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DELTA, AND UPRATED DELTA, CAPABILITIES, CONSTRAINTS, COSTS

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GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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Abstract

The current Delta and proposed Uprated Delta, are described for potential users. Performance, flight environment, spacecraft integration requirements, launch availability and costs are provided. Delta is composed of a Long Tank Thor first stage, Delta second stage (evolved from the Vanguard and Able-Star programs), and a solid propellant third stage motor that is an adaptation of the Surveyor spacecraft retro-motor. Uprated Delta is composed of the present first and third stages and fairing, integrated with a new hydrogen/oxygen second stage that utilizes shortened Thor tanks, Saturn IVB hydrogen tank internal insulation, a single Centaur RL-10 engine, and Delta second stage control electronics and peripheral tracking, data, range safety, and power systems. Uprated Delta increases the synchronous transfer capability from 785 to 2395 pounds. A Delta launch cost of $4.0 million is detailed. The projected Uprated Delta launch cost of $5.5 million and its $30 million estimated development cost are discussed. The Uprated Delta is being proposed to NASA for consideration in its launch vehicle development programs. It is not at this time an officially approved program.

I. Introduction

The evolution of the Delta launch vehicle reaches back thirteen years when, in 1955, the United States participated in the International Geophysical Year (IGY) and undertook the development of the Vanguard three-stage launch vehicle; in the same year the Air Force initiated the development of the Thor IRBM. With modifications, the Thor became the first stage of Delta; the Vanguard second stage propulsion system, evolved through the Able programs, became the Delta second stage propulsion system; and the Vanguard X-248 third stage solid propellant rocket motor was adapted as the third stage for Delta. The development and integration of these systems and the production of twelve (12) vehicles was started in early 1959 under prime contract to the Douglas Aircraft Company. The initial objective of the Delta program was to provide an interim space launch vehicle capability for the medium-class payloads until more sophisticated vehicles as Scout and Agena, then under development, could be brought to operational status. The development program spanned 18 months. In a little over two years, following the development period, eleven of the twelve vehicles were launched successfully carrying, among others, the first passive communications satellite, Echo I (August 1960) and the first private industry satellite, AT&T's Telstar I (July 1962). The total development cost, including the twelve vehicles (Model DM-19) and launch support, was approximately $43,000,000, compared to the $40,000,000 estimated at the outset of the program.

Before the development program was complete the number of missions planned for Delta outstripped the interim buy of twelve vehicles, so an order was placed for fourteen production vehicles. This follow-on buy of Deltas (Models A and B) incorporated lengthened second stage propellant tanks, a higher energy second stage oxidizer, transistorized guidance electronics, and assiduous application of high-reliability semiconductors in flight critical circuits. This model of Delta carried NASA's first active communications satellite, Relay I (December 1962), the second AT&T Telstar (May 1963), and the first synchronous satellites, Syncom I and II (February and July 1963).

The next production order of Deltas (Models C and D) in 1963 brought the adaption of the USAF developed improved Thor booster with thrust augmentation provided by three strap-on solid propellant motors and the adaption of the Scout developed X-258 to replace the X-248 third stage motor. The first thrust augmented Delta (TAD) carried Syncom III (August 1964), the first equatorial synchronous communications satellite. The second TAD vehicle orbited the first commercial communications satellite, Comsat Corporation's Early Bird Satellite (April 1965).

Another order of Deltas in 1964 brought the development of the Improved Delta (Model E). The Improved Delta model adapted and extended the large diameter propellant tanks from the Able-Star stage, and thereby nearly doubled the propellant capacity of the previous Delta second stage. The larger diameter tanks in addition permitted adaption of the five foot diameter Nimbus fairing developed for the Agena stage. Improved Delta also adapted the USAF developed FW-4 solid propellant motor to replace the X-258 third stage motor. The first Improved Delta was launched November 1965 and has performed thus far without failure. Missions have ranged from injecting the 1000 pound Biological Satellite into a 175 nautical mile (n.m.) orbit and the 175 pound Anchored Interplanetary Monitoring Probe (A-IMP) into orbit around the moon to injecting the Intelsat II series of satellites to synchronous altitude.

II. The Delta Launch Vehicle

The thread of the evolution of communications satellites crosses Delta's at a key milestone again when a new Delta (Model M) will launch the first Intelsat III or British Skynet series of communications satellites. Delta is shown in Figure 1 and incorporates the USAF newly developed Long Tank Thor first stage, the Improved

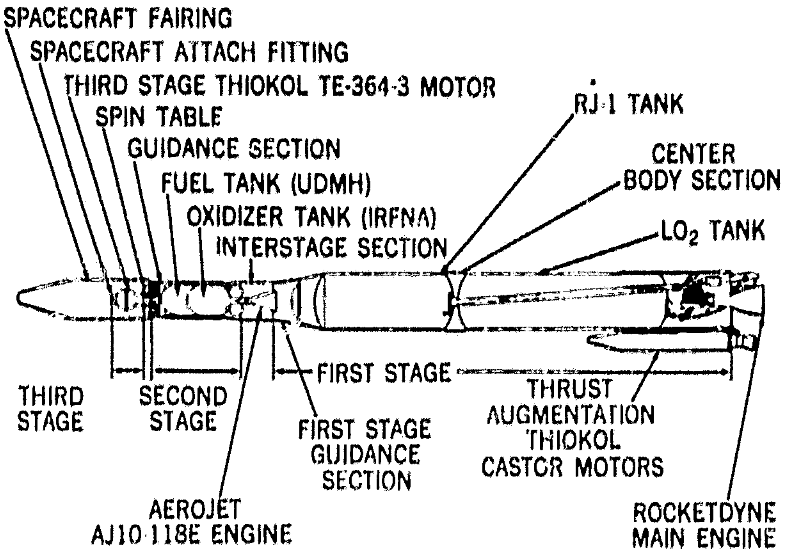

Figure 1. Delta (Model M)

Delta second stage, and substitutes for the FW-4 third stage motor an adaptation of the retro-motor from the Surveyor spacecraft. This current evolution of Delta is described together with its performance, flight environment, the spacecraft mission milestones, and launch costs.

A. Vehicle Description

The three-stage Delta vehicle (Figure 1) stands 106 feet, weighs 200,400 pounds at lift-off, can ascend through a 99 percent Eastern Test Range (ETR) and Western Test Range (WTR) upper atmosphere annual wind profile, lift-off in 40 knot ground winds and hold on the pad for hours in launch readiness to meet a launch window only seconds wide.

