepts and predict tank weights. The areal weight of the T-stiffened panel with one blade is up to seven-percent lower than the other panel concepts. Two tank construction methods are compared for a representative tank design with three barrels. In the first method, 45-degree circumferential sections of a barrel are each designed to carry the same maximum load in the barrel. In the second method, each barrel section is designed for the maximum load in that section. Representative tanks designed with the first method are over 250 lb heavier than tanks designed using the second method. Optimized panel designs and areal weights are also computed for variation of the nominal panel length and skin thickness.
Author
Compression Loads; Design Analysis; Fuel Tanks; Panels; Reinforced Plates; Reusable Launch Vehicles; Structural Members

19960007723 North Carolina State Univ., Raleigh, NC, USA
Highly reusable space transportation: Approaches for reducing ETO launch costs to $100 - $200 per pound of payload
Olds, John R.; Jan 1, 1995; In English
Report No.(s): NASA-CR-199561; NAS 1.15:199561; NIPS-95-05542; No Copyright; Avail: CASI; A04, Hardcopy

The Commercial Space Transportation Study (CSTS) suggests that considerable market expansion in earth-to-orbit transportation would take place if current launch prices could be reduced to around $400 per pound of payload. If these low prices can be achieved, annual payload delivered to low earth orbit (LEO) is predicted to reach 6.7 million pounds. The primary market growth will occur in communications, government missions, and civil transportation. By establishing a cost target of $100-$200 per pound of payload for a new launch system, the Highly Reusable Space Transportation (HRST) program has clearly set its sights on removing the current restriction on market growth imposed by today's high launch costs. In particular, achieving the goal of $100-$200 per pound of payload will require significant coordinated efforts in (1) marketing strategy development, (2) business planning, (3) system operational strategy, (4) vehicle technical design, and (5) vehicle maintenance strategy.
Derived from text
Cost Reduction; Low Earth Orbits; Payloads; Reusable Launch Vehicles; Space Commercialization; Space Transportation; Spacecraft Launching

19960003439 NASA Marshall Space Flight Center, Huntsville, AL, USA
A guidance and control assessment of three vertical landing options for RLV
Gallaher, M.; Coughlin, D.; Krupp, D; Sep 1, 1995; In English
Report No.(s): NASA-TM-108500; NAS 1.15:108500; NIPS-95-05719; No Copyright; Avail: CASI; A04, Hardcopy

The National Aeronautics and Space Administration is considering a vertical lander as a candidate concept for a single-stage-to-orbit reusable launch vehicle (RLV). Three strategies for guiding and controlling the inversion of a reentering RLV from a nose-first attitude to a vertical landing attitude are suggested. Each option is simulated from a common reentry state to touchdown, using a common guidance algorithm and different controllers. Results demonstrate the characteristics that typify and distinguish each concept and help to identify peculiar problems, level of guidance and control sophistication required, feasibility concerns, and areas in which stringent subsystem requirements will be imposed by guidance and control.
Author
Command and Control; Command Guidance; Control Systems Design; Controllers; Single Stage to Orbit Vehicles; Space Transportation; Spacecraft Control; Technology Assessment; Vertical Landing

19960003337 Eloret Corp., Palo Alto, CA, USA
Hypersonic flows as related to the National Aerospace Plane
Kussoy, Marvin; Huang, George; Menter, Florian; May 9, 1995; In English
Contract(s)/Grant(s): NCC2-452
Report No.(s): NASA-CR-199365; NAS 1.26:199365; No Copyright; Avail: CASI; A03, Hardcopy

The object of Cooperative Agreement NCC2-452 was to identify, develop, and document reliable turbulence models for incorporation into CFD codes, which would then subsequently be incorporated into numerical design procedures for the NASP and any other hypersonic vehicles. In a two-pronged effort, consisting of an experimental and a theoretical approach, several key features of flows over complex vehicles were identified, and test bodies were designed which were composed of simple geometric shapes over which these flow features were measured. The experiments were conducted in the 3.5' Hypersonic Wind Tunnel at NASA Ames Research Center, at nominal Mach numbers from 7 to 8.3 and Re/m from 4.9 x 10(exp 6) to 5.8 x 10(exp 6). Boundary layers approaching the interaction region were 2.5 to 3.7 cm thick. Surface and flow field measurements were conducted, and the initial boundary conditions were experimentally documented.
Author
Aerospace Planes; Boundary Layers; Computational Fluid Dynamics; Flow Distribution; Flow Measurement; Hypersonic Flow; Hypersonic Speed; Shock Wave Interaction; Turbulence Models; Wind Tunnel Tests

19960003127 NASA Langley Research Center, Hampton, VA, USA
Single-stage-to-orbit: Meeting the challenge
Freeman, Delma C., Jr.; Talay, Theodore A.; Austin, Robert Eugene; Oct 6, 1995; In English; 46th International Astronautical Congress, 2-6 Oct. 1995, Oslo, Norway
Report No.(s): NASA-TM-111127; NIPS-95-05489; NAS 1.15:111127; IAF-95-V-5-07; No Copyright; Avail: CASI; A03, Hardcopy

There has been and continues to be significant discussion about the viability of fully reusable, single-stage-to-orbit (SSTO) concepts for delivery of payloads to orbit. Often, these discussions have focused in detail on performance and technology requirements relating to the technical feasibility of the concept, with only broad generalizations on how the SSTO

will achieve its economic goals of greatly reduced vehicle ground and flight operations costs. With the current industry and NASA Reusable Launch Vehicle Technology Program efforts underway to mature and demonstrate technologies leading to a viable commercial launch system that also satisfies national needs, achieving acceptable recurring costs becomes a significant challenge. This paper reviews the current status of the Reusable Launch Vehicle Technology Program including the DC-XA, X-33, and X-34 flight systems and associated technology programs. The paper also examines lessons learned from the recently completed DC-X reusable rocket demonstrator program. It examines how these technologies and flight systems address the technical and operability challenges of SSTO whose solutions are necessary to reduce costs. The paper also discusses the management and operational approaches that address the challenge of a new cost-effective, reusable launch vehicle system.
Author
Cost Effectiveness; Flight Operations; Single Stage to Orbit Vehicles; Space Transportation; Technology Assessment

19960003035 Joint Inst. for Advancement of Flight Sciences, Hampton, VA, USA
Abort performance for a winged-body single-stage to orbit vehicle
Lyon, Jeffery A.; Aug 1, 1995; In English
Contract(s)/Grant(s): NCC1-104; RTOP 242-80-01-02
Report No.(s): NASA-CR-4690; NAS 1.26:4690; No Copyright; Avail: CASI; A05, Hardcopy

Optimal control theory is employed to determine the performance of abort to orbit (ATO) and return to launch site (RTLS) maneuvers for a single-stage to orbit vehicle. The vehicle configuration examined is a seven engine, winged-body vehicle, that lifts-off vertically and lands horizontally. The abort maneuvers occur as the vehicle ascends to orbit and are initiated when the vehicle suffers an engine failure. The optimal control problems are numerically solved in discretized form via a nonlinear programming (NLP) algorithm. A description highlighting the attributes of this NLP method is provided. ATO maneuver results show that the vehicle is capable of ascending to orbit with a single engine failure at lift-off. Two engine out ATO maneuvers are not possible from the launch pad, but are possible after launch when the thrust to weight ratio becomes sufficiently large. Results show that single engine out RTLS maneuvers can be made for up to 180 seconds after lift-off and that there are scenarios for which RTLS maneuvers should be performed instead of ATP maneuvers.
Author
Abort Trajectories; Aerodynamic Characteristics; Engine Failure; Multiengine Vehicles; Single Stage to Orbit Vehicles; Winged Vehicles

19960002345 NASA Lewis Research Center, Cleveland, OH, USA
Human lunar mission capabilities using SSTO, ISRU and LOX-augmented NTR technologies: A preliminary assessment
Borowski, Stanley K.; Oct 1, 1995; In English; 31st Joint Propulsion Conference and Exhibit, 10-12 Jul. 1995, San Diego, CA, USA
Contract(s)/Grant(s): RTOP 242-10-01
Report No.(s): NASA-TM-107095; NAS 1.15:107095; E-9973; AIAA PAPER 95-2631; No Copyright; Avail: CASI; A03, Hardcopy

The feasibility of conducting human missions to the Moon is examined assuming the use of three ‘high leverage’ technologies: (1) a single-stage-to-orbit (SSTO) launch vehicle, (2) ‘in-situ’ resource utilization (ISRU)--specifically ‘lunar-derived’ liquid oxygen (LUNOX), and (3) LOX-augmented nuclear thermal rocket (LANTR) propulsion. Lunar transportation system elements consisting of a LANTR-powered lunar transfer vehicle (LTV) and a chemical propulsion lunar landing/Earth return vehicle (LERV) are configured to fit within the ‘compact’ dimensions of the SSTO cargo bay (diameter: 4.6 m/length: 9.0 m) while satisfying an initial mass in low Earth orbit (IMLEO) limit of approximately 60 t (3 SSTO launches). Using approximately 8 t of LUNOX to ‘reoxidize’ the LERV for a ‘direct return’ flight to Earth reduces its size and mass allowing delivery to LEO on a single 20 t SSTO launch. Similarly, the LANTR engine’s ability to operate at any oxygen/ hydrogen mixture ratio from 0 to 7 with high specific impulse (approximately 940 to 515 s) is exploited to reduce hydrogen tank volume, thereby improving packaging of the LANTR LTV’s ‘propulsion’ and ‘propellant modules’. Expendable and reusable, piloted and cargo missions and vehicle designs are presented along with estimates of LUNOX production required to support the different mission modes. Concluding remarks address the issue of lunar transportation system costs from the launch vehicle perspective.
Author
Chemical Propulsion; Low Earth Orbits; Lunar Landing; Moon; Nuclear Rocket Engines; Propulsion; Rocket Engines; Single Stage to Orbit Vehicles

19960000793 Santa Clara Univ., CA, USA

Advances in hypersonic vehicle synthesis with application to studies of advanced thermal protection system

Ardema, Mark D.; Sep 30, 1995; In English

Contract(s)/Grant(s): NCC2-5069

Report No.(s): NASA-CR-199333; NAS 1.26:199333; No Copyright; Avail: CASI; A04, Hardcopy

This report summarizes the work entitled 'Advances in Hypersonic Vehicle Synthesis with Application to Studies of Advanced Thermal Protection Systems.' The effort was in two areas: (1) development of advanced methods of trajectory and propulsion system optimization; and (2) development of advanced methods of structural weight estimation. The majority of the effort was spent in the trajectory area.

Derived from text

Engine Design; Hypersonic Vehicles; Propulsion System Configurations; Single Stage to Orbit Vehicles; Supersonic Combustion Ramjet Engines; Thermal Protection; Trajectory Optimization

19950028302 NASA Ames Research Center, Moffett Field, CA, USA

Air-breathing aerospace plane development essential: Hypersonic propulsion flight tests

Mehta, Unmeel B.; ESA, Proceedings of the 2nd European Symposium on Aerothermodynamics for Space Vehicles; Feb 1, 1995; In English

Report No.(s): NASA-TM-108857-REV; NAS 1.15:108857-REV; Copyright; Avail: CASI; A03, Hardcopy

Hypersonic airbreathing propulsion utilizing scramjets can change transatmospheric accelerators for low earth-to-orbit and return transportation. The value and limitation of ground tests, of flight tests, and of computations are presented, and scramjet development requirements are discussed. It is proposed that near full-scale hypersonic propulsion flight tests are essential for developing computational design technology so that it can be used for designing this system. In order to determine how these objectives should be achieved, some lessons learned from past programs are presented. A conceptual two-stage-to-orbit (TSTO) prototype/experimental aerospace plane is recommended as a means of providing access-to-space and for conducting flight tests.

Author

Aerospace Planes; Engine Tests; Flight Tests; Full Scale Tests; Hypersonic Speed; Spacecraft Propulsion; Supersonic Combustion Ramjet Engines

19950026746 NASA, Washington, DC, USA

Shuttle to Space Station. Heart assist implant. Hubble update. X-30 mock-up

Aug 1, 1992; In English

Report No.(s): NASA-TM-110837; NONP-NASA-VT-95-63907; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

Shuttle to Space Station, Heart Assist Implant, Hubble Update, and X-30 Mockup are the four parts that are discussed in this video. The first part, Shuttle to Space Station, is focussed on the construction and function of the Space Station Freedom. While part two, Heart Assist Implant, discusses a newly developed electromechanical device that helps to reduce heart attack by using electric shocks. Interviews with the co-inventor and patients are also included. Brief introduction to Hubble Telescope, problem behind its poor image quality (mirror aberration), and the plan to correct this problem are the three issues that are discussed in part three, Hubble Update. The last part, part four, reviews the X-30 Mockup designed by the staff and students of Mississippi State University.

CASI

Cardiovascular System; Heart Diseases; Hubble Space Telescope; Space Station Freedom; X-30 Vehicle

19950024291 NASA Marshall Space Flight Center, Huntsville, AL, USA

Transonic aerodynamic characteristics of a proposed wing-body reusable launch vehicle concept

Springer, A. M.; Mar 1, 1995; In English

Report No.(s): NASA-TM-108489; NAS 1.15:108489; No Copyright; Avail: CASI; A06, Hardcopy

A proposed wing-body reusable launch vehicle was tested in the NASA Marshall Space Flight Center's 14 x 14-inch trisonic wind tunnel during the winter of 1994. This test resulted in the vehicle's subsonic and transonic, Mach 0.3 to 1.96, longitudinal and lateral aerodynamic characteristics. The effects of control surface deflections on the basic vehicle's

aerodynamics, including a body flap, elevons, ailerons, and tip fins, are presented.
Author
Aerodynamic Characteristics; Aerodynamic Forces; Aerodynamic Stability; Angle of Attack; Body-Wing Configurations; Control Surfaces; Lateral Stability; Longitudinal Stability; Mach Number; Sideslip; Single Stage to Orbit Vehicles; Spacecraft Stability; Subsonic Speed; Transonic Speed; Wind Tunnel Tests

19950021839 Rockwell International Corp., Downey, CA, USA
Task 4 supporting technology. Densification requirements definition and test objectives. Propellant densification requirements definition
Lak, Tibor; Weeks, D. P.; May 30, 1995; In English
Contract(s)/Grant(s): NCC8-79
Report No.(s): NASA-CR-198629; NAS 1.26:198629; SSD95D0241; No Copyright; Avail: CASI; A03, Hardcopy

The primary challenge of the X-33 CAN is to build and test a prototype LO2 and LH2 densification ground support equipment (GSE) unit, and perform tank thermodynamic testing within the 15 month phase 1 period. The LO2 and LH2 propellant densification system will be scaled for the IPTD LO2 and LH2 tank configurations. The IPTD tanks were selected for the propellant technology demonstration because of the potential benefits to the phase 1 plan: tanks will be built in time to support thermodynamic testing; minimum cost; minimum schedule risk; future testing at MSFC will build on phase 1 data base; and densification system will be available to support X-33 and RLV engine test at IPTD. The objective of the task 1 effort is to define the preliminary requirements of the propellant densification GSE and tank recirculation system. The key densification system design parameters to be established in Task 1 are: recirculation flow rate; heat exchanger inlet temperature; heat exchanger outlet temperature; maximum heat rejection rate; vent flow rate (GN2 and GH2); densification time; and tank pressure level.
Derived from text
Densification; Heat Exchangers; Liquid Hydrogen; Liquid Oxygen; Liquid Rocket Propellants; Propellant Storage; Reusable Launch Vehicles

19950021838 Rockwell International Corp., Downey, CA, USA
Task 4 supporting technology. Part 2: Detailed test plan for thermal seals. Thermal seals evaluation, improvement and test. CAN8-1, Reusable Launch Vehicle (RLV), advanced technology demonstrator: X-33. Leading edge and seals thermal protection system technology demonstration
Hogenson, P. A.; Lu, Tina; May 30, 1995; In English
Contract(s)/Grant(s): NCC8-79
Report No.(s): NASA-CR-198628; NAS 1.26:198628; SSD95D0240-PT-2; No Copyright; Avail: CASI; A03, Hardcopy

The objective is to develop the advanced thermal seals to a technology readiness level (TRL) of 6 to support the rapid turnaround time and low maintenance requirements of the X-33 and the future reusable launch vehicle (RLV). This program is divided into three subtasks: (1) orbiter thermal seals operation history review; (2) material, process, and design improvement; and (3) fabrication and evaluation of the advanced thermal seals.
CASI
Compression Tests; High Temperature Tests; Reentry Shielding; Reusable Launch Vehicles; Sealers; Technology Assessment; Thermal Insulation; Thermal Protection

19950021837 Rockwell International Corp., Downey, CA, USA
Task 4 supporting technology. Part 1: Detailed test plan for leading edge tile development. Leading edge material development and testing
Hogenson, P. A.; Staszak, Paul; Hinkle, Karrie; May 30, 1995; In English
Contract(s)/Grant(s): NCC8-79
Report No.(s): NASA-CR-198633; NAS 1.26:198633; SSD95D0239-PT-1; No Copyright; Avail: CASI; A03, Hardcopy

This task develops two alternative candidate tile materials for leading edge applications: coated alumina enhanced thermal barrier (AETB) tile and silicone impregnated reusable ceramic ablator (SIRCA) tile. Upon reentry of the X-33/RLV space vehicle, the leading edges experience the highest heating rates and temperatures. The wing leading edge and nose cap experience peak temperatures in the range 2000 to 2700 F. Replacing reinforced carbon-carbon (RCC) with tile-based thermal protection system (TPS) materials is the primary objective. Weight, complexity, coating impact damage, and repairability are among the problems that this tile technology development addresses. The following subtasks will be performed in this

development effort: tile coating development; SIRCA tile development; robustness testing of tiles; tile repair development; tile operations/processing; tile leading edge configuration; and life cycle testing.
CASI
Ablative Materials; Aluminum Oxides; Ceramics; Coatings; Leading Edges; Reentry Shielding; Reusable Heat Shielding; Reusable Launch Vehicles; Silicones; Thermal Protection; Tiles

19950020825 GPS Solutions, Inc., Carson City, NV, USA
Reusable launch vehicle development research
Jan 16, 1995; In English
Contract(s)/Grant(s): NAS8-40257
Report No.(s): NASA-CR-196614; NAS 1.26:196614; No Copyright; Avail: CASI; A03, Hardcopy

NASA has generated a program approach for a SSTO reusable launch vehicle technology (RLV) development which includes a follow-on to the Ballistic Missile Defense Organization's (BMDO) successful DC-X program, the DC-XA (Advanced). Also, a separate sub-scale flight demonstrator, designated the X-33, will be built and flight tested along with numerous ground based technologies programs. For this to be a successful effort, a balance between technical, schedule, and budgetary risks must be attained. The adoption of BMDO's 'fast track' management practices will be a key element in the eventual success of NASA's effort.
Derived from text
Ballistic Missiles; Missile Defense; NASA Space Programs; Reusable Launch Vehicles; Single Stage to Orbit Vehicles

19950019884 Vanderbilt Univ., Nashville, TN, USA
The 1994 NASA/USRA/ADP Design Projects
Cruse, Thomas; Richardson, Joseph; Tryon, Robert; JAN 1, 1994; In English; See also N95-26305 through N95-26310; 7 functional color pages
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-197415; NAS 1.26:197415; No Copyright; Avail: CASI; A07, Hardcopy; 7 functional color pages

The NASA/USRA/ADP Design Projects from Vanderbilt University, Department of Mechanical Engineering (1994) are enclosed in this final report. Design projects include: (1) Protein Crystal Growth, both facilities and methodology; (2) ACES Deployable Space Boom; (3) Hybrid Launch System designs for both manned and unmanned systems; (4) LH2 Fuel Tank design (SSTO); (5) SSTO design; and (6) Pressure Tank Feed System design.
Experiment Design; Functional Design Specifications; Government/Industry Relations; Launch Vehicle Configurations; Mechanical Engineering; Pressure Vessel Design; Single Stage to Orbit Vehicles; Spacecraft Design; Structural Design; University Program

19950017766 NASA Lewis Research Center, Cleveland, OH, USA
A summary of the slush hydrogen technology program for the National Aero-Space Plane
Mcnelis, Nancy B.; Hardy, Terry L.; Whalen, Margaret V.; Kudlac, Maureen T.; Moran, Matthew E.; Tomsik, Thomas M.; Haberbusch, Mark S.; Apr 1, 1995; In English; Hypersonics Technologies Conference, 3-7 Apr. 1995, Chattanooga, TN, USA
Contract(s)/Grant(s): RTOP 763-22-21
Report No.(s): NASA-TM-106863; E-9469; NAS 1.15:106863; AIAA PAPER 95-6056; No Copyright; Avail: CASI; A03, Hardcopy

Slush hydrogen, a mixture of solid and liquid hydrogen, offers advantages of higher density (16 percent) and higher heat capacity (18 percent) than normal boiling point hydrogen. The combination of increased density and heat capacity of slush hydrogen provided a potential to decrease the gross takeoff weight of the National Aero-Space Plane (NASP) and therefore slush hydrogen was selected as the propellant. However, no large-scale data was available on the production, transfer and tank pressure control characteristics required to use slush hydrogen as a fuel. Extensive testing has been performed at the NASA Lewis Research Center K-Site and Small Scale Hydrogen Test Facility between 1990 and the present to provide a database for the use of slush hydrogen. This paper summarizes the results of this testing.
Author
Aerospace Planes; Fuel Tests; Hydrogen Fuels; National Aerospace Plane Program; Slush; Slush Hydrogen

19950016772 Rockwell International Corp., Downey, CA, USA, Northrop Grumman Corp., Los Angeles, CA, USA, North American Aviation, Inc., Tulsa, OK, USA, Hercules, Inc., Wilmington, DE, USA

Structural arrangement trade study. Volume 3: Reusable Hydrogen Composite Tank System (RHCTS) and Graphite Composite Primary Structures (GCPS). Addendum

Mar 14, 1995; In English

Contract(s)/Grant(s): NCC8-39; NCC1-193

Report No.(s): NASA-CR-197687; NAS 1.26:197687; SSD95D0069-VOL-3; No Copyright; Avail: CASI; A12, Hardcopy

This volume is the third of a 3 volume set that addresses the structural trade study plan that will identify the most suitable structural configuration for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 deg inclination. The most suitable Reusable Hydrogen Composite Tank System (RHCTS), and Graphite Composite Tank System (GCPS) composite materials for intertank, wing and thrust structures are identified. Vehicle resizing charts, selection criteria and back-up charts, parametric costing approach and the finite element method analysis are discussed.

