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# THE DELTA LAUNCH VEHICLE MODEL 2914 SERIES



CHARLES R. GUNN

SEPTEMBER 1973



GSFC

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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ABSTRACT

The newest Delta launch vehicle configuration, Model 2914 is described for potential users together with recent flight results. A functional description of the vehicle, its performance, flight profile, flight environment, injection accuracy, spacecraft integration requirements, user organizational interfaces, launch operations, costs and reimbursable users payment plan are provided.

Delta is a medium class launch vehicle that carries the majority of NASA's unmanned spacecraft, and is used by private industry and foreign governments to launch their scientific and applications satellites. The versatile, relatively low cost Delta has a flight demonstrated reliability record of 92 percent that has been established in 96 launches over twelve years while concurrently undergoing ten major upratings to keep pace with the ever increasing performance and reliability requirements of its users. At least 40 more launches are scheduled over the next three years from the Eastern and Western Test Ranges.

The first stage of the three stage Delta Model 2914 is the Extended Long Tank Universal Boattail Thor. The liquid propellent capacity is increased and the high performance H-1 main engine developed for the Saturn 1B vehicle is adapted to the stage which can be thrust augmented with up to nine strap-on Castor II (TX-M-354-5) solid propellent motors, depending on mission performance requirements. The Delta second stage, recently uprated with a strap-down inertial guidance system and reconfigured to accept a new 2.44 meter diameter metal spacecraft fairing, is now being modified to adapt the liquid propellent Descent Engine from the Apollo Lunar Excursion Module (LEM). In conjunction with the introduction of the LEM engine which replaces the Titan Transtage Engine now in use, several reliability improvements in the second stage hydraulic and pneumatic systems have been incorporated. The Delta third stage is the spin stabilized TE-M-364-4 solid propellent motor.

The Delta Model 2914 is to be available in early 1974, cost about $9 million and capable of injecting 2040 kg into low earth orbit, 705 kg into geosynchronous transfer or escape about 455 kg of payload.

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# THE DELTA LAUNCH VEHICLE
# MODEL 2914 SERIES

## INTRODUCTION

The Delta launch vehicle is a versatile, relatively low cost space transportation system that is extensively used by NASA, private industry and foreign governments to launch a broad spectrum of unmanned scientific and applications satellites. Over the last thirteen years in 96 launches, Delta has established a flight demonstrated reliability record of 91 percent while concurrently undergoing twelve major upratings to keep pace with the ever increasing requirements of its users. Delta offers mission planners a wide choice in performance capability together with unprecedented mission flexibility and a quick response capability to support call-ups for spacecraft replacements or follow-on missions.

The vehicle is designed to permit the performance capability to be configured to the specific requirements of the mission by use of either two or three stages and by adding from three to nine strap-on thrust augmentation solid propellent motors to the first stage. Mission peculiar trajectory and special spacecraft sequencing requirements are readily programmed by software changes in the Delta inertial guidance system computer which provide users broad flexibility and accommodates late changes in mission requirements. To support between 15 to 20 scheduled launches per year over the next three years, a production base and hardware inventory is established and insures a capability to support call-up missions.

Over the past thirteen years, Delta has been improved and uprated in reliability and performance through an evolutionary process of incrementally adopting available technology and flight proven components from other space programs. This approach has allowed vehicle changes without an interruption in the launch program and at the same time minimized the cost of improvements.

Since many of the newly adopted components are not an optimum design for the particular application selected, the Delta vehicle is generally heavier than need be, but is relatively inexpensive and offers a high probability of performing repeatedly and reliably from the outset of each new change. There have been twelve major incremental improvements made to Delta without a preoperational or development flight test launch; and with the single exception of the first Delta launch in 1960, there has never been a failure of the first flight of an improved vehicle.

The first of the new Delta, Model 2914, series of vehicles is now scheduled to be launched late this year and is described here together with its capabilities, constraints and costs.