The first stage is the reliable Thor system used since the first Delta. Thor recently celebrated its 100th consecutive successful boost of space vehicles. The Long Tank Thor, with extended liquid oxygen tank and RP-1 fuel tank converted to a constant 8 foot diameter, stands 60 feet and carries 147,000 pounds of propellants, or about 47 percent more propellants than previous models. Thrust augmentation is provided by three Thiokol Castor II solid motors (TE-354-5) that ignite at lift-off, burnout in 40 seconds and are jettisoned at 60 seconds. Jettison is effected by firing an explosive bolt holding a clamped ball-socket joint. The relative acceleration of the vehicle plus aerodynamic drag on the spent motor ejects the cases away from the booster. The turbopump fed Rocketdyne main engine develops 175,000 pounds thrust at lift-off and the three Castor II solids develop 52,000 pounds thrust each. At an altitude between 60 and 70 n.m. the main engine burns to propellant depletion about 220 seconds after lift-off. A prepunched film tape in the programmer initiates sequential functions and torques the autopilot to guide the first stage through a predetermined trajectory. Pitch and yaw steering commands to the autopilot are overriden by the Western Electric Co. (Weco) ground radio-guidance system. This system trims out the small deviation of the vehicle from the nominal trajectory by sending corrective steering commands whenever the ground guidance tracking station senses off-course flight.

During powered flight pitch and yaw steering is exerted by gimballing the main engine. Roll control is effected by a pair of small outboard vernier engines that are gimballed differentially. Subsequent to main engine shut-down the verniers continue to operate for about twelve seconds, damping shut-down transients and stabilizing the vehicle for staging of the second stage.

The interstage section between the first and second stages is provided with exhaust and access ports to allow the escape of exhaust gases during second stage engine start in the interstage. These ports are covered in flight with band assemblies that are jettisoned (Figure 2) at first stage main engine shut-down. Four seconds later explosive bolts that attach the two stages are fired and the second stage engine is started.

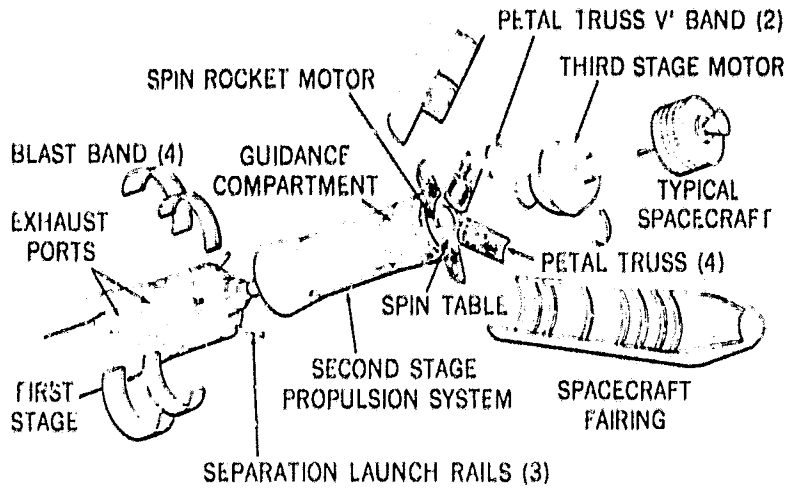

Figure 2. Delta Separation Schematic

The Delta second stage is 17 feet long, approximately 5 feet in diameter and weighs 12,600 pounds at ignition. The pressure fed Aerojet engine delivers 7500 pounds thrust and operates for about 400 seconds on storable propellants. A flight control system stabilizes the vehicle and initiates flight functions that guide the vehicle through a predetermined powered trajectory and coast flight. The programmer in the flight controller has the capability of supplying five discrete steering commands and six flight-sequence signals: engine start, fairing jettison, enable engine cut-off circuitry, third stage spin-up and start of its pyrotechnic time delay ignition squib, and third stage separation. The maximum length of the programmer time is 4096 seconds from second stage start signal, or equivalent to about two thirds of one revolution in a 100 nautical mile circular parking orbit before third stage spin-up, and separation must be commanded. During powered flight steering commands from the programmer are overriden by the ground radio guidance system if the vehicle deviates from the nominal trajectory. Since second stage engine cut-off occurs over the radio horizon from the ground guidance tracking station at the launch site, radio guidance is active only through about half of the second stage powered flight. Prior to reaching the horizon the radio guidance ground station sends a command signal to start an onboard integrating accelerometer system that will cut-off the engine when a pre-set velocity to be gained is reached. This start command signal is sent at the time when the required second stage velocity to be gained as determined by the radio-guidance ground tracking, equals the preset value in the integrating accelerometer system. Also just prior to the time the integrating accelerometer is started the ground guidance station sends a final set of discrete steering commands to adjust the second stage attitude during the remaining powered flight so that after cut-off and coast the third stage will be pointed in the correct direction to compensate for accumulated first and second stage attitude or velocity errors.

During powered flight pitch and yaw steering is provided by gimballing the second stage engine and roll is controlled by cold nitrogen gas jets. Cold nitrogen gas jets control the vehicle in all axes during coast. The control system electrical power and nitrogen gas supply is capable of maintaining second stage attitude for a little over two hours. For long second stage coast periods before third stage spin-up and separation the second stage may be reoriented with respect to the sun or the vehicle placed in a slow yawing or pitching tumble to alleviate assymetric solar heating of the spacecraft.

Peripheral second stage systems include a 'C' band tracking beacon, a PDM/FM/FM 15 × 20 telemetry system, dual command destruct receivers and associated power supplies. The command destruct receiver system is capable of both cutting-off the second stage engine or destroying the errant vehicle. In special instances where non-catastrophic flight failures of the first or second stages could still permit the spacecraft to achieve an orbit with a nominal third stage operation, Range Safety is provided displays on the tracking plot boards that permit the Range Safety Officer to determine that an orbit can be achieved without hazard of land impact from any of the stages. In this instance only second stage engine cut-off would be commanded so possibly all or part of the mission may be salvaged.

The third stage assembly consists of a spin-table, the Thiokol TE-364-3 solid propellant motor, spacecraft attach fitting, spacecraft and the spacecraft fairing. The spin-table consists of a bearing support structure and a conical third stage motor pedestal truss that is divided into four petals hinged at the base and clamped to the equator of the motor by a 'V' band. The 'V' band is held in tension by two explosive bolts that are fired two seconds after the motor and spacecraft are spun-up and the 15 second time delay squib that ignites the TE-364-3 motor is started. The released petals fly outward under centrifugal force, releasing the third stage from the spin table (Figure 2). At the same instant the second stage is backed away from the free spinning third stage by venting propellant residual pressurant (helium) overboard through two retro-jets. Approximately thirteen seconds later the third stage motor is ignited by the time delay squib. The TE-364-3 burns for approximately 41 seconds and develops an average thrust of 9,000 pounds.

Torque to the spin table is imparted by combinations of small solid propellant rocket motors, which provide spin rates from 30 to 100 rpm (±10 percent) for spacecraft roll moment of inertia ranging from 20 to 170 slug-feet squared. A lower limit of approximately 30 rpm is dictated by minimum dynamic stability of the third stage/spacecraft assembly during third stage motor burning. If less than 30 rpm is desired the effect upon orbit injection errors must be carefully assessed. The anticipated maximum spin rate users would desire was 100 rpm, consequently the third stage motor was qualified only up to this spin rate.