Derived from text

Composite Structures; Dynamic Structural Analysis; Finite Element Method; Graphite; Hydrogen Fuels; Single Stage to Orbit Vehicles; Spacecraft Structures

19950016771 Rockwell International Corp., Downey, CA, USA, Northrop Grumman Corp., Los Angeles, CA, USA, North American Aviation, Inc., Tulsa, OK, USA, Hercules, Inc., Wilmington, DE, USA

Structural Arrangement Trade Study. Volume 1: Reusable Hydrogen Composite Tank System (RHCTS) and Graphite Composite Primary Structures (GCPS). Executive summary

Mar 14, 1995; In English

Contract(s)/Grant(s): NCC8-39; NCC1-193

Report No.(s): NASA-CR-197685; NAS 1.26:197685; SSD95D0069-VOL-1; No Copyright; Avail: CASI; A03, Hardcopy

This volume is the first of a three volume set that discusses the structural arrangement trade study plan that will identify the most suitable configuration for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 deg inclination. The Reusable Hydrogen Composite Tank System (RHCTS), and Graphite Composite Primary Structures most suitable for intertank, wing and thrust structures are identified. This executive summary presents the trade study process, the selection process, requirements used, analysis performed and data generated. Conclusions and recommendations are also presented.

Derived from text

Composite Structures; Graphite; Hydrogen Fuels; Project Planning; Single Stage to Orbit Vehicles; Spacecraft Structures

19950010487 NASA Langley Research Center, Hampton, VA, USA

Constrained minimization of smooth functions using a genetic algorithm

Moerder, Daniel D.; Pamadi, Bandu N.; Nov 1, 1994; In English

Contract(s)/Grant(s): RTOP 506-59-61-01

Report No.(s): NASA-TP-3329; L-17221; NAS 1.60-3329; No Copyright; Avail: CASI; A03, Hardcopy

The use of genetic algorithms for minimization of differentiable functions that are subject to differentiable constraints is considered. A technique is demonstrated for converting the solution of the necessary conditions for a constrained minimum into an unconstrained function minimization. This technique is extended as a global constrained optimization algorithm. The theory is applied to calculating minimum-fuel ascent control settings for an energy state model of an aerospace plane.

Author

Aerospace Planes; Genetic Algorithms; Lagrange Multipliers; Mathematical Models; Minima; Optimization

19950009268 NASA Lewis Research Center, Cleveland, OH, USA

The rationale/benefits of nuclear thermal rocket propulsion for NASA's lunar space transportation system

Borowski, Stanley K.; Sep 1, 1994; In English; 27th Joint Propulsion Conference, 24-26 Jun. 1991, Sacramento, CA, USA

Contract(s)/Grant(s): RTOP 232-01-06

Report No.(s): NASA-TM-106739; E-9142; NAS 1.15:106739; AIAA PAPER 91-2052; No Copyright; Avail: CASI; A03, Hardcopy

The solid core nuclear thermal rocket (NTR) represents the next major evolutionary step in propulsion technology. With its attractive operating characteristics, which include high specific impulse (approximately 850-1000 s) and engine thrust-to-weight (approximately 4-20), the NTR can form the basis for an efficient lunar space transportation system (LTS)

capable of supporting both piloted and cargo missions. Studies conducted at the NASA Lewis Research Center indicate that an NTR-based LTS could transport a fully-fueled, cargo-laden, lunar excursion vehicle to the Moon, and return it to low Earth orbit (LEO) after mission completion, for less initial mass in LEO than an aerobraked chemical system of the type studied by NASA during its '90-Day Study.' The all-propulsive NTR-powered LTS would also be 'fully reusable' and would have a 'return payload' mass fraction of approximately 23 percent--twice that of the 'partially reusable' aerobraked chemical system. Two NTR technology options are examined--one derived from the graphite-moderated reactor concept developed by NASA and the AEC under the Rover/NERVA (Nuclear Engine for Rocket Vehicle Application) programs, and a second concept, the Particle Bed Reactor (PBR). The paper also summarizes NASA's lunar outpost scenario, compares relative performance provided by different LTS concepts, and discusses important operational issues (e.g., reusability, engine 'end-of life' disposal, etc.) associated with using this important propulsion technology.
Author
Lunar Bases; Lunar Orbital Rendezvous; Mission Planning; Nuclear Engine For Rocket Vehicles; Nuclear Propulsion; Propulsion System Configurations; Reusable Launch Vehicles; Space Transportation System; Thrust-Weight Ratio

19950008507 NASA Ames Research Center, Moffett Field, CA, USA
Air-breathing aerospace plane development essential: Hypersonic propulsion flight tests
Mehta, Unmeel B.; Nov 1, 1994; In English; 2nd European Symposium on Aerothermodynamics for Space Vehicles, ESTEC, 21-23 Nov. 1994, Noordwijk, Netherlands
Contract(s)/Grant(s): RTOP 505-59-53
Report No.(s): NASA-TM-108857; A-95017; NAS 1.15:108857; No Copyright; Avail: CASI; A03, Hardcopy

Hypersonic air-breathing propulsion utilizing scramjets can fundamentally change transatmospheric accelerators for low earth-to-orbit and return transportation. The value and limitations of ground tests, of flight tests, and of computations are presented, and scramjet development requirements are discussed. It is proposed that near full-scale hypersonic propulsion flight tests are essential for developing a prototype hypersonic propulsion system and for developing computational-design technology so that it can be used for designing this system. In order to determine how these objectives should be achieved, some lessons learned from past programs are presented. A conceptual two-stage-to-orbit (TSTO) prototype/experimental aerospace plane is recommended as a means of providing access-to-space and for conducting flight tests. A road map for achieving these objectives is also presented.
Author
Aerospace Planes; Flight Tests; Hypersonic Flight; Space Transportation; Supersonic Combustion Ramjet Engines

19950008387 Rockwell International Corp., Downey, CA, USA
Milestone 4: Test plan for Reusable Hydrogen Composite Tank System (RHCTS). Task 3: Composite tank materials
Greenberg, H. S.; Nov 4, 1994; In English
Contract(s)/Grant(s): NCC8-39
Report No.(s): NASA-CR-197036; NAS 1.26:197036; SD94D0329; No Copyright; Avail: CASI; A04, Hardcopy

This document is the detailed test plan for the series of tests enumerated in the preceding section. The purpose of this plan is to present the test objectives, test parameters and procedures, expected performance and data analysis plans, criteria for success, test schedules, and related safety provisions and to describe the test articles, test instrumentation, and test facility requirements. Initial testing will be performed to screen four composite materials for suitability for SSTO LH2 tank loads and environmental conditions. The laminates for this testing will be fabricated by fiber placement, which is the manufacturing approach identified as baseline for the tank wall. Even though hand layup will be involved in fabricating many of the internal structural members of the tank, no hand-layup laminates will be evaluated in the screening or subsequent characterization testing. This decision is based on the understanding that mechanical properties measured for hand-layup material should be at least equivalent to properties measured for fiber-placed material, so that the latter should provide no less than a conservative approximation of the former. A single material will be downselected from these screening tests. This material will be subsequently characterized for impact-damage tolerance and durability under conditions of mechanical and thermal cycling, and to establish a preliminary design database to support ongoing analysis. Next, testing will be performed on critical structural elements fabricated from the selected material. Finally, the 8-foot diameter tank article, containing the critical structural features of the full-scale tank, will be fabricated by fiber placement and tested to verify its structural integrity and LH2 containment.
Derived from text
Fatigue Tests; Impact Tests; Laminates; Liquid Hydrogen; Performance Tests; Propellant Tanks; Single Stage to Orbit Vehicles; Thermal Cycling Tests

19950008386 Rockwell International Corp., Downey, CA, USA
Milestone 4: Thrust structure concepts and IHM screening Graphite Composite Primary Structure (GCPS)
Greenberg, H. S.; Nov 4, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-197053; NAS 1.26:197053; SSD94D0328; No Copyright; Avail: CASI; A03, Hardcopy

The first part of the task was to select up to three promising thrust structure constructions and to select materials for screening tests. Part of the nondestructive evaluation and inspection (NDE/I) and integrated health management (IHM) task is to acquire and develop NDE/I sensor technologies and to integrate those sensors into the full scale test articles which will be produced under the TA2 program. Review of the anticipated fault modes and the available sensor technology data indicates that three sensor technologies should be assessed for the in-situ monitoring of the composite primary structure elements. These are: ultrasonics (dry contact), acoustic emissions, and fiber optics (embedded or attached). In fact, a combination of sensor technologies will be needed to detect and evaluate the fault modes; not only do sensor technology have specific capabilities and applicability, but the three Gr/Ep primary structures being demonstrated under the TA2 effort have differing requirements based on their respective failure modes and designs.
Derived from text
Composite Structures; Graphite-Epoxy Composites; Nondestructive Tests; Single Stage to Orbit Vehicles; Spacecraft Structures; Structural Design

19950008380 Rockwell International Corp., Downey, CA, USA
Requirements report for SSTO vertical take-off and horizontal landing vehicle
Greenberg, H. S.; Nov 4, 1994; In English
Contract(s)/Grant(s): NCC1-193; NCC2-9003; NCC8-39
Report No.(s): NASA-CR-197029; NAS 1.26:197029; SSD94D0217B; No Copyright; Avail: CASI; A08, Hardcopy

This document describes the detailed design requirements and design criteria to support Structures/TPS Technology development for SSTO winged vehicle configurations that use vertical take-off and horizontal landing and delivers 25,000 lb payloads to a 220 nm circular orbit at an inclination of 51.6 degrees or 40,000 lb payloads to a 150 nm circular orbit at a 28.5 degree inclination.
Derived from text
Design Analysis; Launch Vehicle Configurations; Single Stage to Orbit Vehicles; Spacecraft Design; Structural Design; Thermal Protection

19950008107 Minnesota Univ., Minneapolis, MN, USA
Access to space
Jul 1, 1994; In English
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-197153; NAS 1.26:197153; No Copyright; Avail: CASI; A07, Hardcopy

The goal of this conceptual design was to devise a reusable, commercially viable, single-stage-to-orbit vehicle. The vehicle has the ability to deliver a 9100 kg (20,000 lb) payload to a low earth orbit of 433 km to 933 km (250 n.mi. - 450 n.mi.). The SSTO vehicle is 51 meters in length and has a gross takeoff mass of 680,400 kg (1,500,000 lb). The vehicle incorporates three RD-701 engines for the main propulsion system and two RL-10 engines for the orbital maneuvering system. The vehicle is designed for a three day stay on orbit with two crew members.
Author
Aerospace Planes; Launch Vehicle Configurations; Life Cycle Costs; Propulsion System Configurations; Single Stage to Orbit Vehicles; Space Commercialization; Space Transportation; Spacecraft Design; Structural Design; Transatmospheric Vehicles; Wind Tunnel Models

19950007252 NASA Lewis Research Center, Cleveland, OH, USA
Computational analysis in support of the SSTO flowpath test
Duncan, Beverly S.; Trefny, Charles J.; Oct 1, 1994; In English
Contract(s)/Grant(s): NAS3-27186; RTOP 505-70-59
Report No.(s): NASA-TM-106757; E-9179; NAS 1.15:106757; No Copyright; Avail: CASI; A03, Hardcopy

A synergistic approach of combining computational methods and experimental measurements is used in the analysis of a hypersonic inlet. There are four major focal points within this study which examine the boundary layer growth on a

compression ramp upstream of the cowl lip of a scramjet inlet. Initially, the boundary layer growth on the NASP Concept Demonstrator Engine (CDE) is examined. The follow-up study determines the optimum diverter height required by the SSTO Flowpath test to best duplicate the CDE results. These flow field computations are then compared to the experimental measurements and the mass average Mach number is determined for this inlet.
Author
Boundary Layer Flow; Cowlings; Diverters; Engine Inlets; Hypersonic Inlets; Inlet Flow

19950006425 NASA Langley Research Center, Hampton, VA, USA
Hypersonic vehicle model and control law development using H(infinity) and micron synthesis
Gregory, Irene M.; Chowdhry, Rajiv S.; Mcminn, John D.; Shaughnessy, John D.; Oct 1, 1994; In English
Contract(s)/Grant(s): RTOP 505-70-64-01
Report No.(s): NASA-TM-4562; L-17217; NAS 1.15:4562; No Copyright; Avail: CASI; A03, Hardcopy

The control system design for a Single Stage To Orbit (SSTO) air breathing vehicle will be central to a successful mission because a precise ascent trajectory will preserve narrow payload margins. The air breathing propulsion system requires the vehicle to fly roughly halfway around the Earth through atmospheric turbulence. The turbulence, the high sensitivity of the propulsion system to inlet flow conditions, the relatively large uncertainty of the parameters characterizing the vehicle, and continuous acceleration make the problem especially challenging. Adequate stability margins must be provided without sacrificing payload mass since payload margins are critical. Therefore, a multivariable control theory capable of explicitly including both uncertainty and performance is needed. The H(infinity) controller in general provides good robustness but can result in conservative solutions for practical problems involving structured uncertainty. Structured singular value mu framework for analysis and synthesis is potentially much less conservative and hence more appropriate for problems with tight margins. An SSTO control system requires: highly accurate tracking of velocity and altitude commands while limiting angle-of-attack oscillations, minimized control power usage, and a stabilized vehicle when atmospheric turbulence and system uncertainty are present. The controller designs using H(infinity) and mu-synthesis procedures were compared. An integrated flight/propulsion dynamic mathematical model of a conical accelerator vehicle was linearized as the vehicle accelerated through Mach 8. Vehicle acceleration through the selected flight condition gives rise to parametric variation that was modeled as a structured uncertainty. The mu-analysis approach was used in the frequency domain to conduct controller analysis and was confirmed by time history plots. Results demonstrate the inherent advantages of the mu framework for this class of problems.
Author (revised)
Aerodynamic Characteristics; Control Systems Design; Dynamic Models; Hypersonic Vehicles; Mathematical Models; Propulsion System Configurations; Robustness (Mathematics); Single Stage to Orbit Vehicles

19950006252 California Univ., Los Angeles, CA, USA, Universities Space Research Association, Columbia, MD, USA
Ion engine propelled Earth-Mars cycler with nuclear thermal propelled transfer vehicle, volume 2
Meyer, Rudolf X.; Baker, Myles; Melko, Joseph; JAN 1, 1994; In English
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-197208; NAS 1.26:197208; No Copyright; Avail: CASI; A04, Hardcopy

The goal of this project was to perform a preliminary design of a long term, reusable transportation system between earth and Mars which would be capable of providing both artificial gravity and shelter from solar flare radiation. The heart of this system was assumed to be a Cycler spacecraft propelled by an ion propulsion system. The crew transfer vehicle was designed to be propelled by a nuclear-thermal propulsion system. Several Mars transportation system architectures and their associated space vehicles were designed.
Author
Ion Engines; Ion Propulsion; Manned Mars Missions; Manned Spacecraft; Nuclear Propulsion; Nuclear Rocket Engines; Reusable Spacecraft; Rotating Environments; Space Transportation; Spacecraft Design

19950006217 Vanderbilt Univ., Nashville, TN, USA, Universities Space Research Association, Columbia, MD, USA
Various advanced design projects promoting engineering education
JAN 1, 1994; In English; 6 functional color pages
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-197170; NAS 1.26:197170; No Copyright; Avail: CASI; A07, Hardcopy; 6 functional color pages

The Universities Space Research Association (USRA) Advanced Design Program (ADP) program promotes engineering

education in the field of design by presenting students with challenging design projects drawn from actual NASA interests. In doing so, the program yields two very positive results. Firstly, the students gain a valuable experience that will prepare them for design problems with which they will be faced in their professional careers. Secondly, NASA is able to use the work done by students as an additional resource in meeting its own design objectives. The 1994 projects include: Universal Test Facility; Automated Protein Crystal Growth Facility; Stiffening of the ACES Deployable Space Boom; Launch System Design for Access to Space; LH2 Fuel Tank Design for SSTO Vehicle; and Feed System Design for a Reduced Pressure Tank.
CASI

Feed Systems; Fuel Tanks; Hybrid Rocket Engines; Launch Vehicles; Mechanical Engineering; Single Stage to Orbit Vehicles; Space Erectable Structures; Space Processing; Structural Design

19950006206 Lockheed Engineering and Sciences Co., Hampton, VA, USA

An implementation of a chemical and thermal nonequilibrium flow solver on unstructured meshes and application to blunt bodies

Prabhu, Ramadas K.; Aug 1, 1994; In English

Contract(s)/Grant(s): NAS1-19000; RTOP 232-01-04-06

Report No.(s): NASA-CR-194967; NAS 1.26:194967; No Copyright; Avail: CASI; A04, Hardcopy

This paper presents a nonequilibrium flow solver, implementation of the algorithm on unstructured meshes, and application to hypersonic flow past blunt bodies. Air is modeled as a mixture of five chemical species, namely O2, N2, O, NO, and N, having two temperatures namely translational and vibrational. The solution algorithm is a cell centered, point implicit upwind scheme that employs Roe's flux difference splitting technique. Implementation of this algorithm on unstructured meshes is described. The computer code is applied to solve Mach 15 flow with and without a Type IV shock interference on a cylindrical body of 2.5mm radius representing a cowl lip. Adaptively generated meshes are employed, and the meshes are refined several times until the solution exhibits detailed flow features and surface pressure and heat flux distributions. Effects of a catalytic wall on surface heat flux distribution are studied. For the Mach 15 Type IV shock interference flow, present results showed a peak heat flux of 544 MW/m2 for a fully catalytic wall and 431 MW/m(exp 2) for a noncatalytic wall. Some of the results are compared with available computational data.
Author

Air Flow; Blunt Bodies; Computer Programs; Cylindrical Bodies; Grid Generation (Mathematics); Hypersonic Flow; Mach Number; Nonequilibrium Flow; Vibration

19950005762 Vigyan Research Associates, Inc., Hampton, VA, USA

A simple analytical aerodynamic model of Langley Winged-Cone Aerospace Plane concept

Pamadi, Bandu N.; Oct 1, 1994; In English

Contract(s)/Grant(s): NAS1-19341; RTOP 232-01-04-05

Report No.(s): NASA-CR-194987; NAS 1.26:194987; No Copyright; Avail: CASI; A03, Hardcopy

A simple three DOF analytical aerodynamic model of the Langley Winged-Coned Aerospace Plane concept is presented in a form suitable for simulation, trajectory optimization, and guidance and control studies. The analytical model is especially suitable for methods based on variational calculus. Analytical expressions are presented for lift, drag, and pitching moment coefficients from subsonic to hypersonic Mach numbers and angles of attack up to +/- 20 deg. This analytical model has break points at Mach numbers of 1.0, 1.4, 4.0, and 6.0. Across these Mach number break points, the lift, drag, and pitching moment coefficients are made continuous but their derivatives are not. There are no break points in angle of attack. The effect of control surface deflection is not considered. The present analytical model compares well with the APAS calculations and wind tunnel test data for most angles of attack and Mach numbers.
Author (revised)

Aerodynamic Coefficients; Aerodynamic Drag; Aerospace Planes; Angle of Attack; Hypersonic Aircraft; Lift; Mach Number; Mathematical Models; Pitching Moments

19950005532 Rockwell International Corp., Downey, CA, USA, Hercules, Inc., Wilmington, DE, USA

Trade study plan for Reusable Hydrogen Composite Tank System (RHCTS)

Greenberg, H. S.; Jul 29, 1994; In English

Contract(s)/Grant(s): NCC8-39

Report No.(s): NASA-CR-196871; NAS 1.26:196871; SSD94D0219; RHCTS-TSP-1; No Copyright; Avail: CASI; A03, Hardcopy

This TA 1 document describes the trade study plan (with support from TA 2) that will identify the most suitable structural configuration for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 degree inclination. The analysis uses information derived in the TA 2 study as identified within the study plan. In view of this, for convenience, the TA 2 study plan is included as an appendix to this document.
Derived from text
Composite Structures; Launch Vehicle Configurations; Liquid Hydrogen; Propellant Tanks; Single Stage to Orbit Vehicles; Spacecraft Design

19950005521 NASA Lyndon B. Johnson Space Center, Houston, TX, USA
Liquid flyback booster pre-phase: A study assessment
Peterson, W.; Ankney, W.; Bell, J.; Berning, M.; Bryant, L.; Bufkin, A.; Cain, L.; Caram, J.; Cockrell, B.; Curry, D., et al.; Sep 1, 1994; In English
Report No.(s): NASA-TM-104801; S-778; NAS 1.15:104801; No Copyright; Avail: CASI; A08, Hardcopy