The spacecraft is clamped to the attach fitting by a circular V-clamp-band assembly that releases by firing two explosive bolt cutters subsequent to third stage motor burn-out. Separation from the expended third stage is then effected by a separation spring, or springs, which provides the spacecraft with a relative separation velocity of 6 to 8 fps with respect to the expended third stage motor. Although peculiar spacecraft requirements may dictate the design of a special spacecraft attach fitting, a number of standard Delta fittings are available. These are specified in Table I. These fittings use either a small rocket or yo weight system to tumble the expended third stage motor after spacecraft separation to preclude possible motor outgassing from accelerating it into the spacecraft. Also available is a yo-yo weight despin system which can despin the third stage and spacecraft combination prior to spacecraft separation. Attach fittings include timer assemblies, battery, and delay squib switches. The timers are initiated by the second stage programmer and run on mechanical energy until reaching a predetermined time to fire the spacecraft separation clamp-band bolt cutters. Two seconds later the squib switches initiate a small rocket or yo weight to tumble the expended third stage motor.

Table I
Delta Standard Attach Fittings

Spacecraft Interface Diameter, Inches	Height Inches	Weight* Pounds	Flt. Qualified On	Cost Thousand Dollars
18	9.5	24	GEOS Series	28
20	30	25	Intelsat II Series	28
25	12	24.5	Intelsat III Series	24
37	31	44.5	TIROS M Series	30

*Includes electrical systems for separation and tumble

The spacecraft fairing is fiberglass, and constructed in two half-shells that are brought up around the spacecraft laterally and clamped together by three strap assemblies that are released in flight by explosive bolts. Spring cartridges thrust the half-shells laterally and pivots at the base of the fairing cause the shells to rotate rearwards and clear of the vehicle (Figure 2). Normally the fairing is jettisoned within 5 to 20 seconds after second stage start. Fairing jettison time is dictated by the free molecular heating rate that can be tolerated by the spacecraft. Normally the heating rate is held below 0.1 BTU/ft^2-sec. or about equivalent to the solar heating rate to the spacecraft. Aerodynamic heating of the fairing is controlled by application of ablative materials to hold the fairing internal temperature to below 450°F. This precludes any possibility of spacecraft contamination from outgassing of the fiberglass phenolic.

Access ports through the fairing are provided at the locations that meet the needs of the vehicle user. The

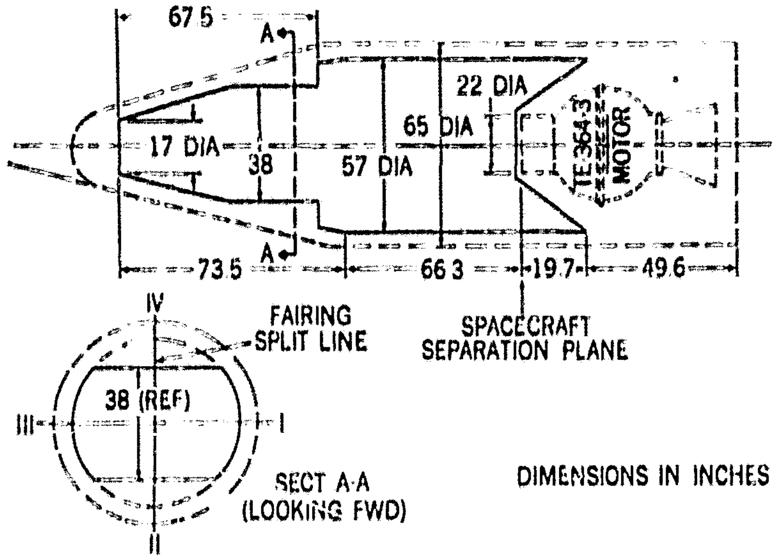

Figure 3. Delta Fairing Spacecraft Envelope

available fairing internal envelop is shown in Figure 3.

B. Flight Sequence and Performance

The Delta flight sequences for a synchronous transfer mission having a perigee altitude of 130 nautical miles (n.m.) an apogee altitude of 19,400 n.m. and an inclination of 33 degrees is shown in Figure 4. The vehicle is

NO	TIME (SEC)	EVENT	NO	TIME (SEC)	EVENT
0	0	LIFT OFF	5	470	START ACCELEROMETER
1	60	JETTISON SOLID MOTORS	6	600	SECOND STAGE CUT OFF
2	218	FIRST STAGE BURNOUT	7	1171	THIRD STAGE IGNITION
3	222	SECOND STAGE IGNITION	8	1212	THIRD STAGE BURN OUT
4	228	FAIRING JETTISON	9	1771	SPACECRAFT/THIRD STAGE SEPARATION

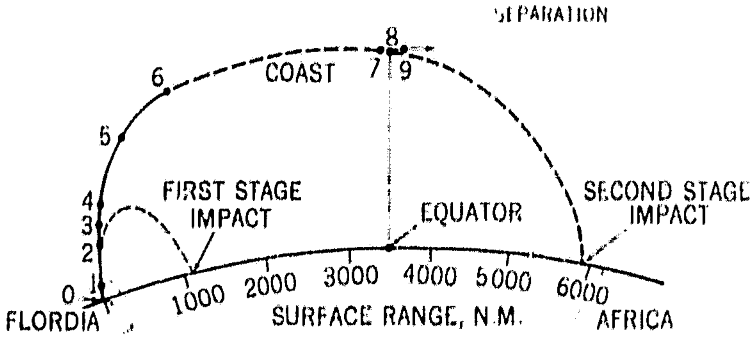

Figure 4. Delta Flight Sequence of Events for a Synchronous Transfer Mission

launched from ETR on an azimuth of 108 degrees. The pitch program consists of five discrete first stage pitch rates, two second stage rates and a coast phase rate. The first and second stages have insufficient performance to carry the third stage assembly into orbit, so after second stage engine cut-off the stage coasts in a sub-orbital elliptical trajectory to a point just short of the equator where third stage spin-up, separation and ignition occur. The third stage burns out directly over the equator at an altitude of 130 n.m., an inertial flight path angle of zero degrees, and with sufficient velocity to coast the spacecraft to an altitude of 19,400 n.m. on the opposite side of the earth so that the line of apsides lies in the equatorial plane to permit the spacecraft apogee motor to rotate the transfer orbital plane into the equatorial plane as part of the circularization maneuver. If it were not for the Range Safety constraint of keeping the impact of the second stage off of Africa the vehicle could place about 30 more pounds of spacecraft into orbit. Considering this constraint Delta can inject 785 pounds of payload into the synchronous transfer orbit described.

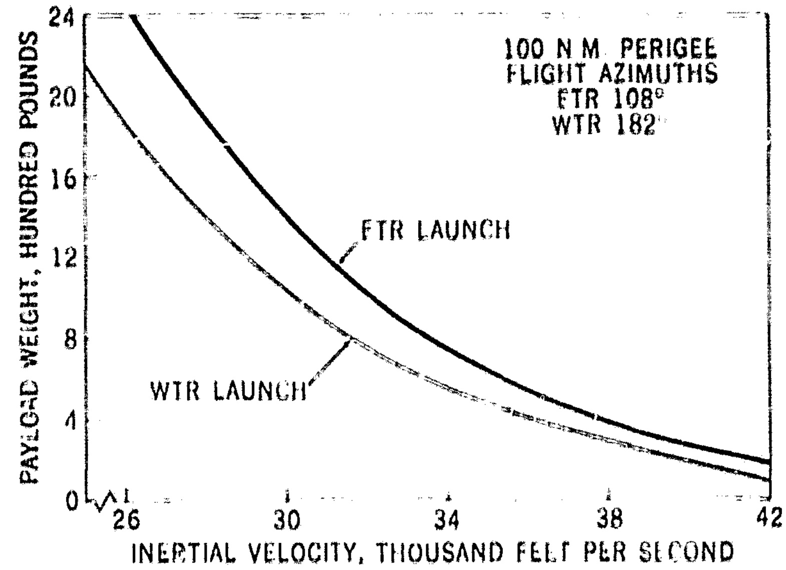

Figure 5. Characteristic Velocity for Delta (Model M)

Payload weight vs characteristic inertial velocity for Delta from ETR and WTR is shown in Figure 5. Payload weight includes the spacecraft attach fitting and spacecraft. The performance shown does not include considerations for Range Safety.

The injection accuracy of Delta is strongly dependent on the trajectory profile. The single largest injection error source is the unguided, spin-stabilized third stage. Nearly two thirds of the errors in injection velocity and attitude are caused by dispersions in the motor total impulse and lateral tip-off impulses applied during separation from the second stage and at motor ignition. Typically the 99 percent probability dispersions for a nominal 130 n.m. by 19,400 n.m. synchronous transfer trajectory is shown in Table II.

Table II
Synchronous Transfer Orbit Dispersions

Parameter	Nominal	99% Probable Disperson
Apogee Altitude, n.m.	19,400	±900
Perigee Altitude, n.m.	130	±30
Orbit Period, minutes	633	±31
Orbit Eccentricity	0.73	±0.01
Orbit Inclination, degrees	33	±0.70

Despite that these dispersions appear large it should be remembered that since a synchronous communications satellite must carry a propulsion system for station keeping the penalty paid in additional propellant to trim out injection errors is small.

C. Flight Environment

The spacecraft environment is estimated from previous flight measurements. The Delta critical environmental inputs to the spacecraft is expected to be essentially identical to previous models, with the exception of the last thirty seconds of the long tank Thor boost flight. At this time and lift-off the spacecraft is subjected to the most severe lateral and longitudinal sinusoidal vibration that may appreciably load the spacecraft structure dynamically. At the time the vehicle leaves the launch pins and umbilicals simultaneously retract at stations along the length of the vehicle, the spacecraft can experience a maximum of ±2.0 g, zero-to-peak (O-P) in the vehicle lateral modal frequencies (2 to 16 cps). Superimposed at this time is a ±3.0 g (O-P) longitudinal 13 cps oscillation. These combined lift-off oscillations typically last for two to five seconds with the peak acceleration lasting one to two cycles. The last thirty seconds of first stage flight the Thor exhibits a 20 cps "pogo" longitudinal oscillation thatbuilds-up to ±4 g (O-P) at the time the steady state longitudinal acceleration has reached about 6.5 g. The maximum first stage steady stage acceleration of 8 g's is the highest imposed on spacecrafts weighing 1000 pounds or greater. For lighter spacecraft the maximum steady stage acceleration is dictated by the TE-364-3 third stage and reaches 16 g's for a 500 pound spacecraft.

Above 30 cps no appreciable (less than 0.2 g) sinusoidal vibration is present. Random vibration measured at the third stage attach fitting and spacecraft, however, show power spectrum densities between 0.001 and 0.005 $g's^2$/cps from 20 cps to 2000 cps in both lateral and longitudinal axes. The principal source of random acceleration is boundary layer turbulance over the fairing and the reverse slope of the second stage guidance compartment that excites the structure and feeds up through the third stage assembly to the base of the spacecraft. Acoustical excitation also contributes to the random levels experienced.

At lift-off and transonic the overall acoustical level inside the fairing is approximately 140 db (referenced to 0.0002 dynes/cm^2) from 37.5 to 9600 cps. These levels are present for about 10 seconds at liftoff and again for about 15 seconds at transonic.

Shocks occur at main engine start, thrust augmentation solid motors ignition and jettison, staging, fairing jettison, and spacecraft separation from the expended third stage. For three stage Delta, cutting the bolts to separate the spacecraft from the expended third stage imposes the most severe shock spectrum on the spacecraft. The third stage motor and spin table assembly act to absorb the high frequency excitation from other sources. Cutting the separation bolts results in an estimated shock spectrum equivalent to a one-half millisecond, 1600 g terminal peak saw tooth input.

D. Spacecraft Integration/Launch Operations

Definition of performance capabilities, flight events, orbital accuracies, vehicle interfaces, and launch operations for potential vehicle users starts as soon as the concept of the mission is outlined. This in some instances has been two and three years in advance of actual mission commitment. This advanced and continued coordination while developing mission specifications provides visibility to both the Spacecraft and Delta Project that normally reveals compatibility problem areas before final definition of the spacecraft and vehicle interfaces. In general, however, vehicle planning for new missions follow the pattern outlined in Table III and starts about a year prior to launch (L-52 weeks) when the Spacecraft Project provides a definition of the preliminary spacecraft configuration, mass properties and trajectory and orbital requirements for preliminary vehicle performance evaluation analysis. A preliminary trajectory with attendant injection error studies and thermal studies is completed within eight weeks. With this visability the Delta Project and the Spacecraft Project jointly develop a Detail Mission Definition specification (L-39) that includes such constraints as spacecraft orbital lifetime, apogee and perigee altitude and geocentric location, permissible injection errors, injection attitude orientation, launch window criteria, tracking and data retrieval requirements, spacecraft mass properties, and all other data necessary for the preparation of a reference trajectory and detailed error analysis.

Table III

Milestone	Provided By	Weeks Prior to Launch (L-Weeks)
Preliminary Mission Definition	Spacecraft Project	52
Preliminary Trajectory, Error and Thermal Studies	Delta	44
Detail Mission Definition	Spacecraft Project	39
Fairing, S/C to Blockhouse Wiring, and Separation Sequence Requirements	Spacecraft Project	35
Spacecraft Thermal Input Constraints	Spacecraft Project	30
Mission Reference Trajectory and Preliminary Orbital Error Analysis	Delta	35
S/C RF Characteristics and Spin Rate Requirements	Spacecraft Project	24
Final Mission Error Analysis	Delta	23
Vehicle/Spacecraft Compatibility Drawing	Delta	20
Compatibility Fit Check of S/C and Vehicle Interface	Delta	16
Final Thermal and Fairing Drop Time Analysis	Delta	16
Spacecraft Handling Plan and Countdown Tasks	Delta & S/C Project	8

Table III (Continued)

Milestone	Provided by	Weeks Prior to Launch (L-Weeks)
S/C Delivery at Launch Site	Spacecraft Project	3
S/C Mated to Vehicle	Delta	1-2
Final Weights	Delta	1
Launch	Delta	0