The concept of a flyback booster has been around since early in the shuttle program. The original two-stage shuttle concepts used a manned flyback booster. These boosters were eliminated from the program for funding and size reasons. The current shuttle uses two Redesigned Solid Rocket Motors (RSRM's), which are recovered and refurbished after each flight; this is one of the major cost factors of the program. Replacement options have been studied over the past ten years. The conclusion reached by the most recent study is that the liquid flyback booster (LFBB) is the only competitive option from a life-cycle cost perspective. The purpose of this study was to assess the feasibility and practicality of LFBB's. The study provides an expansion of the recommendations made during the aforementioned study. The primary benefits are the potential for enhanced reusability and a reduction of recurring costs. The potential savings in vehicle turnaround could offset the up-front costs. Development of LFBB's requires a commitment to the shuttle program for 20 to 30 years. LFBB's also offer enhanced safety and abort capabilities. Currently, any failure of an RSRM can be considered catastrophic, since there are no intact abort capabilities during the burn of the RSRM's. The performance goal of the LFBB's was to lift a fully loaded orbiter under optimal conditions, so as not to be the limiting factor of the performance capability of the shuttle. In addition, a final benefit is the availability of growth paths for applications other than shuttle.
Author (revised)
Booster Recovery; Reusable Launch Vehicles; Solid Propellant Rocket Engines; Space Shuttle Boosters; Spacecraft Launching

19950005422 Rockwell International Corp., Downey, CA, USA
Test Plans. Lightweight Durable TPS: Tasks 1,2,4,5, and 6
Greenberg, H. S.; Tu, Tina; Jul 29, 1994; In English
Contract(s)/Grant(s): NCC2-9003
Report No.(s): NASA-CR-196867; NAS 1.26:196867; LDTPS-TP-1; SSD94D0218; No Copyright; Avail: CASI; A10, Hardcopy

The objective of this task is to develop the fluted core flexible blankets, also referred to as the Tailorable Advanced Blanket Insulation (TABI), to a technology readiness level (TRL) of 6. This task is one of the six tasks under TA 3, Lightweight Durable TPS study, of the Single Stage to Orbit (SSTO) program. The purpose of this task is to develop a durable and low maintenance flexible TPS blanket material to be implemented on the SSTO vehicle.
Author
Ceramic Fibers; Composite Materials; Fabrication; Government/Industry Relations; Single Stage to Orbit Vehicles; Thermal Insulation

19950005421 Rockwell International Corp., Downey, CA, USA, Grumman Aerospace Corp., Bethpage, NY, USA
Trade study plan for Graphite Composite Primary Structure (GCPS)
Greenberg, H. S.; Jul 29, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-196868; NAS 1.26:196868; SSD94D0222; GCPS-TSP-1; No Copyright; Avail: CASI; A03, Hardcopy

This TA 2 document (with support from TA 1) describes the trade study plan that will identify the most suitable structural configuration for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 degree inclination For this most suitable configuration the structural attachment of the wing, and the most suitable GCPS composite materials

for intertank, wing, tail and thrust structure are identified. This trade study analysis uses extensive information derived in the TA 1 trade study plan and is identified within the study plan. In view of this, for convenience, the TA 1 study plan is included as an appendix to this document.
Author
Composite Structures; Fuel Tanks; Graphite; Single Stage to Orbit Vehicles; Thrust-Weight Ratio

19950005420 Rockwell International Corp., Downey, CA, USA, Northrop Corp., Los Angeles, CA, USA
Test Plan. GCPS Task 4, subtask 4.2 thrust structure development
Greenberg, H. S.; Sep 2, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-196869; NAS 1.26:196869; SSD94D0211; No Copyright; Avail: CASI; A08, Hardcopy

The Single Stage To Orbit (SSTO) vehicle is designed to lift off from a vertical position, go into orbit, return to earth for a horizontal landing, and be reusable for the next mission. (NASA baseline only) In order to meet its performance goals, the SSTO relies on light weight structure and the use of 8 tri-propellant engines. These engines are mounted to the thrust structure. This test plan addresses selection of the material for this structure, and the integrity of the design through testing of elements and a full-scale subcomponent. This test plan supports the development of the design for an advanced composite thrust structure for a Single Stage to Orbit manned, heavy launch vehicle. The thrust structure is designed to transmit very high thrust loads from the engines to the rest of the vehicle (see Figure 1). The thrust structure will also be used for primary attachment of the twin vertical tails and possibly act as the aft attach point for the wing. The combination of high loading, high vibration, long service life and high acoustic environments will need to be evaluated by tests. To minimize design risk, a building block approach will be used. We will first screen materials to determine which materials show the most promise for this application. Factors in this screening will be the suitability of these materials for chosen design concepts, particularly concerning specific strength, environmental compatibility and applicability to fabrication processes. Next we will characterize two material systems that will be used in the design; the characterization will allow us to generate preliminary design data that will be used for the analysis. Element testing will be performed to evaluate critical structural locations under load. Final testing on the full scale test article will be performed to verify the design and to demonstrate predictability of the analysis. Additionally, risks associated with fabricating full scale thrust structures will be reduced through testing activities. One of the major concerns that stems from full scale fabrication is the realities of size and the associated complexities of handling, manufacturing, and assembly. The need exists to fabricate, assemble and test_representative joint specimens to achieve_confidence in the design and manufacturing technologies being proposed.
Author
Composite Structures; Design to Cost; Full Scale Tests; Propulsion System Configurations; Risk; Single Stage to Orbit Vehicles; Thrust-Weight Ratio

19950005410 Rockwell International Corp., Downey, CA, USA
Integrated propulsion technology demonstrator. Program plan
Aug 15, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-196870; NAS 1.26:196870; SSD94D0207; No Copyright; Avail: CASI; A05, Hardcopy

NASA and Rockwell have embarked on a cooperative agreement to define, develop, fabricate, and operate an integrated propulsion technology demonstrator (IPTD) for the purpose of validating design, process, and technology improvements of launch vehicle propulsion systems. This program, a result of NRA8-11, Task Area 1 A, is jointly funded by both NASA and Rockwell and is sponsored by the Reusable Launch Vehicle office at NASA Marshall Space flight Center. This program plan provides to the joint NASA/Rockwell integrated propulsion technology demonstrator (IPTD) team a description of the activities within tasks / sub tasks and associated schedules required to successfully achieve program objectives. This document also defines the cost elements and manpower allocations for each sub task for purpose of program control. This plan is updated periodically by developing greater depth of direction for outyear tasks as the program matures. Updating is accomplished by adding revisions to existing pages or attaching page revisions to this plan. In either case, revisions will be identified by appropriate highlighting of the change, or specifying a revision page through the use of footnotes on the bottom right of each change page. Authorization for the change is provided by the principal investigators to maintain control of this program plan document and IPTD program activities.
Author
Project Planning; Propulsion System Configurations; Requirements; Rocket Engine Design; Schedules; Single Stage to Orbit Vehicles

19950005403 NASA Marshall Space Flight Center, Huntsville, AL, USA
Damage tolerance of candidate thermoset composites for use on single stage to orbit vehicles
Nettles, A. T.; Lance, D.; Hodge, A.; Sep 1, 1994; In English
Report No.(s): NASA-TP-3506; M-758; NAS 1.60:3506; No Copyright; Avail: CASI; A03, Hardcopy

Four fiber/resin systems were compared for resistance to damage and damage tolerance. One toughened epoxy and three toughened bismaleimide (BMI) resins were used, all with IM7 carbon fiber reinforcement. A statistical design of experiments technique was used to evaluate the effects of impact energy, specimen thickness, and impactor diameter on the damage area, as computed by C-scans, and residual compression-after-impact (CAI) strength. Results showed that two of the BMI systems sustained relatively large damage zones yet had an excellent retention of CAI strength.
Author (revised)
Bismaleimide; Carbon Fiber Reinforced Plastics; Epoxy Resins; Impact Damage; Impact Resistance; Matrix Materials; Polyimide Resins; Single Stage to Orbit Vehicles; Spacecraft Construction Materials

19950005396 Rockwell International Corp., Downey, CA, USA, Hercules, Inc., Wilmington, DE, USA
Selection process for trade study: Reusable Hydrogen Composite Tank System (RHCTS)
Greenberg, H. S.; Sep 2, 1994; In English
Contract(s)/Grant(s): NCC8-39
Report No.(s): NASA-CR-196874; NAS 1.26:196874; SSD94D0209; No Copyright; Avail: CASI; A03, Hardcopy

This document describes the selection process that will be used to identify the most suitable structural configuration option for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 degree inclination. The most suitable RHCTS is within this configuration and will be the prototype design for subsequent design and analysis and the basis for the design and fabrication of a scale test article to be subjected to life cycle testing. The selection process for this TA 1 trade study is the same as that for the TA 2 trade study. As the trade study progresses additional insight may result in modifications to the selection criteria within in this process. Such modifications will result in an update of this document as appropriate.
Author
Circular Orbits; Design Analysis; Horizontal Spacecraft Landing; Hydrogen Fuels; Prototypes; Single Stage to Orbit Vehicles

19950005390 Rockwell International Corp., Downey, CA, USA, Grumman Aerospace Corp., Bethpage, NY, USA
Selection process for trade study: Graphite Composite Primary Structure (GCPS)
Greenberg, H. S.; Sep 2, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-196866; NAS 1.26:196866; SSD94D0210; No Copyright; Avail: CASI; A03, Hardcopy

This TA 2 document describes the selection process that will be used to identify the most suitable structural configuration for an SSTO winged vehicle capable of delivering 25,000 lbs to a 220 nm circular orbit at 51.6 degree inclination. The most suitable unpressurized graphite composite structures and material selections is within this configuration and will be the prototype design for subsequent design and analysis and the basis for the design and fabrication of payload bay, wing, and thrust structure full scale test articles representing segments of the prototype structures. The selection process for this TA 2 trade study is the same as that for the TA 1 trade study. As the trade study progresses additional insight may result in modifications to the selection criteria within this process. Such modifications will result in an update of this document as appropriate.
Author
Composite Structures; Fabrication; Full Scale Tests; Graphite; Propulsion System Configurations; Single Stage to Orbit Vehicles; Structural Design; Thrust-Weight Ratio

19950005389 Rockwell International Corp., Downey, CA, USA, Hercules, Inc., Wilmington, DE, USA
Test plan. Task 5, subtask 5.2: Early on-orbit TPSdebris impact tests
Greenberg, H. S.; Jul 29, 1994; In English
Contract(s)/Grant(s): NCC8-39
Report No.(s): NASA-CR-196873; NAS 1.26:196873; SSD94D0221; RHCTS-TP-1; No Copyright; Avail: CASI; A03, Hardcopy

The limitation of damage to, and survival of, the cryogenic tankage during the on-orbit stay despite potential impact of orbital debris, may be a significant discriminator in the RHCTS trade studies described in the TA-1 trade study plan (ref.

RHCTS-TSP-1) dated July 29, 1994. The objective of this early phase of an overall debris impact test program is to provide the data to support assessment of the relative suitability of integral and non integral tanks.
Author
Cryogenic Rocket Propellants; Damage; Fuel Tanks; Hydrogen Fuels; Impact Tests; Single Stage to Orbit Vehicles; Survival

19950005364 Rockwell International Corp., Downey, CA, USA, Northrop Corp., Los Angeles, CA, USA, Hercules Aerospace Co., Wilmington, DE, USA
Requirements report for SSTO vertical take-off/horizontal landing vehicle
Greenberg, H. S.; Jul 29, 1994; In English
Contract(s)/Grant(s): NCC1-193; NCC2-9003; NCC8-39
Report No.(s): NASA-CR-196875; NAS 1.26:196875; SSD94D0217; SSTO-REQ-1; No Copyright; Avail: CASI; A04, Hardcopy

This document describes the detailed design requirements and design criteria to support Structures/TPS Technology development for SSTO winged vehicle configurations that use vertical take-off and horizontal landing and deliver 25,000 lb payloads to a 220 nm circular orbit at an inclination of 51.6 degrees or 40,000 lb payloads to a 150 nm circular orbit at a 28.5 degree of inclination. This document will be updated on a timely basis as informatIon becomes available throughout the project.
Author
Design Analysis; Horizontal Spacecraft Landing; Reusable Launch Vehicles; Single Stage to Orbit Vehicles

19950005354 Rockwell International Corp., Downey, CA, USA, Northrop Corp., Los Angeles, CA, USA
Test plan. GCPS task 7, subtask 7.1: IHM development
Greenberg, H. S.; Sep 2, 1994; In English
Contract(s)/Grant(s): NCC1-193
Report No.(s): NASA-CR-106876; NAS 1.26:196876; SSD940212; No Copyright; Avail: CASI; A03, Hardcopy

The overall objective of Task 7 is to identify cost-effective life cycle integrated health management (IHM) approaches for a reusable launch vehicle's primary structure. Acceptable IHM approaches must: eliminate and accommodate faults through robust designs, identify optimum inspection/maintenance periods, automate ground and on-board test and check-out, and accommodate and detect structural faults by providing wide and localized area sensor and test coverage as required. These requirements are elements of our targeted primary structure low cost operations approach using airline-like maintenance by exception philosophies. This development plan will follow an evolutionary path paving the way to the ultimate development of flight-quality production, operations, and vehicle systems. This effort will be focused on maturing the recommended sensor technologies required for localized and wide area health monitoring to a technology readiness level (TRL) of 6 and to establish flight ready system design requirements. The following is a brief list of IHM program objectives: design out faults by analyzing material properties, structural geometry, and load and environment variables and identify failure modes and damage tolerance requirements; design in system robustness while meeting performance objectives (weight limitations) of the reusable launch vehicle primary structure; establish structural integrity margins to preclude the need for test and checkout and predict optimum inspection/maintenance periods through life prediction analysis; identify optimum fault protection system concept definitions combining system robustness and integrity margins established above with cost effective health monitoring technologies; and use coupons, panels, and integrated full scale primary structure test articles to identify, evaluate, and characterize the preferred NDE/NDI/IHM sensor technologies that will be a part of the fault protection system.
Derived from text
Fault Tolerance; Nondestructive Tests; Single Stage to Orbit Vehicles; Spacecraft Maintenance; Spacecraft Reliability; Structural Reliability

19950005173 McDonnell-Douglas Aerospace, Huntington Beach, CA, USA
Slush hydrogen technology program
Cady, Edwin C.; Sep 9, 1994; In English
Contract(s)/Grant(s): NAS3-25420; RTOP 763-22-2T
Report No.(s): AD-A284706; NASA-CR-195353; E-9003; NAS 1.26:195353; MDC-94-H00684; No Copyright; Avail: CASI; A04, Hardcopy

A slush hydrogen (SH2) technology facility (STF) was designed, fabricated, and assembled by a contractor team of McDonnell Douglas Aerospace (MDA), Martin Marietta Aerospace Group (MMAG), and Air Products and Chemicals, Inc.

(APCI). The STF consists of a slush generator which uses the freeze-thaw production process, a vacuum subsystem, a test tank which simulates the NASP vehicle, a triple point hydrogen receiver tank, a transfer subsystem, a sample bottle, a pressurization system, and a complete instrumentation and control subsystem. The STF was fabricated, checked-out, and made ready for testing under this contract. The actual SH2 testing was performed under the NASP consortium following NASP teaming. Pre-STF testing verified SH2 production methods, validated special SH2 instrumentation, and performed limited SH2 pressurization and expulsion tests which demonstrated the need for gaseous helium pre-pressurized of SH2 to control pressure collapse. The STF represents cutting-edge technology development by an effective Government-Industry team under very tight cost and schedule constraints.
Author
Cryogenics; Liquid Hydrogen; National Aerospace Plane Program; Research Facilities; Slush Hydrogen; Test Facilities

19950004141 NASA, Washington, DC, USA
Aero-Space Plane: Flexible access to space
Aug 1, 1991; In English
Report No.(s): NASA-TM-109904; NONP-NASA-VT-94-23146; ASR-257; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

The most recently designed X-30 (National Aerospace Plane) is described. The video feature also chronicles the development of the X-plane series, beginning with the X-1.
CASI
Aerospace Planes; National Aerospace Plane Program; X-31 Aircraft

19950002971 Rockwell International Corp., Downey, CA, USA
Propulsion module design review. Milestone 8: Integrated Propulsion Technology Demonstrator
Rogers, James P.; Vesperman, Curtis L.; Apr 17, 1995; In English
Contract(s)/Grant(s): NCC8-47
Report No.(s): NASA-CR-197998; NAS 1.26:197998; SSD95D0173; No Copyright; Avail: CASI; A10, Hardcopy

The propulsion module for the Integrated Propulsion Technology Demonstrator (IPTD) consists of main engine(s), and fluid, electrical, propulsion checkout and control systems (PCCS), and associated instrumentation representative of an X-33 main propulsion system (MPS). These systems are integrated within a structural support fixture which includes mounting provisions for the MPS feed lines and ancillary equipment. The weight of the main engines and associated thrust loads are reacted by a separated thrust structure mounted directly to the large structural beams (i.e., Norman Beams) of the Advanced Engine Test Facility (AETF). This document provides a summary of the design requirements, and associated design features of the major elements of the propulsion module. Significant design detail is provided for the hardware that will be procured, fabricated, assembled and delivered to MSFC in October 1995. Designs of hardware items due for fabrication at a later date (such as the thrust structure and tank modules) are presented only to the degree necessary to indicate critical interface requirements. This report also documents the findings of an independent peer of the design to ensure that major design oversights are caught early in this design and fabrication process. This peer review board consisted of senior propulsion, control, and software engineers at Rockwell who had avionics/ground control and data handling system or cryogenic design experience on the Saturn launch vehicle and/or the Shuttle.
Derived from text
Design Analysis; Liquid Propellant Rocket Engines; Propulsion System Configurations; Propulsion System Performance; Recoverable Launch Vehicles; Rocket Engine Design; Test Stands

19950002820 Rockwell International Corp., Downey, CA, USA
Reusable launch vehicle reliability, maintainability and operability assessment
Weber, T. F., Jr.; Mar 15, 1995; In English
Contract(s)/Grant(s): NAS8-39210
Report No.(s): NASA-CR-197695; NAS 1.26:197695; SSD95D0123; No Copyright; Avail: CASI; A03, Hardcopy

Rockwell's Space Systems Division was funded to conduct thorough reliability, maintainability and operability (RM&O) assessments for 3 NASA conceptually-conceived SSTO vehicles: a Winged Body (WB001), a Vertical Lander (VL001) and, at some future time when input data becomes available, a lifting body. Rockwell was instructed to use their existing 'generic' RM&O analysis models (MAtrix, SIMtrix, and STARSIM), and apply them to the RLV vehicles of interest. The RM&O assessments were to encompass, as a minimum, the following parameters: MTBM (mean time between maintenance), MTBR

(mean time between removal), MTBF (mean time between failure), MTBCF (mean time before critical failure), MTTR (mean time to repair), probability of mission success, turnaround time, crew size per repair, manhours per repair, manhours per mission, flight rate capability, availability, downtime per mission, manhours per processing flow (scheduled and unscheduled), and facility utilization. Additionally, the MAtrix model was to be exercised for the existing Space Shuttle and its infrastructure in order to judge the 'reasonableness' of the SSTO RM&O estimates, and to provide a basis for comparing these SSTO estimates to 'observed' Shuttle experiences. Fully operable Reliability and Maintainability Models for WB001, VL001 and the Space Shuttle were electronically transmitted to MSFC on December 13, 1994.
Derived from text
Maintainability; Reliability; Single Stage to Orbit Vehicles; Spacecraft Maintenance

19950002293 Rockwell International Corp., Downey, CA, USA
Lightweight durable TPS: Milestone 4
Fleming, M. M.; Greenberg, H. S.; Nov 15, 1994; In English
Contract(s)/Grant(s): NCC2-9003
Report No.(s): NASA-CR-197103; NAS 1.26:197103; SSD94D0336; No Copyright; Avail: CASI; A03, Hardcopy

This document, covering the TA3-Lightweight Durable Thermal Protection System, concerns the blanket assembly, the evaluation plan of the hook and loop (Velcro-type) fasteners, and a report relating the adverse environmental effects of the SSTO (Single Stage To Orbit) vehicle and their relationship to TPS components and locations. These adverse environmental effects concern only those part of the natural environment and not induced environmental effects, such as, vibro-acoustics, aerothermal heating, or aerodynamic loads.
CASI
Blankets; Chemical Analysis; Durability; Environment Effects; Fasteners; Materials Tests; Salt Spray Tests; Thermal Analysis; Thermal Protection

19940032138 NASA Lewis Research Center, Cleveland, OH, USA
Hot dynamic test rig for measuring hypersonic engine seal flow and durability
Miller, Jeffrey H.; Steinetz, Bruce M.; Sirocky, Paul J.; Kren, Lawrence A.; Feb 1, 1994; In English
Report No.(s): NASA-TM-106507; E-8031-1; NAS 1.15:10657; No Copyright; Avail: CASI; A03, Hardcopy

A test fixture for measuring the dynamic performance of candidate high-temperature engine seal concepts was developed. The test fixture was developed to evaluate seal concepts under development for advanced hypersonic engines, such as those being considered for the National Aerospace Plane (NASP). The fixture can measure dynamic seal leakage performance from room temperature up to 840 C and air pressure differentials of to 0.7 MPa. Performance of the seals can be measured while sealing against flat or engine-simulated distorted walls. In the fixture, two seals are preloaded against the sides of a 0.3 m long saber that slides transverse to the axis of the seals, simulating the scrubbing motion anticipated in these engines. The capabilities of this text fixture along with preliminary data showing the dependence of seal leakage performance on high temperature cycling are covered.
Author (revised)
Aerospace Planes; Dynamic Tests; Engine Design; Engine Tests; High Temperature; Hypersonic Flow; Hypersonics; Leakage; Propulsion System Performance; Seals (Stoppers); Temperature Effects; Thermal Cycling Tests

19940029244 NASA Lewis Research Center, Cleveland, OH, USA
National aerospace plane
Jul 1, 1990; In English
Report No.(s): LERC-4002; NASA-TM-109843; NONP-NASA-VT-94-13533; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

This video concentrates on materials being developed and tested at LeRC for possible use in NASP.
CASI
Aerospace Planes; Aircraft Construction Materials; National Aerospace Plane Program; Spacecraft Construction Materials

19940029064 NASA John C. Stennis Space Center, Bay Saint Louis, MS, USA
High Heat Flux Facility
JAN 1, 1993; In English
Report No.(s): NASA-TM-109834; NONP-NASA-VT-94-12962; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

This video gives an overview of the High Heat Flux Facility being built at Stennis Space Center in conjunction with Wright-Patterson Air Force Base. This facility will simulate flight heat conditions and will be used to test engine and materials for the National Aerospace Plane.

CASI

Flight Conditions; Heat Flux; National Aerospace Plane Program; Test Facilities

19940029061 NASA John C. Stennis Space Center, Bay Saint Louis, MS, USA

John C. Stennis Space Center overview

May 1, 1994; In English

Report No.(s): NASA-TM-109816; NONP-NASA-VT-94-12944; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

An overview of research being conducted at the John C. Stennis Space Center is given. The Space Center is not only a NASA Space Flight Center, but also houses facilities for 22 other governmental agencies. The programs described are Stennis' High Heat Flux Facility, the Component Test Facility (used to test propulsion rockets and for the development of the National Aerospace Plane), oceanographic and remote sensing research, and contributions to the development of Space Station Freedom.

CASI

National Aerospace Plane Program; Research Facilities; Space Station Freedom; Test Facilities

19940028594 Rockwell International Corp., Huntsville, AL, USA

Advanced transportation system study: Manned launch vehicle concepts for two way transportation system payloads to LEO

Duffy, James B.; JAN 1, 1993; In English

Contract(s)/Grant(s): NAS8-39207

Report No.(s): NASA-CR-193951; NAS 1.26:193951; SSD93M0025-1; No Copyright; Avail: CASI; A25, Hardcopy

The purpose of the Advanced Transportation System Study (ATSS) task area 1 study effort is to examine manned launch vehicle booster concepts and two-way cargo transfer and return vehicle concepts to determine which of the many proposed concepts best meets NASA's needs for two-way transportation to low earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine the integrated booster/spacecraft performance, operations, reliability, and cost characteristics for the payload delivery and return mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts. The booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles and several variations of the NLS reference configurations to deliver larger manned payload concepts (such as the crew logistics vehicle (CLV) proposed by NASA JSC). Advanced, clean sheet concepts such as an F-1A engine derived liquid rocket booster (LRB), the single stage to orbit rocket, and a NASP-derived aerospace plane were also included in the study effort. Existing expendable launch vehicles such as the Titan 4, Ariane 5, Energia, and Proton were also examined. Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the personnel launch system (PLS). A scaled up version of the PLS for combined crew/cargo delivery capability, the HL-42 configuration, was also included in the analyses of cargo transfer and return vehicle (CTRV) booster concepts. In addition to strictly manned payloads, two-way cargo transportation systems (CTRV's) were also examined. The study provided detailed design and analysis of the performance, reliability, and operations of these concepts. The study analyzed these concepts as unique systems and also analyzed several combined CTRV/booster configurations as integrated launch systems (such as for launch abort analyses). Included in the set of CTRV concepts analyzed were the medium CTRV, the integral CTRV (in both a pressurized and unpressurized configuration), the winged CTRV, and an attached cargo carrier for the PLS system known as the PLS caboose.

Author (revised)

Launch Vehicle Configurations; Launch Vehicles; Manned Space Flight; Manned Spacecraft; Space Logistics; Space Transportation System; Spacecraft Launching

19940027298 NASA Lewis Research Center, Cleveland, OH, USA
High temperature NASP engine seal development
JAN 1, 1992; In English
Report No.(s): LERC-92-174; NASA-TM-109750; NONP-NASA-VT-94-9950; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

This video details research being conducted at the Lewis Research Center on high temperature engine seal design for the National Aerospace Plane. To maximize the speed, the jets on the NASP extract oxygen from the air rather than carry large liquid fuel tanks; this creates temperatures within the jet of over 5000 F. To prevent these potentially explosive gases from escaping, researchers are developing new technologies for use in the engine seals. Two examples explained are the ceramic wafer seal and the braided ceramic rope seal. Computer simulations and laboratory footage are used to illustrate the workings of these seals. Benefits for other aerospace and industrial applications, as well as for the space shuttle, are explored.
CASI
Aerospace Planes; Ceramics; Engine Parts; High Temperature; National Aerospace Plane Program; Refractory Materials; Seals (Stoppers)

19940023286 Maryland Univ., College Park, MD, USA
A parametric sensitivity study for single-stage-to-orbit hypersonic vehicles using trajectory optimization
Lovell, T. Alan; Schmidt, D. K.; Mar 21, 1994; In English
Contract(s)/Grant(s): NAG1-1540
Report No.(s): NASA-CR-195703; NAS 1.26:195703; No Copyright; Avail: CASI; A03, Hardcopy

The class of hypersonic vehicle configurations with single stage-to-orbit (SSTO) capability reflect highly integrated airframe and propulsion systems. These designs are also known to exhibit a large degree of interaction between the airframe and engine dynamics. Consequently, even simplified hypersonic models are characterized by tightly coupled nonlinear equations of motion. In addition, hypersonic SSTO vehicles present a major system design challenge; the vehicle's overall mission performance is a function of its subsystem efficiencies including structural, aerodynamic, propulsive, and operational. Further, all subsystem efficiencies are interrelated, hence, independent optimization of the subsystems is not likely to lead to an optimum design. Thus, it is desired to know the effect of various subsystem efficiencies on overall mission performance. For the purposes of this analysis, mission performance will be measured in terms of the payload weight inserted into orbit. In this report, a trajectory optimization problem is formulated for a generic hypersonic lifting body for a specified orbit-injection mission. A solution method is outlined, and results are detailed for the generic vehicle, referred to as the baseline model. After evaluating the performance of the baseline model, a sensitivity study is presented to determine the effect of various subsystem efficiencies on mission performance. This consists of performing a parametric analysis of the basic design parameters, generating a matrix of configurations, and determining the mission performance of each configuration. Also, the performance loss due to constraining the total head load experienced by the vehicle is evaluated. The key results from this analysis include the formulation of the sizing problem for this vehicle class using trajectory optimization, characteristics of the optimal trajectories, and the subsystem design sensitivities.
Author (revised)
Airframes; Equations of Motion; Hypersonic Vehicles; Nonlinear Equations; Parameter Identification; Propulsion System Configurations; Propulsion System Performance; Single Stage to Orbit Vehicles; Trajectory Optimization; Trajectory Planning

19940022648 NASA, Washington, DC, USA
Access to space study
Jan 1, 1994; In English
Report No.(s): NASA-TM-109693; NAS 1.15:109693; No Copyright; Avail: CASI; A05, Hardcopy

This report summarizes the results of a comprehensive NASA in-house study to identify and assess alternate approaches to access to space through the year 2030, and to select and recommend a preferred cause of action. The goals of the study were to identify the best vehicles and transportation architectures to make major reductions in the cost of space transportation (at least 50%), while at the same time increasing safety for flight crews by at least an order of magnitude. In addition, vehicle reliability was to exceed 0.98 percent, and, as important, the robustness, pad time, turnaround time, and other aspects of operability were to be vastly improved. This study examined three major optional architectures: (1) retain and upgrade the Space Shuttle and expendable launch vehicles; (2) develop new expendable vehicles using conventional technologies and transition from current vehicles beginning in 2005; and (3) develop new reusable vehicles using advanced technology, and transition from current vehicles beginning in 2008. The launch-needs, mission model utilized for for the study was based upon

today's projection of civil, defense, and commercial mission payload requirements.
Derived from text
Expendable Stages (Spacecraft); Launch Vehicles; Mission Planning; Reliability Analysis; Single Stage to Orbit Vehicles; Space Commercialization; Space Missions; Space Shuttles; Space Transportation; Spacecraft Launching

19940020603 Ohio State Univ., Columbus, OH, USA
A conceptual design of an unmanned test vehicle using an airbreathing propulsion system
JAN 1, 1992; In English
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-195550; NAS 1.26:195550; No Copyright; Avail: CASI; A04, Hardcopy

According to Aviation Week and Space Technology (Nov. 16, 1992), without a redefined approach to the problem of achieving single stage-to-orbit flight, the X-30 program is virtually assured of cancellation. One of the significant design goals of the X-30 program is to achieve single stage to low-earth orbit using airbreathing propulsion systems. In an attempt to avoid cancellation, the NASP Program has decided to design a test vehicle to achieve these goals. This report recommends a conceptual design of an unmanned test vehicle using an airbreathing propulsion system.
Author (revised)
Aerospace Planes; Air Breathing Engines; Aircraft Design; National Aerospace Plane Program; Pilotless Aircraft; Propulsion System Performance; Research Aircraft; Single Stage to Orbit Vehicles

19940020118 Ohio State Univ., Columbus, OH, USA
Configuration development study of the OSU 1 hypersonic research vehicle
Stein, Matthew D.; Frankhauser, Chris; Zee, Warner; Kosanchick, Melvin, III; Nelson, Nick; Hunt, William; May 31, 1993; In English
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-195522; NAS 1.26:195522; No Copyright; Avail: CASI; A08, Hardcopy

In an effort to insure the future development of hypersonic cruise aircraft, the possible vehicle configurations were examined to develop a single-stage-to-orbit hypersonic research vehicle (HRV). Based on the needs of hypersonic research and development, the mission goals and requirements are determined. A body type is chosen. Three modes of propulsion and two liquid rocket fuels are compared, followed by the optimization of the body configuration through aerodynamic, weight, and trajectory studies. A cost analysis is included.
Author (revised)
Aircraft Configurations; Hydrocarbon Fuels; Hydrogen Fuels; Hypersonic Vehicles; Research Vehicles; Single Stage to Orbit Vehicles; Thermal Protection

19940017090 NASA Lewis Research Center, Cleveland, OH, USA
Thrust augmentation options for the Beta 2 two-stage-to-orbit vehicle
Snyder, Christopher A.; Dec 1, 1993; In English; AIAA Aircraft Design, Systems and Operations Meeting, 11-13 Aug. 1993, Monterey, CA, USA
Contract(s)/Grant(s): RTOP 505-70-00
Report No.(s): NASA-TM-106448; E-8301; NAS 1.15:106448; AIAA PAPER 93-4014; No Copyright; Avail: CASI; A03, Hardcopy

NASA LeRC is continuing to study propulsion concepts for a horizontal takeoff and landing, fully reusable, two-stage-to-orbit vehicle. This will be capable of launching and returning a 10,000 pound payload to a 100 nautical mile polar orbit using low-risk technology. The vehicle, Beta 2, is a derivative of the USAF/Boeing Beta vehicle which was designed to deliver a 50,000 pound payload to a similar orbit. Beta 2 stages at Mach 6.5 and about 100,000 ft altitude. The propulsion system for the booster is an over/under turbine bypass engine/ramjet configuration. In this paper, several options for thrust augmentation were studied in order to improve the performance of this engine where there was a critical need. Options studies were turbine engine overspeed in the transonic region, water injection at a various turbine engine locations also during the transonic region, and water injection at the turbine engine face during high speed operation. The methodology, constraints, propulsion performance, and mission study results are presented.
Author (revised)
Aerospace Planes; Air Breathing Boosters; Gas Turbine Engines; Hypersonic Speed; Propulsion System Configurations; Propulsive Efficiency; Thrust Augmentation; Water Injection

19940012847 NASA Lyndon B. Johnson Space Center, Houston, TX, USA
Human Transportation System (HTS) study, volume 2
Lance, N.; Geyer, M. S.; Gaunce, M. T.; Oct 1, 1993; In English
Contract(s)/Grant(s): RTOP 906-11-01-01
Report No.(s): NASA-TM-104779-VOL-2; S-739; NAS 1.15:104779-VOL-2; No Copyright; Avail: CASI; A24, Hardcopy

This report summarizes work completed under the Human Transportation System Study. This study was conducted by the New Initiatives Office at JSC with the technical support of Boeing, General Dynamics, Lockheed, McDonnell-Douglas, Martin Marietta, and Rockwell. The study was designed to generate information on determining the appropriate path to follow for new system development to meet the Nation's space transportation needs. The study evaluates 18 transportation architecture options using a parametric set of mission requirements. These options include use of current systems (e.g., Shuttle, Titan, etc.) as well as proposed systems (e.g., PLS, Single-Stage-to-Orbit, etc.) to assess the impact of various considerations, such as the cost of alternate access, or the benefit of separating people and cargo. The architecture options are compared to each other with six measurable evaluation criteria or attributes. They are: funding profile, human safety, probability of mission success, architecture cost risk, launch schedule confidence, and environmental impact. Values for these attributes are presented for the architecture options, with pertinent conclusions and recommendations.

Derived from text

Human Factors Engineering; Manned Space Flight; Manned Spacecraft; Mission Planning; Space Transportation; Systems Engineering

19940012582 NASA Langley Research Center, Hampton, VA, USA
Aerothermoelastic analysis of a NASP demonstrator model
Heeg, Jennifer; Zeiler, Thomas A.; Pototzky, Anthony S.; Spain, Charles V.; Engelund, Walter C.; Oct 1, 1993; In English
Contract(s)/Grant(s): RTOP 505-63-50-15
Report No.(s): NASA-TM-109007; NAS 1.15:109007; No Copyright; Avail: CASI; A03, Hardcopy

The proposed National Aerospace Plane (NASP) is designed to travel at speeds up to Mach 25. Because aerodynamic heating during high-speed flight through the atmosphere could destiffen a structure, significant couplings between the elastic and rigid body modes could result in lower flutter speeds and more pronounced aeroelastic response characteristics. These speeds will also generate thermal loads on the structure. The purpose of this research is to develop methodologies applicable to the NASP and to apply them to a representative model to determine its aerothermoelastic characteristics when subjected to these thermal loads. This paper describes an aerothermoelastic analysis of the generic hypersonic vehicle configuration. The steps involved in this analysis were: generating vehicle surface temperatures at the appropriate flight conditions; applying these temperatures to the vehicle's structure to predict changes in the stiffness resulting from material property degradation; predicting the vibration characteristics of the heated structure at the various temperature conditions; performing aerodynamic analyses; and conducting flutter analysis of the heated vehicle. Results of these analyses and conclusions representative of a NASP vehicle are provided in this paper.

Author

Aerodynamic Heating; Aerodynamic Stability; Aerospace Planes; Aerothermoelasticity; Dynamic Response; Hypersonic Vehicles; Mach Number; Supersonic Flutter; Temperature Distribution

19940010851 NASA, Washington, DC, USA
National Aero-Space Plane resource reel
Aug 1, 1991; In English
Report No.(s): NASA-TM-109451; NONP-NASA-VT-93-190248; No Copyright; Avail: CASI; B02, Videotape-Beta; V02, Videotape-VHS

This document presents a series of takes and sequences of model photography of the 1991 NASP design.

CASI

Aircraft Models; National Aerospace Plane Program; Photography

19940010850 NASA, Washington, DC, USA
National Aero-Space Plane
Jul 1, 1990; In English
Report No.(s): ASR-253; NASA-TM-109450; NONP-NASA-VT-93-190247; No Copyright; Avail: CASI; B01, Videotape-Beta; V01, Videotape-VHS

This document presents updated model photography of ‘old’ NASP design.
CASI
Aircraft Models; National Aerospace Plane Program; Photography

19940009631 Iowa State Univ. of Science and Technology, Ames, IA, USA
Trajectory optimization for the national aerospace plane
Lu, Ping; Jul 1, 1993; In English
Contract(s)/Grant(s): NAG1-1255
Report No.(s): NASA-CR-194146; NAS 1.26:194146; No Copyright; Avail: CASI; A03, Hardcopy

During the past six months the research objectives outlined in the last semi-annual report were accomplished. Specifically, these are: three-dimensional (3-D) fuel-optimal ascent trajectory of the aerospace plane and the effects of thrust vectoring control (TVC) on the fuel consumption and trajectory shaping were investigated; the maximum abort landing area (footprint) was studied; preliminary assessment of simultaneous design of the ascent trajectory and the vehicle configuration for the aerospace plane was also conducted. The work accomplished in the reporting period is summarized.
Author (revised)
Aerospace Planes; Ascent Trajectories; Fuel Consumption; National Aerospace Plane Program; Trajectory Optimization

19940009145 North Carolina State Univ., Raleigh, NC, USA
Multidisciplinary design techniques applied to conceptual aerospace vehicle design
Olds, John Robert; Walberg, Gerald D.; JAN 1, 1993; In English
Contract(s)/Grant(s): NCC1-168
Report No.(s): NASA-CR-194409; NAS 1.26:194409; No Copyright; Avail: CASI; A09, Hardcopy

Multidisciplinary design optimization (MDO) is an emerging discipline within aerospace engineering. Its goal is to bring structure and efficiency to the complex design process associated with advanced aerospace launch vehicles. Aerospace vehicles generally require input from a variety of traditional aerospace disciplines - aerodynamics, structures, performance, etc. As such, traditional optimization methods cannot always be applied. Several multidisciplinary techniques and methods were proposed as potentially applicable to this class of design problem. Among the candidate options are calculus-based (or gradient-based) optimization schemes and parametric schemes based on design of experiments theory. A brief overview of several applicable multidisciplinary design optimization methods is included. Methods from the calculus-based class and the parametric class are reviewed, but the research application reported focuses on methods from the parametric class. A vehicle of current interest was chosen as a test application for this research. The rocket-based combined-cycle (RBCC) single-stage-to-orbit (SSTO) launch vehicle combines elements of rocket and airbreathing propulsion in an attempt to produce an attractive option for launching medium sized payloads into low earth orbit. The RBCC SSTO presents a particularly difficult problem for traditional one-variable-at-a-time optimization methods because of the lack of an adequate experience base and the highly coupled nature of the design variables. MDO, however, with it’s structured approach to design, is well suited to this problem. The result of the application of Taguchi methods, central composite designs, and response surface methods to the design optimization of the RBCC SSTO are presented. Attention is given to the aspect of Taguchi methods that attempts to locate a ‘robust’ design - that is, a design that is least sensitive to uncontrollable influences on the design. Near-optimum minimum dry weight solutions are determined for the vehicle. A summary and evaluation of the various parametric MDO methods employed in the research are included. Recommendations for additional research are provided.
Author (revised)
Aerospace Vehicles; Design Analysis; Optimization; Single Stage to Orbit Vehicles

19940008783 NASA Hugh L. Dryden Flight Research Facility, Edwards, CA, USA
Perspective on the National Aero-Space Plane Program instrumentation development
Bogue, Rodney K.; Erbland, Peter; May 1, 1993; In English; Aero Space Planes Conference, 1-3 Dec. 1992, Orlando, FL, USA
Contract(s)/Grant(s): RTOP 763-21-41
Report No.(s): NASA-TM-4505; H-1916; NAS 1.15:4505; Copyright; Avail: CASI; A03, Hardcopy

A review of the requirement for, and development of, advanced measurement technology for the National Aerospace Plane program is presented. The objective is to discuss the technical need and the program commitment required to ensure that adequate and timely measurement capabilities are provided for ground and flight testing in the NASP program. The scope of the measurement problem is presented, the measurement process is described, how instrumentation technology

development has been affected by NASP program evolution is examined, the national effort to define measurement requirements and assess the adequacy of current technology to support the NASP program is discussed, and the measurement requirements are summarized. The unique features of the NASP program that complicate the understanding of requirements and the development of viable solutions are illustrated.
Author (revised)
Aerospace Planes; Measuring Instruments; National Aerospace Plane Program; Spacecraft Instruments

19940008377 NASA Langley Research Center, Hampton, VA, USA
A method of predicting quasi-steady aerodynamics for flutter analysis of high speed vehicles using steady CFD calculations
Scott, Robert C.; Pototzky, Anthony S.; Jul 1, 1993; In English; AIAA Structures, Structural Dynamics, and Materials Conference,, 19-21 Apr. 1993, LaJolla, CA, USA
Contract(s)/Grant(s): RTOP 505-63-50-15
Report No.(s): NASA-TM-109009; NAS 1.15:109009; No Copyright; Avail: CASI; A02, Hardcopy

High speed linear aerodynamic theories like piston theory and Newtonian impact theory are relatively inexpensive to use for flutter analysis. These theories have limited areas of applicability depending on the configuration and the flow conditions. In addition, these theories lack the ability to capture viscous, shock, and real gas effects. CFD methods can model all of these effects accurately, but the unsteady calculations required for flutter are expensive and often impractical. This paper describes a method for using steady CFD calculations to approximate the generalized aerodynamic forces for a flutter analysis. Example two-and three-dimensional aerodynamic force calculations are provided. In addition, a flutter analysis of a NASP-type wing will be discussed.
Author (revised)
Aerodynamic Characteristics; Aerodynamic Forces; Aerospace Planes; Computational Fluid Dynamics; Dynamic Structural Analysis; Flutter Analysis; Prediction Analysis Techniques; Unsteady Aerodynamics; Wings

19940006270 California Univ., Los Angeles, CA, USA
Lee waves: Benign and malignant
Wurtele, M. G.; Datta, A.; Sharman, R. D.; Jun 1, 1993; In English
Contract(s)/Grant(s): NCC2-0374; RTOP 505-68-50
Report No.(s): NASA-CR-186024; H-1890; NAS 1.26:186024; No Copyright; Avail: CASI; A03, Hardcopy

The flow of an incompressible fluid over an obstacle will produce an oscillation in which buoyancy is the restoring force, called a gravity wave. For disturbances of this scale, the atmosphere may be treated as dynamically incompressible, even though there exists a mean static upward density gradient. Even in the linear approximation - i.e., for small disturbances - this model explains a great many of the flow phenomena observed in the lee of mountains. However, nonlinearities do arise importantly, in three ways: (1) through amplification due to the decrease of mean density with height; (2) through the large (scaled) size of the obstacle, such as a mountain range; and (3) from dynamically singular levels in the fluid field. These effects produce a complicated array of phenomena - large departure of the streamlines from their equilibrium levels, high winds, generation of small scales, turbulence, etc. - that present hazards to aircraft and to lee surface areas. The nonlinear disturbances also interact with the larger-scale flow in such a manner as to impact global weather forecasts and the climatological momentum balance. If there is no dynamic barrier, these waves can penetrate vertically into the middle atmosphere (30-100 km), where recent observations show them to be of a length scale that must involve the coriolis force in any modeling. At these altitudes, the amplitude of the waves is very large, and the phenomena associated with these wave dynamics are being studied with a view to their potential impact on high performance aircraft, including the projected National Aerospace Plane (NASP). The presentation shows the results of analysis and of state-of-the-art numerical simulations, validated where possible by observational data, and illustrated with photographs from nature.
Author (revised)
Aircraft Hazards; Atmospheric Turbulence; Flow Distribution; Gravity Waves; Incompressible Fluids; Lee Waves