The Reference Trajectory is provided to the Spacecraft Project eight weeks later (L-35) and includes all the technical data defining the flight mode, sequence of flight events, vehicle weights and propulsion system characteristics. Tabulations of trajectory parameters, weight history, radar look angles, and instantaneous impact loci are provided together with a preliminary analysis of the expected dispersions in orbital parameters. Final definition of the maximum and minimum allowable spin rate, spacecraft RF systems, and permissible in-flight thermal inputs are provided by L-24 weeks. A full scale compatibility drawing based on the Spacecraft Project's final configuration drawings is prepared normally at L-20 weeks. This drawing is primarily to show all clearance between the spacecraft and fairing, attach fitting, and third stage motor and locate the orientation of such features as umbilical connectors, access ports through the fairing, and any special interface wiring between the attach fitting and spacecraft. A hardware compatibility fit check is conducted, when required, about four weeks later (L-16) with third stage flight hardware and the prototype or flight spacecraft. A Spacecraft Handling Plan is jointly developed and finalized about L-8 weeks and describes all hazardous systems, spacecraft test procedures, and details pre-launch work schedules. Typically the spacecraft arrives at the launch site three weeks before launch and is built-up on the third stage motor assembly the following week and mated to the vehicle on the pad one to two weeks before launch for RFI testing with the vehicle and Range RF systems. Final weights are inputed to trim the final trajectory and radio-guidance hand-set parameters the week before launch.

The spacecraft must be staticly and dynamically balanced prior to receipt at the launch site. The allowable spacecraft center-of-gravity offset and principal axis misalignment is 0.015 inches and 0.002 radians, respectively. For missions where injection attitude is extremely critical for mission success a third stage assembly composite spin balance is conducted at the launch site.

The Delta final launch countdown is spaced over three days. Spacecraft and vehicle checkout are interspersed throughout this period and are scheduled to accommodate spacecraft checkout requirements. If necessary complete access to the spacecraft can be provided up to four hours prior to lift-off, though normally the fairing is installed about 12 to 16 hours prior to launch. Provisions to continuously power and monitor the spacecraft from the blockhouse is provided through the vehicle wiring.

Launches off the same pad at ETR can be provided on three week centers. A called-up launch can be made on 90 days notice at no increase in launch costs provided it is an identical mission, mission peculiar hardware (attach fitting, etc.) have been provided and the launch does not impact another scheduled mission. Call-up time may be reduced to 60 days at a cost of about $100,000 for factory and launch checkout overtime or to 30 days, provided the vehicle has been previously configured for the mission and completed factory checkout in anticipation of call-up. The 30 day option, however, requires commitment of about $175,000 of non-recoverable funds if call-up is not exercised. Based on past experience at ETR the probability of launching in a window 15 seconds wide on a given day is 66 percent. The probability for a thirty minute launch window, typical for a synchronous transfer mission, is 82 percent.

E. Cost

The cost of a Delta launch from ETR today averages about $4.0 million. This includes hardware, trajectory software, spacecraft integration, launch and range support services, and NASA administrative charges. A breakdown is provided in Table IV, and is based on actual or estimated expenses billed to outside agency users such as ESSA, Comsat, or ESRO for reimbursement to NASA.

Table IV
Delta Launch Costs (1968)

	Initial Launch	Follow-on Launch
	(Thousand Dollars)	
Hardware		
First Stage	$1,200	$1,200
Second Stage & Fairing	1,050	1,050
Third Stage	90	90
Attach fitting	30	30
Launch Services		
Software		
Douglas	650	400
Weco	80	30
Vehicle Checkout		
Production Area	170	160
Launch Site	600	520
Launch Support		
Range*	200	200
Weco	60	50
Transportation	10	10
Propellants	30	30
NASA Administrative Charges	100	100
	$4,270	$3,870

*Estimated minimum cost. Range tracking and data acquisition dependent on spacecraft data requirements.

Since the beginning of the Delta program launch costs have increased from about $2.5 million to the $4.0 million figure, yet payload into synchronous transfer has increased from about 100 pounds to 785 pounds as shown

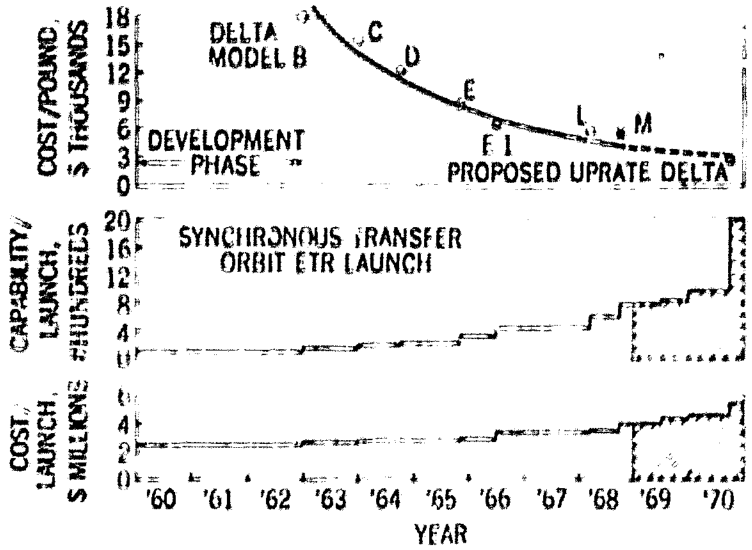

Figure 6. Delta Cost Effective History

in Figure 6. The net cost per pound in synchronous transfer orbit then has decreased from $25,000 to $5,100. This history of cost effective growth accompanied by a demonstrated flight reliability of 93 percent attests to the soundness of Delta's approach to product improvement.

The increase in cost of Delta has been principally in the first and third stage hardware. Second stage hardware costs which includes spintable, and fairing have actually decreased slightly despite inflation and upratings of the vehicle systems and performance capability. Launch services costs have escalated slowly. When the Delta launch capability at WTR was inaugurated in 1966 the prepad vehicle checkout and mission peculiar modification work was shifted from the launch site back to the production plant. This reduced factory and launch checkout costs sufficiently to nearly make-up the added costs of sustaining a launch capability at both ETR and WTR. In addition this approach permits greater flexibility and faster response to call-up missions or last minute schedule changes.

For launches conducted for outside government agencies and private industry, identifiable launch service charges are segregated and charged directly against the mission. Indirect or cost not identifiable to a peculiar mission are prorated normally over the duration of a launch services contract or a number of Delta launches and allocated accordingly.

III. The Uprated Delta Launch Vehicle

Delta is launching nearly fifty percent of NASA's unmanned spacecraft each year and is the only vehicle private industry has thus far selected as economically suitable for commercial use. The reliability and cost effective history of Delta is, in the large part, attributable to the technical approach taken at the outset of the program and still adhered to today. This approach is to use current technology and flight proven components whenever possible. The resultant vehicle is normally heavy but cheap, and has a high probability of performing repeatably and reliably from the outset. The proposed next evolutionary uprating of Delta is consistent with this past pattern of change.