19940003888 VRA, Inc., Blacksburg, VA, USA
PNS predictions of external/internal hypersonic flows for NASP propulsion applications
Bhutta, Bilal A.; Lewis, Clark H.; Jun 1, 1990; In English
Contract(s)/Grant(s): NAS3-25450; SBIR-01.05-2036
Report No.(s): NASA-CR-194246; NAS 1.26:194246; VRA-TR-90-01; No Copyright; Avail: CASI; A10, Hardcopy

Because the development of National Aerospace Plane (NASP) propulsion technology and airframe design relies heavily upon computational fluid dynamics, a keen need exists for developing accurate, reliable, and computationally efficient numerical schemes for predicting large angle-of-attack, supersonic/hypersonic external flows over realistic (3-D) NASP configurations. In this SBIR Phase II study a new three dimensional fully iterative PNS scheme has been developed to study hypersonic external flows over 3-D lifting configurations. It has been demonstrated that this baseline PNS scheme can also be extended to predict 2-D/axisymmetric supersonic/hypersonic internal flows. The three dimensional PNS scheme developed is inherently stable in the subsonic as well as the supersonic flow regions and, thus, does not require any sublayer approximation. It has the capability of treating perfect-gas, equilibrium-air, and nonequilibrium-air gas models in a unified manner. A fourth-order crossflow and a second-order streamwise smoothing approach are used to damp the solution oscillations. In addition, a pseudo-unsteady approach is used to dramatically improve the solution efficiency without compromising the solution accuracy. The scheme is formulated in terms of a general curvilinear coordinate system and is capable of treating cylindrical, parabolic, body-normal, a modified body-normal, and elliptic grid-generation algorithms for highly nonaxisymmetric configurations. A new fully implicit and crossflow-coupled shock-fitting approach has also been developed. With this solution approach the shock is fully coupled in the crossflow direction, while within the shock-layer the solution is obtained using a new predictor-corrector solution scheme which can treat strong crossflow solution-coupling effects. The resulting overall computing times are of the order of the noniterative PNS schemes using approximate factorization, because apart from being numerically efficient the present solution scheme allows much larger marching steps to be taken. In addition to these mathematical and numerical developments, significant effort was spent in developing a vectorization strategy for this 3-D PNS scheme. Six cases were considered to test this new PNS scheme for external and internal hypersonic flows. The first three test cases were for the 5-deg angle of attack flow over a 3-D lifting configuration at Mach 20 and 125 kft altitude. Perfect-gas, equilibrium-air, and nonequilibrium-air gas models were used to study these cases, and it was observed that the type of gas model used had a significant influence on the flowfield predictions. The next three test cases considered 2-D supersonic/hypersonic internal flows through different channel configurations. In each of these internal flow cases, comparisons were made with the predictions obtained using classical 2-D inviscid theory, and the agreement between inviscid and viscous predictions was found to be excellent. In general, the results of these numerical tests were very encouraging and substantiated the accuracy, efficiency, and stability claims of the PNS scheme.
Author (revised)

Aerospace Planes; Channel Flow; Computational Fluid Dynamics; Grid Generation (Mathematics); Hypersonic Flow; Hypersonic Vehicles; Navier-Stokes Equation; Supersonic Flow; Three Dimensional Flow; Two Dimensional Flow

19930022451 Cornell Univ., Ithaca, NY, USA

Real-time trajectory optimization on parallel processors

Psiaki, Mark L.; Jul 10, 1993; In English

Contract(s)/Grant(s): NAG1-1009

Report No.(s): NASA-CR-193303; NAS 1.26:193303; No Copyright; Avail: CASI; A10, Hardcopy

A parallel algorithm has been developed for rapidly solving trajectory optimization problems. The goal of the work has been to develop an algorithm that is suitable to do real-time, on-line optimal guidance through repeated solution of a trajectory optimization problem. The algorithm has been developed on an INTEL iPSC/860 message passing parallel processor. It uses a zero-order-hold discretization of a continuous-time problem and solves the resulting nonlinear programming problem using a custom-designed augmented Lagrangian nonlinear programming algorithm. The algorithm achieves parallelism of function, derivative, and search direction calculations through the principle of domain decomposition applied along the time axis. It has been encoded and tested on 3 example problems, the Goddard problem, the acceleration-limited, planar minimum-time to the origin problem, and a National Aerospace Plane minimum-fuel ascent guidance problem. Execution times as fast as 118 sec of wall clock time have been achieved for a 128-stage Goddard problem solved on 32 processors. A 32-stage minimum-time problem has been solved in 151 sec on 32 processors. A 32-stage National Aerospace Plane problem required 2 hours when solved on 32 processors. A speed-up factor of 7.2 has been achieved by using 32-nodes instead of 1-node to solve a 64-stage Goddard problem.
Derived from text

Aerospace Planes; Lagrangian Function; Message Processing; Nonlinear Programming; On-Line Systems; Parallel Processing (Computers); Real Time Operation; Trajectory Optimization

19930017829 NASA Lewis Research Center, Cleveland, OH, USA

The design and evolution of the beta two-stage-to-orbit horizontal takeoff and landing launch system

Burkardt, Leo A.; Norris, Rick B.; Dec 1, 1992; In English; 4th International Aerospace Planes Conference, 1-4 Dec. 1992, Orlando, FL, USA

Contract(s)/Grant(s): RTOP 505-69-40

Report No.(s): NASA-TM-106118; E-7776; NAS 1.15:106118; AIAA PAPER 92-5080; No Copyright; Avail: CASI; A03, Hardcopy

The Beta launch system was originally conceived in 1986 as a horizontal takeoff and landing, fully reusable, two-stage-to-orbit, manned launch vehicle to replace the Shuttle. It was to be capable of delivering a 50,000 lb. payload to low polar orbit. The booster propulsion system consisted of JP fueled turbojets and LH fueled ramjets mounted in pods in an over/under arrangement, and a single LOX/LH fueled SSME rocket. The second stage orbiter, which staged at Mach 8, was powered by an SSME rocket. A major goal was to develop a vehicle design consistent with near term technology. The vehicle design was completed with a GLOW of approximately 2,000,000 lbs. All design goals were met. Since then, interest has shifted to the 10,000 lbs. to low polar orbit payload class. The original Beta was down-sized to meet this payload class. The GLOW of the down-sized vehicle was approximately 1,000,000 lbs. The booster was converted to exclusively air-breathing operation. Because the booster depends on conventional air-breathing propulsion only, the staging Mach number was reduced to 5.5. The orbiter remains an SSME rocket-powered stage.

Author

Air Breathing Engines; Air Launching; Horizontal Spacecraft Landing; Hydrogen Oxygen Engines; Hypersonic Vehicles; Multistage Rocket Vehicles; Ramjet Engines; Reusable Launch Vehicles; Space Shuttle Main Engine; Spacecraft Launching

19930017827 NASA Langley Research Center, Hampton, VA, USA

Analysis of the staging maneuver and booster glideback guidance for a two-stage, winged, fully reusable launch vehicle

Naftel, J. Christopher; Powell, Richard W.; Apr 1, 1993; In English

Contract(s)/Grant(s): RTOP 906-11-01-01

Report No.(s): NASA-TP-3335; L-17066; NAS 1.60:3335; No Copyright; Avail: CASI; A03, Hardcopy

One of the promising launch concepts that could replace the current space shuttle launch system is a two-stage, winged, vertical-takeoff, fully reusable launch vehicle. During the boost phase of ascent, the booster provides propellant for the orbiter engines through a cross-feed system. When the vehicle reaches a Mach number of 3, the booster propellants are depleted and the booster is staged and glides unpowered to a horizontal landing at a launch site runway. Two major design issues for this class of vehicle are the staging maneuver and the booster glideback. For the staging maneuver analysis, a technique was developed that provides for a successful separation of the booster from the orbiter over a wide range of staging angles of attack. A longitudinal flight control system was developed for control of the booster during the staging maneuver. For the booster glide back analysis, a guidance algorithm was developed that successfully guides the booster from the completion of the staging maneuver to a launch site runway while encountering many off-nominal atmospheric, aerodynamic, and staging conditions.

Author (revised)

Booster Rocket Engines; Control Systems Design; Multistage Rocket Vehicles; Reusable Launch Vehicles; Spacecraft Control; Spacecraft Launching; Spacecraft Maneuvers; Vertical Takeoff

19930017364 NASA Langley Research Center, Hampton, VA, USA

Supersonic aeroelastic instability results for a NASP-like wing model

Cole, Stanley R.; Florance, James R.; Thomason, Lee B.; Spain, Charles V.; Bullock, Ellen P.; Apr 1, 1993; In English; AIAA 34th SDM Conference, 19-23 Apr. 1993, Hampton, VA, USA

Contract(s)/Grant(s): RTOP 763-23-41

Report No.(s): NASA-TM-107739; NAS 1.15:107739; No Copyright; Avail: CASI; A03, Hardcopy

An experimental study and an analytical study have been conducted to examine static divergence for hypersonic-vehicle wing models at supersonic conditions. A supersonic test in the Langley Unitary Plan Wind Tunnel facility was conducted for two wind-tunnel models. These models were nearly identical with the exception of airfoil shape. One model had a four-percent maximum thickness airfoil and the other model had an eight-percent maximum thickness airfoil. The wing models had low-aspect ratios and highly swept leading edges. The all-movable wing models were supported by a single-pivot mechanism along the wing root. For both of the wind-tunnel models, configuration changes could be made in the wing-pivot location along the wing root and in the wing-pivot pitch stiffness. Three divergence conditions were measured for the four-percent thick airfoil model in the Mach number range of 2.6 to 3.6 and one divergence condition was measured for the eight-percent thick

airfoil model at a Mach number of 2.9. Analytical divergence calculations were made for comparison with experimental results and to evaluate the parametric effects of wing-pivot stiffness, wing-pivot location, and airfoil thickness variations. These analyses showed that decreasing airfoil thickness, moving the wing-pivot location upstream, or increasing the pitch-pivot stiffness have the beneficial effect of increasing the divergence dynamic pressures. The calculations predicted the trend of experimental divergence dynamic pressure with Mach number accurately; however, the calculations were approximately 25 percent conservative with respect to dynamic pressure.
Author
Aeroelasticity; Aerospace Planes; Dynamic Stability; Hypersonic Vehicles; Low Aspect Ratio Wings; Supersonic Speed; Swept Wings; Wind Tunnel Tests

19930016481 Iowa State Univ. of Science and Technology, Ames, IA, USA
Trajectory optimization for the National aerospace plane
Lu, Ping; Jan 1, 1993; In English
Contract(s)/Grant(s): NAG1-1255
Report No.(s): NASA-CR-192954; NAS 1.26:192954; No Copyright; Avail: CASI; A02, Hardcopy

While continuing the application of the inverse dynamics approach in obtaining the optimal numerical solutions, the research during the past six months has been focused on the formulation and derivation of closed-form solutions for constrained hypersonic flight trajectories. Since it was found in the research of the first year that a dominant portion of the optimal ascent trajectory of the aerospace plane is constrained by dynamic pressure and heating constraints, the application of the analytical solutions significantly enhances the efficiency in trajectory optimization, provides a better insight to understanding of the trajectory and conceivably has great potential in guidance of the vehicle. Work of this period has been reported in four technical papers. Two of the papers were presented in the AIAA Guidance, Navigation, and Control Conference (Hilton Head, SC, August, 1992) and Fourth International Aerospace Planes Conference (Orlando, FL, December, 1992). The other two papers have been accepted for publication by Journal of Guidance, Control, and Dynamics, and will appear in 1993. This report briefly summarizes the work done in the past six months and work currently underway.
Derived from text
Aerospace Planes; Hypersonic Flight; National Aerospace Plane Program; Spacecraft Guidance; Trajectory Optimization

19930014233 NASA Langley Research Center, Hampton, VA, USA
An overview of aeroelasticity studies for the National Aerospace Plane
Ricketts, Rodney H.; Noll, Thomas E.; Huttsell, Lawrence J.; Hutsell, Lawrence J.; Mar 1, 1993; In English; AIAA 34th SDM Conference,, 19-23 Apr. 1993, Hampton, VA, USA
Contract(s)/Grant(s): RTOP 763-23-41
Report No.(s): NASA-TM-107728; NAS 1.15:107728; No Copyright; Avail: CASI; A03, Hardcopy

The National Aero-Space Plane (NASP), or X-30, is a single-stage-to-orbit vehicle that is designed to takeoff and land on conventional runways. Research in aeroelasticity was conducted by NASA and the Wright Laboratory to support the design of a flight vehicle by the national contractor team. This research includes the development of new computational codes for predicting unsteady aerodynamic pressures. In addition, studies were conducted to determine the aerodynamic heating effects on vehicle aeroelasticity and to determine the effects of fuselage flexibility on the stability of the control systems. It also includes the testing of scale models to better understand the aeroelastic behavior of the X-30 and to obtain data for code validation and correlation. This paper presents an overview of the aeroelastic research which has been conducted to support the airframe design.
Author (revised)
Aerodynamic Heating; Aeroelasticity; Airframes; Fuselages; National Aerospace Plane Program; Single Stage to Orbit Vehicles; Unsteady Aerodynamics; X-30 Vehicle

19930013369 NASA Langley Research Center, Hampton, VA, USA
Subsonic aerodynamic characteristics of a proposed advanced manned launch system orbiter configuration
Ware, George M.; Fox, Charles H., Jr.; Feb 1, 1993; In English
Contract(s)/Grant(s): RTOP 506-40-61-01
Report No.(s): NASA-TM-4439; L-17182; NAS 1.15:4439; No Copyright; Avail: CASI; A03, Hardcopy

The Advanced Manned Launch System is a proposed near-term technology, two-stage, fully reusable launch system that consists of an unmanned glide-back booster and a manned orbiter. An orbiter model that featured a large fuselage and an aft

delta wing with tip fins was tested in the Langley 7- by 10-Foot High-Speed Tunnel. A crew cabin, large payload fairing, and crew access tunnel were mounted on the upper body. The results of the investigation indicated that the configuration was longitudinally stable to an angle of attack of about 6 deg about a center-of-gravity position of 0.7 body length. The model had an untrimmed lift-drag ratio of 6.6, but could not be trimmed at positive lift. The orbiter model was also directionally unstable. The payload fairing was responsible for about half the instability. The tip-fin controllers, which are designed as active controls to produce artificial directional stability, were effective in producing yawing moment, but sizable adverse rolling moment occurred at angles of attack above 6 deg. Differential deflection of the elevon surfaces was effective in producing rolling moment with only small values of adverse yawing moment.
Author
Aerodynamic Characteristics; Directional Stability; Reusable Launch Vehicles; Space Transportation System; Spacecraft Reentry; Wind Tunnel Tests

19930011110 NASA Langley Research Center, Hampton, VA, USA
Computational parametric study of sidewall-compression scramjet inlet performance at Mach 10
Holland, Scott D.; Feb 1, 1993; In English
Contract(s)/Grant(s): RTOP 506-40-71-04
Report No.(s): NASA-TM-4411; L-17134; NAS 1.15:4411; No Copyright; Avail: CASI; A03, Hardcopy

A computational parametric study of three-dimensional, sidewall-compression scramjet inlets was performed to identify the effects of geometric parameters on inlet performance. The parameters were the leading-edge sweep angle, varied between 30 and 60 deg, and the leading-edge position of the cowl, located at the throat and at two forward positions. A laminar boundary layer with cold-wall (T(sub wall) = 300 K (540 R)) boundary conditions was imposed. The parametric study was performed for a Mach number of 10 and a unit free-stream Reynolds number of 7.06 x 10(exp 6) per meter (2.15 x 10(exp 6) per foot) at a geometric contraction ratio of 5. The performance of each configuration was evaluated in terms of the mass capture, throat Mach number, total pressure recovery, kinetic energy efficiency, and internal compression. One computation of an unswept configuration was included as a baseline to determine the effects of introducing leading-edge sweep on the flow-field parameters. The purpose of the computational parametric study was to perform a trade-off of the effects of various parameters on the global performance of the inlet. Although no single optimal configuration emerged, trade-offs among the stated performance parameters identified a leading-edge sweep angle of 45 deg as possessing the most attractive performance characteristics.
Author
Computational Fluid Dynamics; Engine Inlets; Hypersonic Speed; Leading Edge Sweep; Supersonic Combustion Ramjet Engines; Sweep Angle

19930011048 Sverdrup Technology, Inc., Brook Park, OH, USA
Computation of H2/air reacting flowfields in drag-reduction external combustion
Lai, H. T.; Jul 1, 1992; In English; 28th Joint Propulsion Conference and Exhibit, 6-8 Jul. 1992, Nashville, TN, USA
Report No.(s): NASA-CR-191071; E-7624; NAS 1.26:191071; AIAA PAPER 92-3672; No Copyright; Avail: CASI; A03, Hardcopy

Numerical simulation and analysis of the solution are presented for a laminar reacting flowfield of air and hydrogen in the case of external combustion employed to reduce base drag in hypersonic vehicles operating at transonic speeds. The flowfield consists of a transonic air stream at a Mach number of 1.26 and a sonic transverse hydrogen injection along a row of 26 orifices. Self-sustained combustion is computed over an expansion ramp downstream of the injection and a flameholder, using the recently developed RPLUS code. Measured data is available only for surface pressure distributions and is used for validation of the code in practical 3D reacting flowfields. Pressure comparison shows generally good agreements, and the main effects of combustion are also qualitatively consistent with experiment.
Author (revised)
Aerodynamic Drag; Base Flow; Combustion Physics; Drag Reduction; Flow Distribution; Hydrogen Fuels; Hypersonic Vehicles; Supersonic Combustion; Transonic Speed

19930011046 MCAT Inst., San Jose, CA, USA
Turbulence modeling for hypersonic flight
Bardina, Jorge E.; Feb 1, 1993; In English
Contract(s)/Grant(s): NCC2-585
Report No.(s): NASA-CR-192288; NAS 1.26:192288; MCAT-93-07; No Copyright; Avail: CASI; A03, Hardcopy

The objective of the proposed work is to continue to develop, verify, and incorporate the baseline two-equation turbulence models, which account for the effects of compressibility at high speeds, into a three-dimensional Reynolds averaged Navier-Stokes (RANS) code. Additionally, we plan to provide documented descriptions of the models and their numerical procedures so that they can be implemented into the NASP CFD codes.
Derived from text
Aerospace Planes; Compressibility; Hypersonic Flight; National Aerospace Plane Program; Navier-Stokes Equation; Turbulence Models

19930010263 North Carolina Agricultural and Technical State Univ., Greensboro, NC, USA, Futron Corp., Bethesda, MD, USA
The Center for Aerospace Research: A NASA Center of Excellence at North Carolina Agricultural and Technical State University
Lai, Steven H.-Y.; Dec 12, 1992; In English; See also N93-19453 through N93-19466
Contract(s)/Grant(s): NAGW-2924
Report No.(s): NASA-CR-191362; NAS 1.26:191362; No Copyright; Avail: CASI; A11, Hardcopy

This report documents the efforts and outcomes of our research and educational programs at NASA-CORE in NCA&TSU. The goal of the center was to establish a quality aerospace research base and to develop an educational program to increase the participation of minority faculty and students in the areas of aerospace engineering. The major accomplishments of this center in the first year are summarized in terms of three different areas, namely, the center's research programs area, the center's educational programs area, and the center's management area. In the center's research programs area, we focus on developing capabilities needed to support the development of the aerospace plane and high speed civil transportation system technologies. In the educational programs area, we developed an aerospace engineering option program ready for university approval.
Aerospace Engineering; Aerospace Planes; Education; Hypersonic Vehicles; Research and Development; University Program

19930009413 Iowa State Univ. of Science and Technology, Ames, IA, USA
Analytical solutions to constrained hypersonic flight trajectories
Lu, Ping; Nov 30, 1992; In English
Contract(s)/Grant(s): NAG1-1255
Report No.(s): NASA-CR-191987; NAS 1.26:191987; No Copyright; Avail: CASI; A03, Hardcopy

The flight trajectory of aerospace vehicles subject to a class of path constraints is considered. The constrained dynamics is shown to be a natural two-time-scale system. Asymptotic analytical solutions are obtained. Problems of trajectory optimization and guidance can be dramatically simplified with these solutions. Applications in trajectory design for an aerospace plane strongly support the theoretical development.
Author
Aerospace Planes; Aerospace Vehicles; Ascent Trajectories; Cruising Flight; Hypersonic Flight; Trajectory Analysis; Trajectory Optimization

19930009189 Eloret Corp., Sunnyvale, CA, USA
Hypersonic flows as related to the national aerospace plane
Kussoy, Marvin; Menter, F.; Huang, P. G.; Jul 30, 1992; In English
Contract(s)/Grant(s): NCC2-452
Report No.(s): NASA-CR-191980; NAS 1.26:191980; No Copyright; Avail: CASI; A02, Hardcopy