Several means of uprating the capability of Delta were investigated. The primary design selection criteria was to meet the increased energy requirements envisioned by NASA at the lowest possible cost and risk without compromise of Delta's reliability record. The results of the study, which was conducted by the Delta Office of the McDonnell-Douglas Corporation under NASA funding, shows that a hydrogen/oxygen second stage replacement for the current storable propellant second stage best meets the criteria. Figure 6 shows the cost effective projection the Uprated Delta. This proposed next evolution of Delta is described together with its performance and cost.

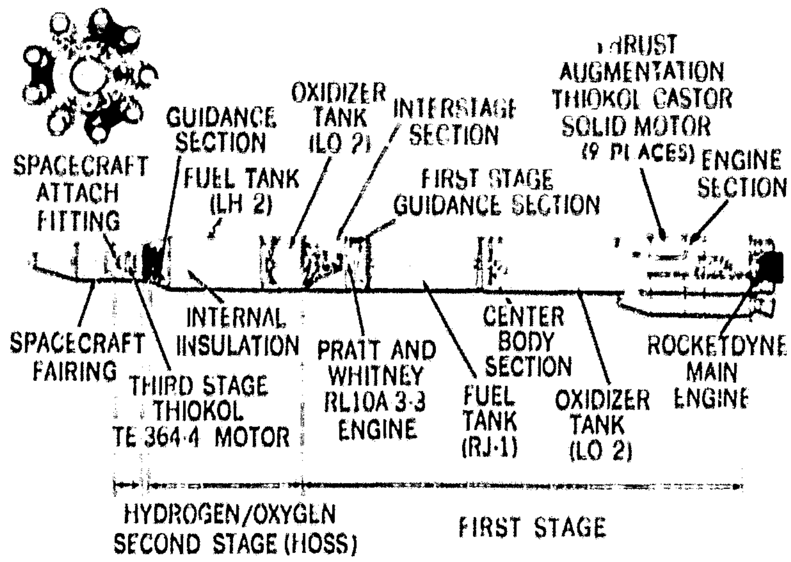

Figure 7. Uprated Delta (Proposed)

A. Vehicle Description

The three stage Uprated Delta shown in Figure 7 is composed of the present Delta first and third stages and fairing with anticipated improvements, integrated with a new liquid Hydrogen/Oxygen Second Stage (HOSS). The total vehicle stands 121 feet, weighs 274,000 lbs. at lift-off and is a constant eight feet in diameter to the top of the HOSS where it decreases to a five feet diameter to join the current Delta spacecraft fairing, spin-table and third stage motor assembly. The vehicle airframe is self-supporting to simplify ground handling and designed to withstand 95 percent ETR and WTR upper atmospheric wind profiles with spacecraft weights up to 5000 pounds as a two-stage vehicle and 3000 pounds as a three-stage vehicle.

The Long Tank Thor first stage control, electrical, and propulsion systems remain unchanged except now there is an option to use either three, six or nine Castor solid motors or alternately three Algol IIB solids plus six Castor solid motors for thrust augmentation. The Algol IIB is presently employed as the first stage for the Scout vehicle. The nine Castor solid motor booster configuration is shown in Figure 7. In this arrangement three motors are ignited at lift-off, three motors three seconds later, and finally the last set of three are ignited about 30 seconds later when the initial six start to tail-down in thrust. The multiple solid Thor is now under development for the Delta I-TOS series of missions. All nine expended motors are jettisoned at about 90 seconds after lift-off.

The material thickness of the Thor engine section, centerbody section, and propellant tanks are increased to carry the increased loads from the nine solid motors, the heavier HOSS upper stage and to accept a ten foot diameter spacecraft fairing that is to be adapted from the Titan III vehicle. In addition, the guidance section structure is redesigned to provide a constant eight foot diameter interface for the HOSS. These structural changes add 1390 pounds to the first stage.

Permanently attached to the top of the Thor guidance section is a nine foot long interstage that accommodates the extension of the HOSS engine thrust chamber below the first to second stage separation plane. This new structure, like the Thor centerbody and guidance section, is built on existing tooling. The Thor and HOSS are joined by three explosive bolt and nut assemblies developed for the Lunar Excursion Module (LEM). On firing the assemblies, the stages are released and pushed apart by six spring actuators. In three seconds the engine nozzle clears the interstage section and seven seconds later the engine is started.

The HOSS typifies the "heavy, but cheap" Delta design approach. Normally if performance is penalized by system simplicity, reliability, or cost savings that attends the adaption of flight proven hardware or techniques, the penalty is accepted. Indeed, HOSS is really less than a new stage. It is a system of the best from the old, flight proven designs, techniques, and hardware from other vehicles - newly integrated. Propellant tanks from Thor, hydrogen tank internal insulation from the Saturn IV, the Pratt and Whitney RL-10A-3-3 engine from Centaur, and the guidance, control and peripheral electronics systems from Delta are all combined to form the simple, workhorse stage shown in Figure 8.

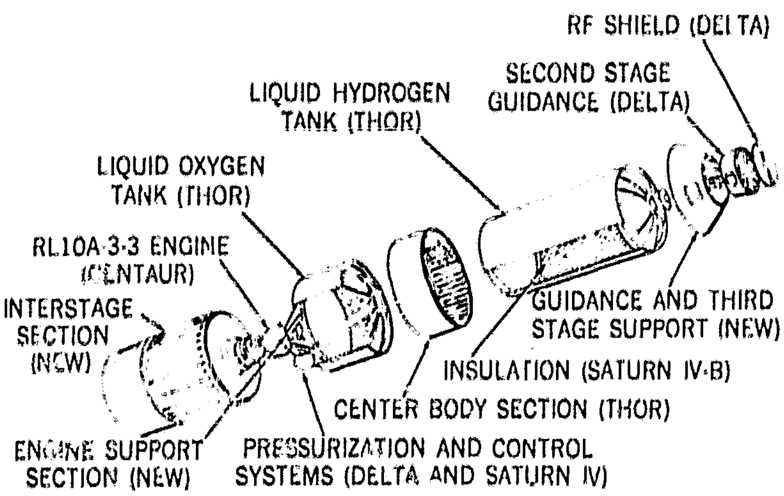

Figure 8. Uprated Delta Hydrogen Oxygen Second Stage (Hoss)

HOSS is 34 feet long, carries 20,000 pounds of propellants, with a dry weight of 3500 pounds, can be restarted in space, and is insulated for a 3200 second coast flight.

Shortened Thor propellant tanks and a Thor centerbody section form the HOSS eight foot diameter airframe. To meet the engine propellant inlet pressure requirements without use of boost pumps, as employed with Centaur, the tanks operate at a working pressure of approximately 48 psia. No testing other than normal production acceptance is necessary for the tanks and centerbody section. A common bulkhead between the propellant tanks could save about 80 pounds, but is not adapted because of cost for new tooling and higher recurring vehicle and launch operations costs. For larger fairing envelope requirements, the rigidity of the airframe permits adaption of the ten foot diameter Titan III fairing without second stage structural beef-up.