The study in the last 6 months has observed a clear evidence that the current two-equation models tend to under-predict flow separation and over-predict heat transfer rate near flow re-attachment regions. In hypersonic flow calculations, these model deficiencies appear to be even more pronounced. This is particularly true in the incapability of the model to predict the extent of the flow separation. Two major deficiencies of the current two-equation models in predicting complex hypersonic flows have been reported, i.e., under-prediction of flow separation and over-prediction of peak heat transfer rate. Two modifications to the k - epsilon model were reported and tested over a range of flows. Based on our limited study, the modified models have been found to give better agreements in both surface pressure and heat transfer predictions for several complex shock-wave boundary-layer interaction flows. However, in order to confirm our observation, more calculations will be

performed in the future study covering a wider range of flows and conditions than reported here.
Author
Aerospace Planes; Boundary Layer Separation; Hypersonic Flow; National Aerospace Plane Program; Pressure Distribution; Separated Flow

19930007564 NASA Flight Research Center, Edwards, CA, USA
The F-18 systems research aircraft facility
Sitz, Joel R.; Dec 1, 1992; In English; 1992 Aerotech Conference, 5-8 Oct. 1992, Anaheim, CA, USA
Contract(s)/Grant(s): RTOP 505-68-52
Report No.(s): NASA-TM-4433; H-1844; NAS 1.15:4433; No Copyright; Avail: CASI; A03, Hardcopy

To help ensure that new aerospace initiatives rapidly transition to competitive U.S. technologies, NASA Dryden Flight Research Facility has dedicated a systems research aircraft facility. The primary goal is to accelerate the transition of new aerospace technologies to commercial, military, and space vehicles. Key technologies include more-electric aircraft concepts, fly-by-light systems, flush airdata systems, and advanced computer architectures. Future aircraft that will benefit are the high-speed civil transport and the National AeroSpace Plane. This paper describes the systems research aircraft flight research vehicle and outlines near-term programs.
Author
Aerospace Engineering; Aircraft Design; F-18 Aircraft; Research Aircraft; Research Facilities

19930005622 McDonnell Aircraft Co., Saint Louis, MO, USA
Advanced aeroservoelastic stabilization techniques for hypersonic flight vehicles
Chan, Samuel Y.; Cheng, Peter Y.; Myers, Thomas T.; Klyde, David H.; Magdaleno, Raymond E.; Mcruer, Duane T.; Nov 1, 1992; In English
Contract(s)/Grant(s): NAS1-18763; RTOP 505-70-64-01
Report No.(s): NASA-CR-189702; NAS 1.26:189702; No Copyright; Avail: CASI; A06, Hardcopy

Advanced high performance vehicles, including Single-Stage-To-Orbit (SSTO) hypersonic flight vehicles, that are statically unstable, require higher bandwidth flight control systems to compensate for the instability resulting in interactions between the flight control system, the engine/propulsion dynamics, and the low frequency structural modes. Military specifications, such as MIL-F-9490D and MIL-F-87242, tend to limit treatment of structural modes to conventional gain stabilization techniques. The conventional gain stabilization techniques, however, introduce low frequency effective time delays which can be troublesome from a flying qualities standpoint. These time delays can be alleviated by appropriate blending of gain and phase stabilization techniques (referred to as Hybrid Phase Stabilization or HPS) for the low frequency structural modes. The potential of using HPS for compensating structural mode interaction was previously explored. It was shown that effective time delay was significantly reduced with the use of HPS; however, the HPS design was seen to have greater residual response than a conventional gain stablized design. Additional work performed to advance and refine the HPS design procedure, to further develop residual response metrics as a basis for alternative structural stability specifications, and to develop strategies for validating HPS design and specification concepts in manned simulation is presented. Stabilization design sensitivity to structural uncertainties and aircraft-centered requirements are also assessed.
Author
Aeroservoelasticity; Aircraft Control; Design Analysis; Dynamic Models; Dynamic Structural Analysis; Flexible Spacecraft; Flight Control; Hypersonic Vehicles; Stabilization; Structural Stability

19930004117 Iowa State Univ. of Science and Technology, Ames, IA, USA
An inverse dynamics approach to trajectory optimization for an aerospace plane
Lu, Ping; JAN 1, 1992; In English
Contract(s)/Grant(s): NAG1-1255
Report No.(s): NASA-CR-191277; NAS 1.26:191277; No Copyright; Avail: CASI; A03, Hardcopy

An inverse dynamics approach for trajectory optimization is proposed. This technique can be useful in many difficult trajectory optimization and control problems. The application of the approach is exemplified by ascent trajectory optimization for an aerospace plane. Both minimum-fuel and minimax types of performance indices are considered. When rocket augmentation is available for ascent, it is shown that accurate orbital insertion can be achieved through the inverse control of the rocket in the presence of disturbances.
Author
Aerospace Planes; Minimax Technique; Spacecraft Control; Spacecraft Trajectories; Trajectory Control; Trajectory Optimization

19930004100 NASA Hugh L. Dryden Flight Research Facility, Edwards, CA, USA

Stratospheric turbulence measurements and models for aerospace plane design

Ehernberger, L. J.; Dec 1, 1992; In English; 4th AIAA International Aerospace Planes Conference, 1-4 Dec. 1992, Orlando, FL, USA

Contract(s)/Grant(s): RTOP 763-21-51

Report No.(s): NASA-TM-104262; H-1865; NAS 1.15:104262; Copyright; Avail: CASI; A03, Hardcopy

Progress in computational atmospheric dynamics is exhibiting the ability of numerical simulation to describe instability processes associated with turbulence observed at altitudes between 15 and 25 km in the lower stratosphere. As these numerical simulation tools mature, they can be used to extend estimates of atmospheric perturbations from the present gust database for airplane design at altitudes below 15 km to altitudes between 25 and 50 km where aerospace plane operation would be at hypersonic speeds. The amount of available gust data and number of temperature perturbation observations are limited at altitudes between 15 and 25 km. On the other hand, in-situ gust data at higher altitudes are virtually nonexistent. The uncertain potential for future airbreathing hypersonic flight research vehicles to encounter strong turbulence at higher altitudes could penalize the design of these vehicles by undue cost or limitations on performance. Because the atmospheric structure changes markedly with altitude, direct extrapolation of gust magnitudes and encounter probabilities to the higher flight altitudes is not advisable. This paper presents a brief review of turbulence characteristics observed in the lower stratosphere and highlights the progress of computational atmospheric dynamics that may be used to estimate the severity of atmospheric transients at higher altitudes.

Author

Aerospace Planes; Atmospheric Models; Atmospheric Turbulence; Flight Conditions; Gusts; Stratosphere; Wind Velocity Measurement

19930003922 NASA, Washington, DC, USA

NASA aeronautics: Research and technology program highlights

JAN 1, 1990; In English; 44 functional color pages

Report No.(s): NASA-NP-159; NAS 1.83:159; No Copyright; Avail: CASI; A05, Hardcopy; 44 functional color pages

This report contains numerous color illustrations to describe the NASA programs in aeronautics. The basic ideas involved are explained in brief paragraphs. The seven chapters deal with Subsonic aircraft, High-speed transport, High-performance military aircraft, Hypersonic/Transatmospheric vehicles, Critical disciplines, National facilities and Organizations & installations. Some individual aircraft discussed are : the SR-71 aircraft, aerospace planes, the high-speed civil transport (HSCT), the X-29 forward-swept wing research aircraft, and the X-31 aircraft. Critical disciplines discussed are numerical aerodynamic simulation, computational fluid dynamics, computational structural dynamics and new experimental testing techniques.

R.L.B.

Aeronautics; Aerospace Planes; Computational Fluid Dynamics; Computerized Simulation; Dynamic Structural Analysis; NASA Programs; Research Aircraft; X-29 Aircraft

19930003880 NASA Langley Research Center, Hampton, VA, USA

Hypersonic vehicle control law development using H infinity and mu-synthesis

Gregory, Irene M.; Chowdhry, Rajiv S.; Mcminn, John D.; Shaughnessy, John D.; Oct 1, 1992; In English; 4th AIAA International Aerospace Planes Conference, 1-4 Dec. 1992, Orlando, FL, USA

Contract(s)/Grant(s): RTOP 505-64-40-01

Report No.(s): NASA-TM-107689; NAS 1.15:107689; No Copyright; Avail: CASI; A03, Hardcopy

Applicability and effectiveness of robust control techniques to a single-stage-to-orbit (SSTO) airbreathing hypersonic vehicle on an ascent accelerating path and their effectiveness are explored in this paper. An SSTO control system design problem, requiring high accuracy tracking of velocity and altitude commands while limiting angle of attack oscillations, minimizing control power usage and stabilizing the vehicle all in the presence of atmospheric turbulence and uncertainty in the system, was formulated to compare results of the control designs using H infinity and mu-synthesis procedures. The math model, an integrated flight/propulsion dynamic model of a conical accelerator class vehicle, was linearized as the vehicle accelerated through Mach 8. Controller analysis was conducted using the singular value technique and the mu-analysis approach. Analysis results were obtained in both the frequency and the time domains. The results clearly demonstrate the inherent advantages of the structured singular value framework for this class of problems. Since payload performance margins

are so critical for the SSTO mission, it is crucial that adequate stability margins be provided without sacrificing any payload mass.
Author
Control Systems Design; Control Theory; H-Infinity Control; Hypersonic Flight; Hypersonic Vehicles; Single Stage to Orbit Vehicles; Spacecraft Control

19930003259 NASA Langley Research Center, Hampton, VA, USA
Current Technology for Thermal Protection Systems
Scotti, Stephen J., compiler; Oct 1, 1992; In English, 11-12 Feb. 1992, Hampton, VA, USA; See also N93-12448 through N93-12460
Contract(s)/Grant(s): RTOP 506-43-31-04
Report No.(s): NASA-CP-3157; L-17118; NAS 1.55:3157; No Copyright; Avail: CASI; A14, Hardcopy

Interest in thermal protection systems for high-speed vehicles is increasing because of the stringent requirements of such new projects as the Space Exploration Initiative, the National Aero-Space Plane, and the High-Speed Civil Transport, as well as the needs for improved capabilities in existing thermal protection systems in the Space Shuttle and in turbojet engines. This selection of 13 papers from NASA and industry summarizes the history and operational experience of thermal protection systems utilized in the national space program to date, and also covers recent development efforts in thermal insulation, refractory materials and coatings, actively cooled structures, and two-phase thermal control systems.
Aerospace Planes; Composite Materials; Composite Structures; Refractory Coatings; Refractory Materials; Space Shuttles; Supersonic Transports; Temperature Control; Thermal Insulation; Thermal Protection; Two Phase Flow

19930003225 Draper (Charles Stark) Lab., Inc., Cambridge, MA, USA
Air-breathing hypersonic vehicle guidance and control studies: An integrated trajectory/control analysis methodology, phase 2
Hattis, Philip D.; Malchow, Harvey L.; Oct 1, 1992; In English
Contract(s)/Grant(s): NAS1-18565; RTOP 505-70-64-01
Report No.(s): NASA-CR-189703; NAS 1.26:189703; No Copyright; Avail: CASI; A05, Hardcopy

An integrated trajectory/control analysis algorithm has been used to generate trajectories and desired control strategies for two different hypersonic air-breathing vehicle models and orbit targets. Both models used cubic spline curve fit tabulated winged-cone accelerator vehicle representations. Near-fuel-optimal, horizontal takeoff trajectories, imposing a dynamic pressure limit of 1000 psf, were developed. The first model analysis case involved a polar orbit and included the dynamic effects of using elevons to maintain longitudinal trim. Analysis results indicated problems with the adequacy of the propulsion model and highlighted dynamic pressure/altitude instabilities when using vehicle angle of attack as a control variable. Also, the magnitude of computed elevon deflections to maintain trim suggested a need for alternative pitch moment management strategies. The second analysis case was reformulated to use vehicle pitch attitude relative to the local vertical as the control variable. A new, more realistic, air-breathing propulsion model was incorporated. Pitch trim calculations were dropped and an equatorial orbit was specified. Changes in flight characteristics due to the new propulsion model have been identified. Flight regimes demanding rapid attitude changes have been noted. Also, some issues that would affect design of closed-loop controllers were ascertained.
Author
Air Breathing Engines; Control Systems Design; Feedback Control; Hypersonic Vehicles; Single Stage to Orbit Vehicles; Spacecraft Control; Spacecraft Design; Spacecraft Guidance; Spacecraft Trajectories; Trajectory Analysis; Trajectory Control

19930001912 NASA Langley Research Center, Hampton, VA, USA
Winds of change: Expanding the frontiers of flight. Langley Research Center's 75 years of accomplishment, 1917-1992
Schultz, James; JAN 1, 1992; In English
Report No.(s): NASA-NP-130; NAS 1.83:130; No Copyright; Avail: CASI; A07, Hardcopy; Limited Reproducibility: More than 20% of this doc

This commemorative volume highlights in pictures and text seventy five years of accomplishments of the Langley Research Center. The introductory matter features wind tunnels and their contribution to the development of aeronautics. A chronological survey details four different periods in Langley's history. The first period, 1917-1939, is subtitled 'Perfecting the Plane' which details Langley's contribution to early aeronautics with examples from specific aircraft. The second period,

1940-1957, focuses on the development of military aircraft during and after World War II. The third period, 1958-1969, tells the story of Langley's involvement with NASA and the satellite and Apollo era. The fourth period, entitled 'Charting New Courses: 1970-1992 and Beyond', treats various new topics from aerospace planes to Mars landing, as well as older topics such as the Space Shuttle and research spinoffs.
R.L.B.
Aeronautics; Aerospace Sciences; Aircraft; Histories; Military Aircraft; NASA Programs; Wind Tunnels

19930001855 Alabama Univ., Huntsville, AL, USA
Rocket Based Combined Cycle (RBCC) Propulsion Workshop, volume 2
Chojnacki, Kent T.; Aug 28, 1992; In English
Contract(s)/Grant(s): NAS8-38609
Report No.(s): NASA-CR-184383; NAS 1.26:184383; UAH-TR-91/001-VOL-2; No Copyright; Avail: CASI; A04, Hardcopy

The goal of the Rocket Based Combined Cycle (RBCC) Propulsion Technology Workshop, was to impart technology information to the propulsion community with respect to hypersonic combined cycle propulsion capabilities. The major recommendation resulting from this technology workshop was as follows: conduct a systems-level applications study to define the desired propulsion system and vehicle technology requirements for LEO launch vehicles. All SSTO and TSTO options using the various propulsion systems (airbreathing combined cycle, rocket-based combined cycle, and all rocket) must be considered. Such a study should be accomplished as soon as possible. It must be conducted with a consistent set of ground rules and assumptions. Additionally, the study should be conducted before any major expenditures on a RBCC technology development program occur.
Author
Air Breathing Engines; Conferences; Hypersonic Flight; Launch Vehicles; Propulsion System Configurations; Propulsion System Performance; Single Stage to Orbit Vehicles

19930001854 Alabama Univ., Huntsville, AL, USA
Rocket Based Combined Cycle (RBCC) Propulsion Technology Workshop. Volume 1: Executive summary
Chojnacki, Kent T.; Aug 28, 1992; In English
Contract(s)/Grant(s): NAS8-38609
Report No.(s): NASA-CR-184382; NAS 1.26:184382; UAH-TR-92/001-VOL-1; No Copyright; Avail: CASI; A03, Hardcopy

The goal of the Rocket-Based Combined Cycle (RBCC) Propulsion Technology Workshop was to assess the RBCC propulsion system's viability for Earth-to-Orbit (ETO) transportation systems. This was accomplished by creating a forum (workshop) in which past work in the field of RBCC propulsion systems was reviewed, current technology status was evaluated, and future technology programs in the field of RBCC propulsion systems were postulated, discussed, and recommended.
Author
Aerospace Planes; Air Breathing Engines; Propulsion System Configurations; Propulsion System Performance; Rocket Engine Design; Single Stage to Orbit Vehicles

19920018966 Draper (Charles Stark) Lab., Inc., Cambridge, MA, USA
Robust intelligent flight control for hypersonic vehicles
Chamitoff, Gregory Errol; Feb 20, 1992; In English
Contract(s)/Grant(s): NAS9-18426
Report No.(s): NASA-CR-185680; NAS 1.26:185680; CSDL-T-1106; Copyright; Avail: CASI; A15, Hardcopy

Intelligent optimization methods are applied to the problem of real-time flight control for a class of airbreathing hypersonic vehicles (AHSV). The extreme flight conditions that will be encountered by single-stage-to-orbit vehicles, such as the National Aerospace Plane, present a tremendous challenge to the entire spectrum of aerospace technologies. Flight control for these vehicles is particularly difficult due to the combination of nonlinear dynamics, complex constraints, and parametric uncertainty. An approach that utilizes all available a priori and in-flight information to perform robust, real time, short-term trajectory planning is presented.
CASI
Aerospace Planes; Air Breathing Engines; Control Systems Design; Flight Control; Hypersonic Vehicles; Real Time Operation; Single Stage to Orbit Vehicles; Spacecraft Control; Trajectory Planning

19920016739 Systems Technology, Inc., Hawthorne, CA, USA, PRC Kentron, Inc., Edwards, CA, USA
Assessment of flying-quality criteria for air-breathing aerospacecraft
Mcruer, Duane T.; Myers, Thomas T.; Hoh, Roger H.; Ashkenas, Irving L.; Johnston, Donald E.; Jun 1, 1992; In English
Contract(s)/Grant(s): NAS2-12722; ATD-88-STI-0221; RTOP 505-64-40
Report No.(s): NASA-CR-4442; H-1758; NAS 1.26:4442; STI-TR-2361-2; No Copyright; Avail: CASI; A05, Hardcopy

A study of flying quality requirements for air breathing aerospacecraft gives special emphasis to the unusual operational requirements and characteristics of these aircraft, including operation at hypersonic speed. The report considers distinguishing characteristics of these vehicles, including dynamic deficiencies and their implications for control. Particular emphasis is given to the interaction of the airframe and propulsion system, and the requirements for dynamic systems integration. Past operational missions are reviewed to define tasks and maneuvers to be considered for this class of aircraft. Areas of special concern with respect to vehicle dynamics and control are identified. Experience with the space shuttle orbiter is reviewed with respect to flight control system mechanization and flight experience in approach and landing flying qualities for the National Aerospace Plane (NASP).
CASI
Aerospace Planes; Air Breathing Engines; Flight Characteristics; Flight Control; Hypersonic Speed; National Aerospace Plane Program; Systems Integration

19920016714 NASA Lewis Research Center, Cleveland, OH, USA
Slush hydrogen transfer studies at the NASA K-Site Test Facility
Hardy, Terry L.; Whalen, Margaret V.; JAN 1, 1992; In English; 28th Joint Propulsion Conference and Exhibit, 6-8 Jul. 1992, Nashville, TN, USA
Contract(s)/Grant(s): RTOP 763-22-21
Report No.(s): NASA-TM-105596; E-6932; NAS 1.15:105596; AIAA PAPER 92-3384; No Copyright; Avail: CASI; A02, Hardcopy

An experimental study was performed as part of the National Aerospace Plane (NASP) effort to determine slush hydrogen production and transfer characteristics. Flow rate and pressure drop characteristics were determined for slush hydrogen flow through a vacuum-jacketed transfer system. These characteristics were compared to similar tests using normal boiling point and triple point hydrogen. In addition, experimental flow characteristic data was compared with predictions from the FLUSH analytical model. Slush hydrogen density loss during the transfer process was also examined.
CASI
Aerospace Planes; Cryogenic Fluid Storage; Fuel Production; Hydrogen Fuels; Liquid Hydrogen; National Aerospace Plane Program; Slush Hydrogen; Transferring

19920016034 NASA Lewis Research Center, Cleveland, OH, USA
Technology issues associated with using densified hydrogen for space vehicles
Hardy, Terry L.; Whalen, Margaret V.; JAN 1, 1992; In English; 28th Joint Propulsion Conference and Exhibit, 6-8 Jul. 1992, Nashville, TN, USA
Contract(s)/Grant(s): RTOP 763-22-21
Report No.(s): NASA-TM-105642; E-6988; NAS 1.15:105642; AIAA PAPER 92-3079; No Copyright; Avail: CASI; A03, Hardcopy

Slush hydrogen and triple-point hydrogen offer the potential for reducing the size and weight of future space vehicles because these fluids have greater densities than normal-boiling-point liquid hydrogen. In addition, these fluids have greater heat capacities, which make them attractive fuels for such applications as the National Aerospace Plane and cryogenic depots. Some of the benefits of using slush hydrogen and triple-point hydrogen for space missions are quantified. Some of the major issues associated with using these densified cryogenic fuels for space applications are examined, and the technology efforts that have been made to address many of these issues are summarized.
CASI
Aerospace Planes; Cryogenics; Hydrogen; Hydrogen Fuels; Liquid Hydrogen; Slush Hydrogen; Spacecraft Propulsion

19920015921 NASA Lewis Research Center, Cleveland, OH, USA
The future challenge for aeropropulsion
Rosen, Robert; Bowditch, David N.; JAN 1, 1992; In English; Aeroengine 1992, 6-12 Apr. 1992, Moscow, USSR
Contract(s)/Grant(s): RTOP 505-62-00
Report No.(s): NASA-TM-105613; E-6943; NAS 1.15:105613; No Copyright; Avail: CASI; A03, Hardcopy

NASA's research in aeropropulsion is focused on improving the efficiency, capability, and environmental compatibility for all classes of future aircraft. The development of innovative concepts, and theoretical, experimental, and computational tools provide the knowledge base for continued propulsion system advances. Key enabling technologies include advances in internal fluid mechanics, structures, light-weight high-strength composite materials, and advanced sensors and controls. Recent emphasis has been on the development of advanced computational tools in internal fluid mechanics, structural mechanics, reacting flows, and computational chemistry. For subsonic transport applications, very high bypass ratio turbofans with increased engine pressure ratio are being investigated to increase fuel efficiency and reduce airport noise levels. In a joint supersonic cruise propulsion program with industry, the critical environmental concerns of emissions and community noise are being addressed. NASA is also providing key technologies for the National Aerospaceplane, and is studying propulsion systems that provide the capability for aircraft to accelerate to and cruise in the Mach 4-6 speed range. The combination of fundamental, component, and focused technology development underway at NASA will make possible dramatic advances in aeropropulsion efficiency and environmental compatibility for future aeronautical vehicles.
CASI
Aerospace Planes; Aircraft Engines; Aircraft Noise; Fluid Mechanics; NASA Programs; Propulsion; Propulsion System Configurations; Propulsion System Performance; Structural Analysis; Turbofans

19920012304 NASA Lewis Research Center, Cleveland, OH, USA
Mission and sizing analysis for the Beta 2 two-stage-to-orbit vehicle
Nadell, Shari-Beth; Baumgarten, William J.; Alexander, Stephen W.; JAN 1, 1992; In English; 1992 Aerospace Design Conference, 3-6 Feb. 1992, Irvine, CA, USA
Contract(s)/Grant(s): RTOP 505-69-40
Report No.(s): NASA-TM-105559; E-6883; NAS 1.15:105559; AIAA PAPER 92-1264; No Copyright; Avail: CASI; A03, Hardcopy

NASA Lewis Research Center studied a horizontal takeoff and landing, fully reusable, two-stage-to-orbit (TSTO) vehicle capable of launching and returning a 10,000 pound payload to low Earth polar orbit using low-risk technology. The vehicle, called Beta 2, was derived from the USAF/Boeing Beta vehicle, a TSTO study vehicle capable of launching a 50,000 pound payload to low Earth polar orbit. Development of the Beta 2 from the USAF/Boeing Beta vehicle occurred in a series of iterations during which the size of the vehicle was decreased to accommodate the smaller payload, the staging Mach number was decreased from 8.0 to 6.5, and the rocket propulsion system was removed from the booster. The final Beta 2 vehicle consisted of a rocket powered orbiter and an all airbreathing booster. The gross takeoff weight of the Beta 2 vehicle was approximately 1.1 million pounds. In addition to its baseline mission, the Beta 2 was capable of delivering approximately 17,500 pounds to the Space Station with the same takeoff gross weight. The mission and sizing analysis performed to arrived at the Beta 2 vehicle is discussed.
CASI
Aerospace Vehicles; Air Breathing Boosters; Aircraft Design; Launch Vehicle Configurations; Reusable Launch Vehicles; Reusable Spacecraft; Spacecraft Design; Spacecraft Launching

19920012274 NASA Lewis Research Center, Cleveland, OH, USA
Rocket-Based Combined-Cycle (RBCC) Propulsion Technology Workshop. Tutorial session
JAN 1, 1992; In English, 23-27 Mar. 1992, Huntsville, AL, USA; See also N92-21518 through N92-21538
Contract(s)/Grant(s): RTOP 590-21-11
Report No.(s): NASA-CP-10090; E-6929; NAS 1.55:10090; No Copyright; Avail: CASI; A12, Hardcopy

The goal of this workshop was to illuminate the nation's space transportation and propulsion engineering community on the potential of hypersonic combined cycle (airbreathing/rocket) propulsion systems for future space transportation applications. Four general topics were examined: (1) selections from the expansive advanced propulsion archival resource; (2) related propulsion systems technical backgrounds; (3) RBCC engine multimode operations related subsystem background; and (4) focused review of propulsion aspects of current related programs.
Aerospace Planes; Engine Parts; Hypersonic Flight; Rocket Engine Design; Rocket Engines

19920011425 Washington Univ., Seattle, WA, USA
Project Antares: A low cost modular launch vehicle for the future
Aarnio, Steve; Anderson, Hobie; Arzaz, El Mehdi; Bailey, Michelle; Beeghly, Jeff; Cartwright, Curt; Chau, William; Dawdy, Andrew; Detert, Bruce; Ervin, Miles, et al.; Jun 14, 1991; In English
Contract(s)/Grant(s): NASW-4435
Report No.(s): NASA-CR-190018; NAS 1.26:190018; No Copyright; Avail: CASI; A10, Hardcopy

The single stage to orbit launch vehicle Antares is based upon the revolutionary concept of modularity, enabling the Antares to efficiently launch communications satellites, as well as heavy payloads, into Earth's orbit and beyond. The basic unit of the modular system, a single Antares vehicle, is aimed at launching approximately 10,000 kg into low Earth orbit (LEO). When coupled with a Centaur upper stage it is capable of placing 3500 kg into geostationary orbit. The Antares incorporates a reusable engine, the Dual Mixture Ratio Engine (DMRE), as its propulsive device. This enables Antares to compete and excel in the satellite launch market by dramatically reducing launch costs. Antares' projected launch costs are $1340 per kg to LEO which offers a tremendous savings over launch vehicles available today. Inherent in the design is the capability to attach several of these vehicles together to provide heavy lift capability. Any number of these vehicles, up to seven, can be attached depending on the payload and mission requirements. With a seven vehicle configuration Antares's modular concept provides a heavy lift capability of approximately 70,000 kg to LEO. This expandability allows for a wider range of payload options such as large Earth satellites, Space Station Freedom support, and interplanetary spacecraft, and also offers a significant cost savings over a mixed fleet based on different launch vehicles.
CASI

Antares Rocket Vehicle; Heavy Lift Launch Vehicles; Single Stage to Orbit Vehicles; Spacecraft Design

19920011424 Case Western Reserve Univ., Cleveland, OH, USA

Conceptual design of two-stage-to-orbit hybrid launch vehicle

Jul 1, 1991; In English

Contract(s)/Grant(s): NASW-4435

Report No.(s): NASA-CR-190006; NAS 1.26:190006; No Copyright; Avail: CASI; A12, Hardcopy

The object of this design class was to design an earth-to orbit vehicle to replace the present NASA space shuttle. The major motivations for designing a new vehicle were to reduce the cost of putting payloads into orbit and to design a vehicle that could better service the space station with a faster turn-around time. Another factor considered in the design was that near-term technology was to be used. Materials, engines and other important technologies were to be realized in the next 10 to 15 years. The first concept put forth by NASA to meet these objectives was the National Aerospace Plane (NASP). The NASP is a single-stage earth-to-orbit air-breathing vehicle. This concept ran into problems with the air-breathing engine providing enough thrust in the upper atmosphere, among other things. The solution of this design class is a two-stage-to-orbit vehicle. The first stage is air-breathing and the second stage is rocket-powered, similar to the space shuttle. The second stage is mounted on the top of the first stage in a piggy-back style. The vehicle takes off horizontally using only air-breathing engines, flies to Mach six at 100,000 feet, and launches the second stage towards its orbital path. The first stage, or booster, will weigh approximately 800,000 pounds and the second stage, or orbiter will weigh approximately 300,000 pounds. The major advantage of this design is the full recoverability of the first stage compared with the present solid rocket booster that are only partially recoverable and used only a few times. This reduces the cost as well as providing a more reliable and more readily available design for servicing the space station. The booster can fly an orbiter up, turn around, land, refuel, and be ready to launch another orbiter in a matter of hours.
CASI

Aerospace Planes; Cost Reduction; Design Analysis; Hypersonic Vehicles; Launch Vehicles; Orbital Launching; Recoverability; Reusable Spacecraft; Transatmospheric Vehicles

19920010472 State Univ. of New York, Buffalo, NY, USA

Development of an integrated BEM approach for hot fluid structure interaction

Dargush, Gary F.; Banerjee, Prasanta K.; Honkala, Keith A.; Nov 1, 1991; In English

Contract(s)/Grant(s): NAG3-712; RTOP 553-13-00

Report No.(s): NASA-CR-189052; NAS 1.26:189052; No Copyright; Avail: CASI; A06, Hardcopy

The development of a boundary element formulation for the study of hot fluid-structure interaction in earth-to-orbit engine hot section components is described. The initial primary thrust of the program to date was directed quite naturally toward the examination of fluid flow, since boundary element methods for fluids are at a much less developed state. This required the development of integral formulations for both the solid and fluid, and some preliminary infrastructural enhancements to a boundary element code to permit coupling of the fluid-structure problem. Boundary element formulations are implemented in two dimensions for both the solid and the fluid. The solid is modeled as an uncoupled thermoelastic medium under plane strain conditions, while several formulations are investigated for the fluid. For example, both vorticity and primitive variable approaches are implemented for viscous, incompressible flow, and a compressible version is developed. All of the above boundary element implementations are incorporated in a general purpose two-dimensional code. Thus, problems involving

intricate geometry, multiple generic modeling regions, and arbitrary boundary conditions are all supported.
CASI
Boundary Element Method; Compressible Flow; Elastic Media; Fluid-Solid Interactions; Incompressible Flow; Single Stage to Orbit Vehicles; Spacecraft Propulsion; Thermal Stresses; Thermoelasticity; Viscous Flow

19920010243 Rockwell International Corp., Canoga Park, CA, USA
Operationally Efficient Propulsion System Study (OEPSS) data book. Volume 2: Ground operations problems
Waldrop, Glen S.; Apr 24, 1990; In English
Contract(s)/Grant(s): NAS10-11568
Report No.(s): NASA-CR-188750; NAS 1.26:188750; RI/RD90-149-2-VOL-2; No Copyright; Avail: CASI; A05, Hardcopy

Operations problems and cost drivers were identified for current propulsion systems and design and technology approaches were identified to increase the operational efficiency and to reduce operations costs for future propulsion systems. To provide readily usable data for the ALS program, the results of the OEPSS study were organized into a series of OEPSS Data Books. This volume presents a detailed description of 25 major problems encountered during launch processing of current expendable and reusable launch vehicles. A concise description of each problem and its operational impact on launch processing is presented, along with potential solutions and technology recommendation.
CASI
Propulsion System Configurations; Propulsion System Performance; Reusable Launch Vehicles; Spacecraft Launching; Spacecraft Propulsion

19920007788 Lockheed Aeronautical Systems Co., Burbank, CA, USA
Heat flux sensor research and development: The cool film calorimeter
Abtahi, A.; Dean, P.; Jun 30, 1990; In English
Contract(s)/Grant(s): NAS1-18570
Report No.(s): NASA-CR-189789; NAS 1.26:189789; LR-31824; No Copyright; Avail: CASI; A04, Hardcopy

The goal was to meet the measurement requirement of the NASP program for a gauge capable of measuring heat flux into a ‘typical’ structure in a ‘typical’ hypersonic flight environment. A device is conceptually described that has fast response times and is small enough to fit in leading edge or cowl lip structures. The device relies heavily on thin film technology. The main conclusion is the description of the limitations of thin film technology both in the art of fabrication and in the assumption that thin films have the same material properties as the original bulk material. Three gauges were designed and fabricated. Thin film deposition processes were evaluated. The effect of different thin film materials on the performance and fabrication of the gauge was studied. The gauges were tested in an arcjet facility. Survivability and accuracy were determined under various hostile environment conditions.
CASI
Calorimeters; Heat Flux; Leading Edges; Thin Films

19920005858 NASA Langley Research Center, Hampton, VA, USA
Trim drag reduction concepts for horizontal takeoff single-stage-to-Orbit vehicles
Shaughnessy, John D.; Gregory, Irene M.; Dec 1, 1991; In English
Contract(s)/Grant(s): RTOP 505-64-40-01
Report No.(s): NASA-TM-102687; NAS 1.15:102687; No Copyright; Avail: CASI; A03, Hardcopy

The results of a study to investigate concepts for minimizing trim drag of horizontal takeoff single-stage-to-orbit (SSTO) vehicles are presented. A generic hypersonic airbreathing conical configuration was used as the subject aircraft. The investigation indicates that extreme forward migration of the aerodynamic center as the vehicle accelerates to orbital velocities causes severe aerodynamic instability and trim moments that must be counteracted. Adequate stability can be provided by active control of elevons and rudder, but use of elevons to produce trim moments results in excessive trim drag and fuel consumption. To alleviate this problem, two solution concepts are examined. Active control of the center of gravity (COG) location to track the aerodynamic center decreases trim moment requirements, reduces elevon deflections, and leads to significant fuel savings. Active control of the direction of the thrust vector produces required trim moments, reduces elevon deflections, and also results in significant fuel savings. It is concluded that the combination of active flight control to provide stabilization, (COG) position control to minimize trim moment requirements, and thrust vectoring to generate required trim

moments has the potential to significantly reduce fuel consumption during ascent to orbit of horizontal takeoff SSTO vehicles.
CASI

Aerodynamic Drag; Aerodynamic Stability; Air Breathing Engines; Drag Reduction; Flight Control; Fuel Consumption; Horizontal Flight; Hypersonic Vehicles; Single Stage to Orbit Vehicles

19920005095 Nevada Univ., Reno, NV, USA

Numerical investigations in three-dimensional internal flows

Rose, William C.; JAN 1, 1991; In English

Contract(s)/Grant(s): NCC2-507

Report No.(s): NASA-CR-189467; NAS 1.26:189467; No Copyright; Avail: CASI; A03, Hardcopy

The general objectives of NASA's Generic Hypersonics Research Program (GHP) are to develop a technology background required for aeronautical research in the hypersonic Mach number flow range. These research efforts are to complement the National Aerospace Plane (NASP) Program and are geared toward the development of experimental and computational fluid dynamics (CFD) techniques. One of the prominent goals of inlet design for high speed applications is to produce an inlet that delivers uniform flow at its exit in the shortest possible distance. In previous studies, the technologies for determining contours for both the ramp and cowl were demonstrated that allowed a nearly shock free exiting flow field to be obtained. This technology was developed further during the present reporting period and applied to a preliminary design investigation of a biconic hypersonic research vehicle with a nearly 2-D inlet attached near the aft end of the vehicle. The results of a parametric investigation of this proposed inlet for freestream Mach numbers between 10 and 15 are described.
CASI

Aircraft Design; Engine Inlets; Hypersonic Flow; Hypersonic Vehicles; Hypersonics; Inlet Flow; Mach Number; Research Vehicles; Three Dimensional Flow

19920004848 Georgia Inst. of Tech., Atlanta, GA, USA

Rapid near-optimal aerospace plane trajectory generation and guidance

Calise, A. J.; Corban, J. E.; Markopoulos, N.; Nov 1, 1991; In English

Contract(s)/Grant(s): NAG1-922

Report No.(s): NASA-CR-189469; NAS 1.26:189469; No Copyright; Avail: CASI; A03, Hardcopy

Effort was directed toward the problems of the real time trajectory optimization and guidance law development for the National Aerospace Plane (NASP) applications. In particular, singular perturbation methods were used to develop guidance algorithms suitable for onboard, real time implementation. The progress made in this research effort is reported.
CASI

Aerospace Planes; Aircraft Guidance; Algorithms; National Aerospace Plane Program; Real Time Operation; Trajectory Control; Trajectory Optimization

19920002784 Draper (Charles Stark) Lab., Inc., Cambridge, MA, USA

Air-breathing hypersonic vehicle guidance and control studies; An integrated trajectory/control analysis methodology: Phase 1

Hattis, Philip D.; Malchow, Harvey L.; Sep 1, 1991; In English

Contract(s)/Grant(s): NAS1-18565; RTOP 505-64-40-01

Report No.(s): NASA-CR-187623; NAS 1.26:187623; No Copyright; Avail: CASI; A05, Hardcopy

A tool which generates optimal trajectory/control histories in an integrated manner is generically adapted to the treatment of single-stage-to-orbit air-breathing hypersonic vehicles. The methodology is implemented as a two point boundary value problem solution technique. Its use permits an assessment of an entire near-minimum-fuel trajectory and desired control strategy from takeoff to orbit while satisfying physically derived inequality constraints and while achieving efficient propulsive mode phasing. A simpler analysis strategy that partitions the trajectory into several boundary condition matched segments is also included to construct preliminary trajectory and control history representations with less computational burden than is required for the overall flight profile assessment. A demonstration was accomplished using a tabulated example (winged-cone accelerator) vehicle model that is combined with a newly developed multidimensional cubic spline data smoothing routine. A constrained near-fuel-optimal trajectory, imposing a dynamic pressure limit of 1000 psf, was developed from horizontal takeoff to 20,000 ft/sec relative air speed while aiming for a polar orbit. Previously unspecified propulsive discontinuities were located. Flight regimes demanding rapid attitude changes were identified, dictating control effector and closed-loop controller authority was ascertained after evaluating effector use for vehicle trim. Also, inadequacies in vehicle model representations

and specific subsystem models with insufficient fidelity were determined based on unusual control characteristics and/or excessive sensitivity to uncertainty.
CASI
Boundary Value Problems; Controllers; Data Smoothing; Feedback Control; Flight Paths; Hypersonic Vehicles; Single Stage to Orbit Vehicles; Trajectories; Trajectory Control

19920002173 Washington Univ., Seattle, WA, USA
Thermoviscoplastic response of Ti-15-3 under various loading conditions
Tuttle, M. E.; Rogacki, J.; Oct 1, 1991; In English
Contract(s)/Grant(s): NAG1-974; RTOP 505-63-50-04
Report No.(s): NASA-CR-187621; NAS 1.26:187621; No Copyright; Avail: CASI; A08, Hardcopy

Metal matrix composites (MMC's) are candidate materials for use in high temperature, high loading applications. In particular, an MMC consisting of a titanium alloy reinforced with silicon-carbide fibers is being considered for use on the National Aerospace Plane (NASP). Compared to other metals and metallic alloys, titanium alloys retain relatively high stiffness, strength, and corrosion resistance at elevated temperatures. However, above roughly 316 C titanium exhibits a significant thermoviscoplastic (creep) response. Since the temperatures encountered in many regions of the NASP are expected to exceed 316 C, the potential thermoviscoplastic behavior of titanium-based MMC's at elevated temperatures must be thoroughly investigated.
CASI
Creep Properties; Cyclic Loads; Heat Resistant Alloys; Metal Matrix Composites; Spacecraft Construction Materials; Thermoplasticity; Titanium Alloys; Viscoplasticity

19920001816 NASA Lewis Research Center, Cleveland, OH, USA
Overview of the Beta II Two-Stage-To-Orbit vehicle design
Plencner, Robert M.; Oct 1, 1991; In English; Aircraft Design Systems and Operations Meeting, 23-25 Sep. 1991, Baltimore, MD, USA
Contract(s)/Grant(s): RTOP 505-69-40
Report No.(s): NASA-TM-105298; E-6641; NAS 1.15:105298; AIAA PAPER 91-3175; Copyright; Avail: CASI; A03, Hardcopy

A study of a near-term, low risk two-stage-to-orbit (TSTO) vehicle was undertaken. The goal of the study was to assess a fully reusable TSTO vehicle with horizontal takeoff and landing capability that could deliver 10,000 pounds to a 120 nm polar orbit. The configuration analysis was based on the Beta vehicle design. A cooperative study was performed to redesign and refine the Beta concept to meet the mission requirements. The vehicle resulting from this study was named Beta II. It has an all-airbreathing first stage and a staging Mach number of 6.5. The second stage is a conventional wing-body configuration with a single SSME.
CASI
Aerospace Planes; Air Breathing Engines; Launch Vehicle Configurations; Multiengine Vehicles; Multistage Rocket Vehicles; Polar Orbits; Space Shuttle Main Engine

19920000786 Eloret Corp., Sunnyvale, CA, USA
Hypersonic flows as related to the National Aerospace plane
Kussoy, Marvin; Levy, Lionel; Menter, F.; Mar 19, 1991; In English
Contract(s)/Grant(s): NCC2-452
Report No.(s): NASA-CR-187985; NAS 1.26:187985; No Copyright; Avail: CASI; A04, Hardcopy

Experimental data for a series of 2-D and 3-D shock wave/boundary layer interaction flows at Mach 8.2 are presented. The test bodies, composed of simple geometric shapes fastened to a flat plate test bed, were designed to generate flows with varying degrees of pressure gradient, boundary layer separation, and turning angle. The data include surface pressure and heat transfer distributions as well as limited mean flowfield surveys both in the undisturbed and interaction regimes. The data are presented in a convenient form to be used to validate existing or future computational models of these hypersonic flows.
CASI
Aerospace Planes; Boundary Layer Flow; Boundary Layer Separation; Flow Distribution; Heat Transfer; Hypersonic Flow; National Aerospace Plane Program; Shock Wave Interaction

19910021224 NASA Lewis Research Center, Cleveland, OH, USA
High temperature NASP engine seals: A technology review
Steinetz, Bruce M.; Dellacorte, Christopher; Tong, Mike; Aug 1, 1991; In English; 10th National Aerospace Plane Technology Symposium, 23-26 Apr. 1991, Monterey, CA, USA
Contract(s)/Grant(s): RTOP 763-21-41
Report No.(s): NASA-TM-104468; E-6317; NAS 1.15:104468; No Copyright; Avail: CASI; A03, Hardcopy

Progress in developing advanced high temperature engine seal concepts and related sealing technologies for advanced hypersonic engines are reviewed. Design attributes and issues requiring further development for both the ceramic wafer seal and the braided ceramic rope seal are examined. Leakage data are presented for these seals for engine simulated pressure and temperature conditions and compared to a target leakage limit. Basic elements of leakage flow models to predict leakage rates for each of these seals over the wide range of pressure and temperature conditions anticipated in the engine are also presented.
CASI
Aerospace Planes; Engine Design; High Temperature; Leakage; Lubrication; Sealing; Seals (Stoppers)

19910020811 Oklahoma Univ., Norman, OK, USA
An integrated aerodynamic/propulsion study for generic aero-space planes based on waverider concepts
Emanuel, G.; Rasmussen, M. L.; Aug 1, 1991; In English
Contract(s)/Grant(s): NAG1-886
Report No.(s): NASA-CR-188691; NAS 1.26:188691; No Copyright; Avail: CASI; A03, Hardcopy