The tapered guidance and third stage support structure forward of the liquid hydrogen tank is designed to permit the current Delta second stage guidance section (Figure 1), containing all guidance, control, power, range safety and RF systems, to slip down inside it. This approach retains the identical spintable and fairing structural interfaces, but penalizes stage weight about 60 pounds. This new structure is to be statically load tested.

The fiber reinforced polyurethene insulation for the Saturn IV hydrogen tank was developed by Douglas Aircraft Company using a Thor propellant tank. Manufacturing and processing procedures are available so that no testing is necessary other than thermal surveys during tanking and static test firings of the total propulsion system. Internal insulation is heavier and less efficient than the ejectable external insulation panels used on Centaur, but cost, simplicity and flight reliability dictates its use.

The turbopump fed RL-10A-3-3 engine develops 15,000 pounds of thrust and is flight proven on Centaur and Saturn IV. Testing is limited to survey firings over the spectrum of engine inlet start conditions peculiar to HOSS. At some loss in performance, HOSS uses an open loop propellant utilization system. This permits the RL-10 mixture ratio control valve to be locked and the associated vehicle propellant management electrical systems eliminated. The chill-down of the propellant pumps preparatory to engine start is to be accomplished in-flight during the terminal portion of the first stage boost and during the short coast period after staging. Again this approach, used on Saturn IV, penalizes performance but dramatically reduces system complexity, and launch and vehicle costs by eliminating the special GSE, procedures, and thermal problems experienced by Centaur with a liquid helium pre-chill before launch. From Saturn IV and Centaur there are approximately 15 to 20 surplus RL-10 engines that either exist or can be built-up out of parts. These engines will be used for non-critical performance missions, as they deliver 432 seconds of specific impulse compared to 444 seconds from the current RL-10A-3-3 model.

A growth configuration of the RL-10 engine can be introduced as cost effectively as the addition of the multi-solids to the booster. This uprated engine has a thrust of 20,000 pounds and a specific impulse of 453 seconds. The ten second increase in specific impulse is achieved by adding a thirty inch radiation cooled nozzle extension.

The uprated RL-10 engine (20,000 pounds thrust with nozzle extension) increase the useful load into synchronous orbit about 400 pounds while the adaption of the ten foot diameter Titan fairing reduces the useful load about 300 pounds.

Pitch and yaw control during powered flight is effected by gimballing the engine and roll is controlled by cold gas jets. At a weight penalty of about 13 pounds, the Saturn IV gimbal actuation system is adapted directly on HOSS. Only systems integration and functional tests during vibration and static load testing of the engine thrust structure are necessary to qualify it.

Propellant tank pressurization is provided by ambient helium stored in Delta high pressure titanium spheres. Two (non-restart) or four (restart) spheres are employed. During main engine operation, cold gaseous hydrogen is bled from the engine, as on Saturn IV, to pressurize the fuel tank.

Valves for propellant fill, drain, and venting are almost exclusively Saturn IV parts as are the pressurization regulation and propellant level sensing components. Again these components if specifically designed to HOSS requirements would save weight, but cost and reliability override this consideration. Guidance, control, power, range safety and RF systems are carried over directly from Delta with an identical layout and mounting arrangement. Other than repositioning umbilicals and antennas on the guidance and third stage support structure skin, the only significant electrical system change is to double the time of the programmer to 8192 seconds and increase the capacity from six variable sequence outputs to twelve variable and eight fixed sequence outputs. This expanded capability is required to handle long coasts, engine chill-down and restart functions. The radio guidance and integrating accelerometer cut-off system operate the same as on Delta. A growth version of HOSS includes a strap-down inertial guidance system to replace the autopilot and radio guidance on both first and second stages. Accuracy is enhanced and trajectory shaping constraints, imposed by radio guidance are eliminated.

The attitude control subsystem is patterned after the current Delta. Control thrust is supplied by ambient nitrogen gas stored in eight (8) high pressure titanium Delta spheres. The Delta pitch, roll, and yaw jets are mounted on the inside periphery of the aft skirt. Again cold gas control is heavy, but simple, reliable, cheap and does not require development or qualification testing. A monopropellant control system similar to that used on Centaur could save about 15 pounds.

The third stage spintable, spacecraft attach fitting and fairing are identical to that being flown today. A program for Delta and other users is currently underway to increase the propellant weight of the TE-364-3 from 1440 to 2336 pounds (designated TE-364-4) by adding a cylindrical section between the two hemispherical halves of the case. The increased length of the motor encroaches nine inches forward into the fairing envelop shown in Figure 3.

The major development testing of Uprated Delta centers on defining the aerodynamics and modal characteristics of the vehicle through wind tunnel and Bungee tests, and conducting second stage propellant loading and ground static firings. These latter tests are to demonstrate satisfactory operation of the second stage propulsion system, verify proper functioning of the GSE and associated checkout and operational procedures. Facilities to static test HOSS are available together with surplus engines and propellant loading GSE from Saturn IV. Static test firings are to be conducted on a production article that will be refurbished and flown.

The time from authority to proceed to the first Uprated Delta launch is slightly over two years. The first launch would be from ETR; in keeping with past evolutionary changes to Delta the first launch is to carry a flight spacecraft.

B. Performance

The Uprated Delta velocity capability from ETR is shown in Figure 9. The performance for three vehicle configurations is shown. The configuration labeled 'Basic' (See Table V) is composed of the Long Tank Thor booster, thrust augmented with nine Castor solids, the HOSS, the TE-364-4 third stage, and the current Delta 65 inch diameter spacecraft fairing. The performance degradation that results from adapting the 120 inch diameter Titan fairing to the 'Basic' configuration is illustrated by the performance of configuration 'E'. Configuration H illustrates the enhanced performance from substituting three Algol IIB solids for three of the nine Castors in configuration 'E' and introducing the uprated thrust RL-10 engine with a radiation cooled nozzle extension.

Table V

Summary of Uprated Delta Performance Capability for a Synchronous Mission

Configuration Designation		Long Tank Thor Thrust Augmentation	HOSS + TE-364-4 HOSS Engine Thrust lbs.	HOSS + TE-364-4 HOSS Engine Specific Impulse seconds	Fairing Diameter inches	Useful Load into a Synchronous Transfer Orbit (100 × 19,400 n.m.) lbs.
BASIC		3 Castors I + 6 Castor II	15,000	443	65	2395
	A	3 Castors I				1700
	B	3 Castors I + 3 Castor II				2145
	C	3 Algol IIB				2380
OPTIONS	D	3 Algol IIB + 6 Castor II				2690
	E	3 Costor I + 6 Castor II			120	2105
	F	3 Algol IIB + Castor II				2400
	G	3 Castor I + 6 Castor II	20,300	453		2505
	H	3 Algol IIB + 6 Castor II	20,300	453		2857

The specific synchronous transfer performance capabilities for the Basic and other configurations of the Uprated Delta are summarized in Table V. These performance capabilities are the three sigma minimum and include contingency by way of second stage velocity reserve to cover weight, propulsion and aerodynamic design uncertainties and vehicle-to-vehicle production tolerances. The trajectory flight mode for a synchronous transfer mission consists of injecting HOSS into a 100 nautical mile (n.m.) circular parking orbit, coasting for 600 seconds to near the equator where HOSS is restarted ubsequent to the HOSS shutdown the third stage is separated and ignited such that the spacecraft is injected into the synchronous transfer orbit with a 100 n.m. perigee over the equator. Since the HOSS orbits, range safety constraints do not degrade performance.

To illustrate the performance gap left by the phase out of the Atlas/Agena for NASA use the performance of the current Delta and Atlas/Centaur is included in Figure 9. The Uprated Delta breaches not only the gap in performance but also the gap in launch costs between the $4.0 million dollar Delta and the 16 million dollar Atlas/Centaur.

A logical extension of the HOSS stage is its integration into the Titan III family of boosters. The approximate performance of the HOSS with a TE-364-4 third stage on the Titan III core vehicle (Titan IIIB) and the Titan III core with the 120 inch five segment solids (Titan IIID) is shown in Figure 9.

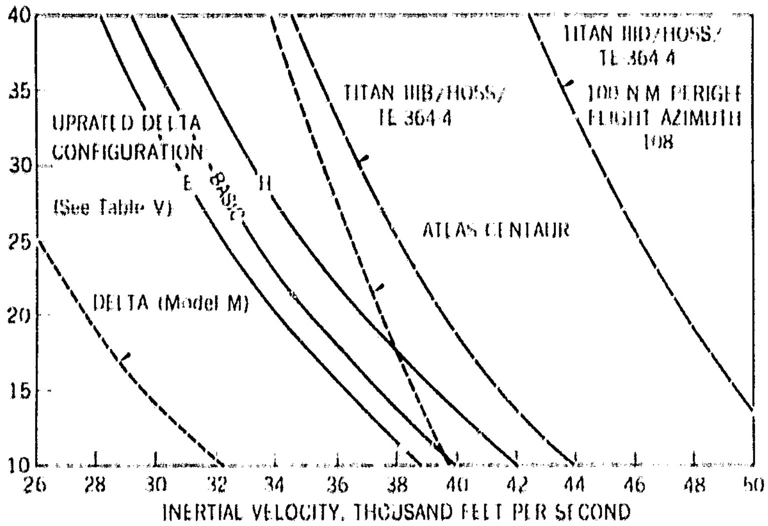

Figure 9. Characteristic Velocity for Uprated Delta

C. Cost

The Uprated Delta can be developed and readied for launch at both ETR and WTR for 31 million dollars as shown in Table VI. This non-recurring cost includes the vehicle hardware for the initial demonstration flight. It is assumed that the multi-solid booster and elongated TE-364-4 motor that are currently under development are incorporated into the Delta stable by the time Uprated Delta is readied for flight. The optional 20,000 pound thrust RL-10 engine with a thirty inch radiation cooled nozzle extension and the

Table VI
Uprated Delta Non-Recurring Costs

UPRATED DELTA		COST (Million Dollars)
Contractor Furnished Services		26.5
Vehicle		25.5
Engineering, Design, Analysis	7.9	
Testing	2.8	
AGE (Factory, ETR & WTR)	7.9	
Manufacturing Development Support	1.3	
Tooling, Design & Fabrication	2.2	
First Flight Vehicle (Second Stage, Interstage & Fairing)	1.8	
Program Plans and Management	1.6	
HOSS Engine		1.0
Engineering, Design, Analysis	0.2	
Survey Firings	0.8	
Government Furnished Equipment		2.5
First Flight Vehicle Booster	1.3	
Propellants	.08	
Surplus Saturn RL-10 Engine	.04	
Facilities*		2.0
ETR	.9	
WTR	1.1	
	TOTAL	31.0

*In Addition to Long Tank Thor Facility Modifications

Table VII
Uprated Delta Launch Costs
(Thousand Dollars - 1971)

Hardware	
First stage	$1,300
Second Stage and Fairing	1,800
Third Stage	100
Attach Fitting	30
Launch Services	
Software	
Douglas	750
Weco	100
Vehicle checkout	
Production Area	200
Launch Sit	750
Launch Support	
Range*	200
Weco	60
Transportation	10
Propellants	50
NASA Administrative Charges	150
Total	$5,500

*Estimated minimum cost. Range tracking and data acquisition dependent on spacecraft data requirements.

Table VIII
Delta and Uprated Delta Second Stage Hardware
Cost Comparison

	Delta	Uprated Delta
	(Thousand Dollars)	
STRUCTURE		
Fairing	$ 80	$ 76
Fairing support section	0	16
Spin table	70	70
Guidance compartment	22	0
Interstage	23	41
PROPULSION		
Tanks and engine support	71	135
Engine	89	390
Valves, vents, and umbilicals	18	109
Insulation	[illegible]	125
ELECTRONICS		
Programmer (6 vs 21 seq.)	40	49
Velocity Cutoff system	29	29
Gyro package	42	42
Sequence distribution box	17	19
Power distribution box	12	12
Weco package	44	44
C band beacon	11	11
Range safety system	14	17
Instrumentation	70	70
Electronics package	22	30
Static inverter	18	18
Umbilicals	4	4
GIMBAL ACTUATOR SYSTEM	21	41
ELECTRONICS INTEGRATION	35	40
FINAL ASSEMBLY	71	110
ENGINEERING AND MANAGEMENT	228	282
Total	$1,051	$1,780

adaption of the ten foot diameter Tital fairing would increase the non-recurring cost approximately nine million dollars.

The recurring launch cost of Uprated Delta is projected to be $5.5 million for the three solid motor thrust augmentation configuration. Additional sets of three solid motors incrementally increase the first stage hardware costs by approximately $200,000. The breakdown between hardware, launch services, and NASA administrative charges is provided in Table VII. The major increase over the current Delta costs is in the HOSS hardware. The engine, insulation, and the propellant loading valves and umbilicals are the principal cause of cost escalation as shown by the cost comparison of Delta and Uprated Delta second stage hardware in Table VIII. The first stage hardware and the RL-10A-3-3 engine costs in the 1971 time period have the greatest degree of uncertainty, possibly as much as ten percent on the first stage and fifty percent on the RL-10. The other HOSS hardware costs are based on experience from Delta, Thor, or Saturn IV and are quite reliable. Launch services costs are also based on Delta experience but increased to allow for the additional mission analysis and checkout attendant a cryogenic stage.

In summary, the Uprated Delta exploits NASA's considerable investment in the Centaur and Saturn IV liquid hydrogen/oxygen technology and hardware. It affords a significant increase in Delta performance at relatively low development cost and risk. It permits Delta to continue to meet the bulk of government and industry needs by maintaining availability of a reliable, economical launch vehicle to carry the scientific and operational satellites presently too small for Atlas/Centaur, but fast outgrowing current Delta. The use of Atlas/Centaur or Titan III-B/Agena would substantially increase user costs above the projected $5.5 million launch cost of Uprated Delta. The recurring savings from Uprated Delta rapidly recovers the estimated $31 million cost for development and deployment. The proposed Uprated Delta is currently before NASA for consideration in its launch vehicle development programs. It is not at this time an officially approved program.