Research efforts related to the development of a unified aerospace plane analysis based on waverider technology are summarized. Viscous effects on the forebodies of cone-derived waverider configurations were studied. A simple means for determining the average skin friction coefficient of laminar boundary layers was established. This was incorporated into a computer program that provides lift and drag coefficients and lift/drag ratio for on-design waveriders when the temperature and Reynolds number based on length are specified. An effort was made to carry out parabolized Navier-Stokes (PNS) calculations for cone-derived waveriders. When the viscous terms were turned off (in the Euler mode) computations for elliptic cone-derived waveriders could be carried out for a wide range of on-design and off-design situations. Work related to waveriders derived from power law shocks is described in some detail.
M.G.
Aerodynamic Characteristics; Aerospace Planes; Spacecraft Design; Spacecraft Propulsion; Waveriders

19910019135 NASA Lewis Research Center, Cleveland, OH, USA
Slush hydrogen propellant production, transfer, and expulsion studies at the NASA K-Site Facility
Hardy, Terry L.; Whalen, Margaret V.; JAN 1, 1991; In English; Conference on Advanced Space Exploration Initiative Technologies, 4-6 Sep. 1991, Cleveland, OH, USA
Contract(s)/Grant(s): RTOP 763-22-21
Report No.(s): NASA-TM-105191; E-6493; NAS 1.15:105191; AIAA PAPER 91-3550; No Copyright; Avail: CASI; A03, Hardcopy

Slush hydrogen is currently being considered as a fuel for the National AeroSpace Plane (NASP) because it offers the potential for decreased vehicle size and weight. However, no large scale data was available on the production, transfer, and tank pressure control characteristics required to use the fuel for the NASP. Therefore, experiments were conducted at NASA-Lewis K-Site Facility to improve the slush hydrogen data base. Slush hydrogen was produced using the evaporative cooling, or freeze-thaw, technique in batches for approx. 800 gallons. This slush hydrogen was pressure transferred to a 5 ft diameter spherical test tank following production, and flow characteristics were measured during this transfer process. The slush hydrogen in the test tank was pressurized and expelled using a pressurized expulsion technique to obtain information on tank pressure control for the NASP. Results from the production, transfer, pressurization, and pressurized expulsion tests are described.
CASI
Cryogenic Fluid Storage; Flow Characteristics; Hydrogen Production; Propellant Transfer; Slush Hydrogen

19910017819 NASA Langley Research Center, Hampton, VA, USA
Structural dynamic and aeroelastic considerations for hypersonic vehicles
Cazier, F. W., Jr.; Doggett, Robert V., Jr.; Ricketts, Rodney H.; Jun 1, 1991; In English; AIAA/ASME/ASCE/AHS/ASC 32d Structures, Structural Dynamics, and Materials Conference, 8-10 Apr. 1991, Baltimore, MD, USA
Contract(s)/Grant(s): RTOP 763-23-41
Report No.(s): NASA-TM-104110; NAS 1.15:104110; No Copyright; Avail: CASI; A03, Hardcopy

The specific geometrical, structural, and operational environment characteristics of hypersonic vehicles are discussed with particular reference to aerospace plane type configurations. A discussion of the structural dynamic and aeroelastic phenomena that must be addressed for this class of vehicles is presented. These phenomena are in the aeroservothermoelasticity technical area. Some illustrative examples of recent experimental and analytical work are given. Some examples of current research are pointed out.
CASI
Aerodynamic Configurations; Aerospace Planes; Aerothermoelasticity; Hypersonic Vehicles

19910017813 NASA Langley Research Center, Hampton, VA, USA
NASP aeroservothermoelasticity studies
Doggett, Robert V., Jr.; Ricketts, Rodney H.; Noll, T. E.; Malone, John B.; Apr 1, 1991; In English
Contract(s)/Grant(s): RTOP 763-23-41-71
Report No.(s): NASA-TM-104058; NAS 1.15:104058; No Copyright; Avail: CASI; A03, Hardcopy

Some illustrative results obtained from work accomplished under the aerothermoelasticity work breakdown structure (WBS) element of the National Aerospace Plane (NASP) Technology Maturation Program (TMP) are presented and discussed. The objectives of the aerothermoelasticity element were to develop analytical methods applicable to aerospace plane type configurations, to conduct analytical studies to identify potential problems, to evaluate potential solutions to problems, and to provide an experimental data base to verify codes and analytical trends. Work accomplished in the three areas of experimental data base, unsteady aerodynamics, and integrated analysis methodology are described. Some of the specific topics discussed are: (1) transonic wind tunnel aeroelastic model tests of cantilever delta wing models, of an all-moveable delta-wing model, and of aileron buzz models; (2) unsteady aerodynamic theory correlation with experiment and theory improvements; and (3) integrated analysis methodology results for thermal effects on vibration, for thermal effects on flutter, and for improving aeroelastic performance by using active controls.
CASI
Aeroservoelasticity; Aerospace Planes; Aerothermoelasticity; Data Bases; Delta Wings; National Aerospace Plane Program; Servocontrol; Unsteady Aerodynamics; Wind Tunnel Tests

19910011134 NASA Langley Research Center, Hampton, VA, USA
Addition of equilibrium air to an upwind Navier-Stokes code and other first steps toward a more generalized flow solver
Rosen, Bruce S.; Mar 1, 1991; In English
Contract(s)/Grant(s): RTOP 505-80-11-02
Report No.(s): NASA-TM-102616; NAS 1.15:102616; No Copyright; Avail: CASI; A04, Hardcopy

An upwind three-dimensional volume Navier-Stokes code is modified to facilitate modeling of complex geometries and flow fields represented by proposed National Aerospace Plane concepts. Code enhancements include an equilibrium air model, a generalized equilibrium gas model and several schemes to simplify treatment of complex geometric configurations. The code is also restructured for inclusion of an arbitrary number of independent and dependent variables. This latter capability is intended for eventual use to incorporate nonequilibrium/chemistry gas models, more sophisticated turbulence and transition models, or other physical phenomena which will require inclusion of additional variables and/or governing equations. Comparisons of computed results with experimental data and results obtained using other methods are presented for code validation purposes. Good correlation is obtained for all of the test cases considered, indicating the success of the current effort.
CASI
Aerospace Planes; Computational Fluid Dynamics; Flow Distribution; National Aerospace Plane Program; Navier-Stokes Equation

19910010870 Arizona State Univ., Tempe, AZ, USA
Dynamics of aerospace vehicles
Schmidt, David K.; Apr 1, 1991; In English
Contract(s)/Grant(s): NAG1-758
Report No.(s): NASA-CR-188015; NAS 1.26:188015; No Copyright; Avail: CASI; A06, Hardcopy

Papers on the following subjects are presented: (1) multivariable flight control synthesis and literal robustness analysis

for an aeroelastic vehicles; (2) numerical and literal aeroelastic-vehicle-model reduction for feedback control synthesis; and (3) dynamics of aerospace vehicles.
K.S.
Aeroelasticity; Aerospace Planes; Aerospace Vehicles; Aircraft Models; Control Systems Design; Dynamic Structural Analysis; Feedback Control; Flexible Spacecraft

19910008803 Boeing Co., Seattle, WA, USA
Personnel launch system autoland development study
Bossi, J. A.; Langehough, M. A.; Tollefson, J. C.; Jan 1, 1991; In English
Contract(s)/Grant(s): NAS1-18762; RTOP 906-11-01-01
Report No.(s): NASA-CR-187495; NAS 1.26:187495; No Copyright; Avail: CASI; A08, Hardcopy

The Personnel Launch System (PLS) Autoland Development Study focused on development of the guidance and control system for the approach and landing (A/L) phase and the terminal area energy management (TAEM) phase. In the A/L phase, a straight-in trajectory profile was developed with an initial high glide slope, a pull-up and flare to lower glide slope, and the final flare touchdown. The TAEM system consisted of using a heading alignment cone spiral profile. The PLS autopilot was developed using integral LQG design techniques. The guidance and control design was verified using a nonlinear 6 DOF simulation. Simulation results demonstrated accurate steering during the TAEM phase and adequate autoland performance in the presence of wind turbulence and wind shear.
CASI
Aerospace Planes; Approach Control; Automatic Landing Control; Automatic Pilots; Control Systems Design; Ferry Spacecraft; Spacecraft Guidance; Spacecraft Landing

19910007821 Loyola Coll., Baltimore, MD, USA
JTEC panel report on space and transatmospheric propulsion technology
Shelton, Duane; Aug 1, 1990; In English
Contract(s)/Grant(s): NSF ECS-89-02528
Report No.(s): NASA-CR-187670; NAS 1.26:187670; PB90-215732; No Copyright; Avail: CASI; A11, Hardcopy

An assessment of Japan’s current capabilities in the areas of space and transatmospheric propulsion is presented. The report focuses primarily upon Japan’s programs in liquid rocket propulsion and in propulsion for spaceplanes and related transatmospheric areas. It also includes brief reference to Japan’s solid rocket programs, as well as to supersonic air-breathing propulsion efforts that are just getting underway. The results are based upon the findings of a panel of U.S. engineers made up of individuals from academia, government, and industry, and are derived from a review of a broad array of the open literature, combined with visits to the primary propulsion laboratories and development agencies in Japan.
NTIS
Aerospace Planes; Japan; Jet Propulsion; Liquid Propellant Rocket Engines; Spacecraft Propulsion; Technology Assessment; Transatmospheric Vehicles

19910007799 NASA Ames Research Center, Moffett Field, CA, USA
The aerospace plane design challenge: Credible computational fluid dynamics results
Mehta, Unmeel B.; Dec 1, 1990; In English; 2nd AIAA International Aerospace Planes Conference, 29-31 Oct. 1990, Orlanda, FL, USA
Contract(s)/Grant(s): RTOP 505-60-00
Report No.(s): NASA-TM-102887; A-91016; NAS 1.15:102887; No Copyright; Avail: CASI; A03, Hardcopy

Computational fluid dynamics (CFD) is necessary in the design processes of all current aerospace plane programs. Single-stage-to-orbit (STTO) aerospace planes with air-breathing supersonic combustion are going to be largely designed by means of CFD. The challenge of the aerospace plane design is to provide credible CFD results to work from, to assess the risk associated with the use of those results, and to certify CFD codes that produce credible results. To establish the credibility of CFD results used in design, the following topics are discussed: CFD validation vis-a-vis measurable fluid dynamics (MFD) validation; responsibility for credibility; credibility requirement; and a guide for establishing credibility. Quantification of CFD uncertainties helps to assess success risk and safety risks, and the development of CFD as a design tool requires code certification. This challenge is managed by designing the designers to use CFD effectively, by ensuring quality control, and by balancing the design process. For designing the designers, the following topics are discussed: how CFD design technology is developed; the reasons Japanese companies, by and large, produce goods of higher quality than the U.S. counterparts;

teamwork as a new way of doing business; and how ideas, quality, and teaming can be brought together. Quality control for reducing the loss imparted to the society begins with the quality of the CFD results used in the design process, and balancing the design process means using a judicious balance of CFD and MFD.
CASI
Aerospace Planes; Computational Fluid Dynamics; Design Analysis; National Aerospace Plane Program; Supersonic Combustion

19910006700 Rice Univ., Houston, TX, USA
Optimal trajectories for an aerospace plane. Part 1: Formulation, results, and analysis
Miele, Angelo; Lee, W. Y.; Wu, G. D.; JAN 1, 1990; In English; American Control Conference, 23-25 May 1990, San Diego, CA, USA
Contract(s)/Grant(s): TATP-003604020; NAG1-1029
Report No.(s): NASA-CR-187868; NAS 1.26:187868; AAR-247; No Copyright; Avail: CASI; A04, Hardcopy

The optimization of the trajectories of an aerospace plane is discussed. This is a hypervelocity vehicle capable of achieving orbital speed, while taking off horizontally. The vehicle is propelled by four types of engines: turbojet engines for flight at subsonic speeds/low supersonic speeds; ramjet engines for flight at moderate supersonic speeds/low hypersonic speeds; scramjet engines for flight at hypersonic speeds; and rocket engines for flight at near-orbital speeds. A single-stage-to-orbit (SSTO) configuration is considered, and the transition from low supersonic speeds to orbital speeds is studied under the following assumptions: the turbojet portion of the trajectory has been completed; the aerospace plane is controlled via the angle of attack and the power setting; the aerodynamic model is the generic hypersonic aerodynamics model example (GHAME). Concerning the engine model, three options are considered: (EM1), a ramjet/scramjet combination in which the scramjet specific impulse tends to a nearly-constant value at large Mach numbers; (EM2), a ramjet/scramjet combination in which the scramjet specific impulse decreases monotonically at large Mach numbers; and (EM3), a ramjet/scramjet/rocket combination in which, owing to stagnation temperature limitations, the scramjet operates only at M approx. less than 15; at higher Mach numbers, the scramjet is shut off and the aerospace plane is driven only by the rocket engines. Under the above assumptions, four optimization problems are solved using the sequential gradient-restoration algorithm for optimal control problems: (P1) minimization of the weight of fuel consumed; (P2) minimization of the peak dynamic pressure; (P3) minimization of the peak heating rate; and (P4) minimization of the peak tangential acceleration.
CASI
Aerodynamic Characteristics; Aerospace Planes; Hypersonic Speed; Hypersonics; Optimization; Supersonic Speed; Trajectories

19910006698 Rice Univ., Houston, TX, USA
Optimal trajectories for an aerospace plane. Part 2: Data, tables, and graphs
Miele, Angelo; Lee, W. Y.; Wu, G. D.; JAN 1, 1990; In English; 1990 American Control Conference, 23-25 May 1990, San Diego, CA, USA
Contract(s)/Grant(s): NAG1-1029
Report No.(s): NASA-CR-187848; NAS 1.26:187848; AAR-248-PT-2; No Copyright; Avail: CASI; A05, Hardcopy

Data, tables, and graphs relative to the optimal trajectories for an aerospace plane are presented. A single-stage-to-orbit (SSTO) configuration is considered, and the transition from low supersonic speeds to orbital speeds is studied for a single aerodynamic model (GHAME) and three engine models. Four optimization problems are solved using the sequential gradient-restoration algorithm for optimal control problems: (1) minimization of the weight of fuel consumed; (2) minimization of the peak dynamic pressure; (3) minimization of the peak heating rate; and (4) minimization of the peak tangential acceleration. The above optimization studies are carried out for different combinations of constraints, specifically: initial path inclination that is either free or given; dynamic pressure that is either free or bounded; and tangential acceleration that is either free or bounded.
CASI
Aerospace Planes; Algorithms; Engine Design; Hypersonic Flight; Mathematical Models; Optimal Control; Supersonic Speed; Trajectory Optimization

19900016844 NASA Lewis Research Center, Cleveland, OH, USA

Prediction of the ullage gas thermal stratification in a NASP vehicle propellant tank experimental simulation using FLOW-3D

Hardy, Terry L.; Tomsik, Thomas M.; Jul 1, 1990; In English

Contract(s)/Grant(s): RTOP 763-01-21

Report No.(s): NASA-TM-103217; E-5629; NAS 1.15:103217; No Copyright; Avail: CASI; A03, Hardcopy

As part of the National Aero-Space Plane (NASP) project, the multi-dimensional effects of gravitational force, initial tank pressure, initial ullage temperature, and heat transfer rate on the 2-D temperature profiles were studied. FLOW-3D, a commercial finite difference fluid flow model, was used for the evaluation. These effects were examined on the basis of previous liquid hydrogen experimental data with gaseous hydrogen pressurant. FLOW-3D results were compared against an existing 1-D model. In addition, the effects of mesh size and convergence criteria on the analytical results were investigated. Suggestions for future modifications and uses of FLOW-3D for modeling of a NASP tank are also presented.

CASI

Aerospace Planes; Computer Programs; Fluid Flow; Heat Transfer; National Aerospace Plane Program; Propellant Tanks; Ullage

19900016739 NASA Lewis Research Center, Cleveland, OH, USA

Background, current status, and prognosis of the ongoing slush hydrogen technology development program for the NASP

Dewitt, R. L.; Hardy, T. L.; Whalen, M. V.; Richter, G. P.; Tomsik, T. M.; JAN 1, 1990; In English; 8th World Hydrogen Energy Conference, 22-27 Jul. 1990, Honolulu, HI, USA

Contract(s)/Grant(s): RTOP 763-01-21

Report No.(s): NASA-TM-103220; E-5634; NAS 1.15:103220; No Copyright; Avail: CASI; A03, Hardcopy

Among the Hydrogen Projects at the NASA Lewis Research Center (NASA LeRC), is the task of implementing and managing the Slush Hydrogen (SLH2) Technology Program for the USA' National AeroSpace Plane Joint Program Office (NASP JPO). The objectives of this NASA LeRC program are to provide verified numerical models of fluid production, storage, transfer, and feed systems and to provide verified design criteria for other engineered aspects of SLH2 systems germane to a NASP. The pursuit of these objectives is multidimensional, covers a range of problem areas, works these to different levels of depth, and takes advantage of the resources available in private industry, academia, and the U.S. Government. A summary of the NASA LeRC overall SLH2 program plan, is presented along with its implementation, the present level of effort in each of the program areas, some of the results already in hand, and the prognosis for the effort in the immediate future.

CASI

Aerospace Planes; Design Analysis; Mathematical Models; National Aerospace Plane Program; Project Planning; Slush Hydrogen

19900009959 Georgia Inst. of Tech., Atlanta, GA, USA

Rapid near-optimal trajectory generation and guidance law development for single-stage-to-orbit airbreathing vehicles

Calise, A. J.; Flandro, G. A.; Corban, J. E.; Jan 1, 1990; In English

Contract(s)/Grant(s): NAG1-922

Report No.(s): NASA-CR-186299; NAS 1.26:186299; No Copyright; Avail: CASI; A04, Hardcopy

General problems associated with on-board trajectory optimization, propulsion system cycle selection, and with the synthesis of guidance laws were addressed for an ascent to low-earth-orbit of an air-breathing single-stage-to-orbit vehicle. The NASA Generic Hypersonic Aerodynamic Model Example and the Langley Accelerator aerodynamic sets were acquired and implemented. Work related to the development of purely analytic aerodynamic models was also performed at a low level. A generic model of a multi-mode propulsion system was developed that includes turbojet, ramjet, scramjet, and rocket engine cycles. Provisions were made in the dynamic model for a component of thrust normal to the flight path. Computational results, which characterize the nonlinear sensitivity of scramjet performance to changes in vehicle angle of attack, were obtained and incorporated into the engine model. Additional trajectory constraints were introduced: maximum dynamic pressure; maximum aerodynamic heating rate per unit area; angle of attack and lift limits; and limits on acceleration both along and normal to the flight path. The remainder of the effort focused on required modifications to a previously derived algorithm when the model complexity cited above was added. In particular, analytic switching conditions were derived which, under appropriate assumptions, govern optimal transition from one propulsion mode to another for two cases: the case in which engine cycle operations can overlap, and the case in which engine cycle operations are mutually exclusive. The resulting guidance

algorithm was implemented in software and exercised extensively. It was found that the approximations associated with the assumed time scale separation employed in this work are reasonable except over the Mach range from roughly 5 to 8. This phenomenon is due to the very large thrust capability of scramjets in this Mach regime when sized to meet the requirement for ascent to orbit. By accounting for flight path angle and flight path angle rate in construction of the flight path over this Mach range, the resulting algorithm provides the means for rapid near-optimal trajectory generation and propulsion cycle selection over the entire Mach range from take-off to orbit.
CASI
Aerospace Planes; Hypersonic Vehicles; Propulsion System Performance; Single Stage to Orbit Vehicles; Spacecraft Guidance; Spacecraft Propulsion; Spacecraft Trajectories; Trajectory Optimization

19900009436 NASA Langley Research Center, Hampton, VA, USA
Structures and materials technology for hypersonic aerospacecraft
Mccomb, Harvey G., Jr.; Murrow, Harold N.; Card, Michael F.; Jan 1, 1990; In English
Contract(s)/Grant(s): RTOP 506-43-71-04
Report No.(s): NASA-TM-102583; NAS 1.15:102583; No Copyright; Avail: CASI; A03, Hardcopy

Major considerations in structural design of a transatmospheric aerospacecraft are discussed. The general direction of progress in structures and materials technology is indicated, and technical areas in structures and materials where further research and development is necessary are indicated. Various structural concepts under study and materials which appear to be most applicable are discussed. Structural design criteria are discussed with particular attention to the factor-of-safety approach and the probabilistic approach. Structural certification requirements for the aerospacecraft are discussed. The kinds of analyses and tests which would be required to certify the structural integrity, safety, and durability of the aerospacecraft are discussed, and the type of test facility needed to perform structural certification tests is identified.
CASI
Aerospace Planes; Aerospace Vehicles; Certification; Hypersonics; Structural Design; Structural Design Criteria; Structural Engineering

19900007739 McDonnell Aircraft Co., Saint Louis, MO, USA
Heat pipes for wing leading edges of hypersonic vehicles
Boman, B. L.; Citrin, K. M.; Garner, E. C.; Stone, J. E.; Jan 1, 1990; In English
Contract(s)/Grant(s): NAS1-18144; RTOP 506-49-11-05
Report No.(s): NASA-CR-181922; NAS 1.26:181922; No Copyright; Avail: CASI; A06, Hardcopy

Wing leading edge heat pipes were conceptually designed for three types of vehicle: an entry research vehicle, aero-space plane, and advanced shuttle. A full scale, internally instrumented sodium/Hastelloy X heat pipe was successfully designed and fabricated for the advanced shuttle application. The 69.4 inch long heat pipe reduces peak leading edge temperatures from 3500 F to 1800 F. It is internally instrumented with thermocouples and pressure transducers to measure sodium vapor qualities. Large thermal gradients and consequently large thermal stresses, which have the potential of limiting heat pipe life, were predicted to occur during startup. A test stand and test plan were developed for subsequent testing of this heat pipe. Heat pipe manufacturing technology was advanced during this program, including the development of an innovative technique for wick installation.
CASI
Fabrication; Heat Pipes; Hypersonic Vehicles; Leading Edges; Wings

Subject Term Index

Personal Author Index

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