## DELTA

### The Evolution of Delta

The evolution of the Delta launch vehicle, shown in Figure 1, reaches back eighteen years when, in 1955, the United States participated in the International Geophysical Year 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 propellent 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, now McDonnell-Douglas Astronautics Corporation (MDAC). The initial objective of the Delta program was to provide an interim space launch

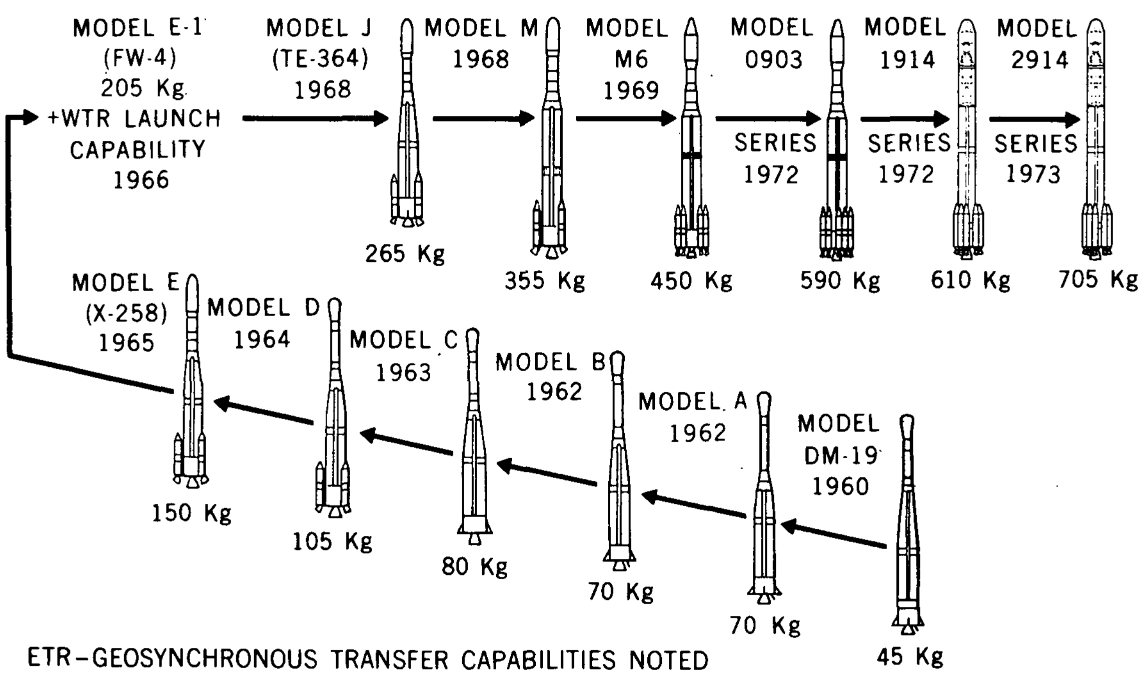



Figure 1. Delta Evolution

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 covered 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), the cooperative NASA/United Kingdom Ariel I (April 1960), the TIROS II through VI series, the first Orbiting Solar Observatory, and the first private industry satellite, American Telephone and Telegraph 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 new missions planned for Delta outstripped the interim buy of twelve vehicles, so an order was placed for fourteen additional 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 comminications satellite, Relay I (December 1962), 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 commerical communications satellite, The International Communications Consortium'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 1.52 meter diameter Nimbus spacecraft fairing developed for the USAF Agena stage. Improved Delta also adapted the USAF developed United Technology Corporation's FW-4 solid propellant motor to replace the X-258 third stage motor (Model E-1). The first Improved Delta was launched November 1965 and among the missions carried on this model of Delta are the near polar Geophysical Orbiting Satellites, GEOS-A and B; the heliocentric Pioneer series A through D; the low earth orbiting Biological Satellite, BIOS A through C; the synchronous communications satellites, Intelsat F1 through

F4; the lunar orbiting Anchored Interplanetary Monitoring Probe, A-IMP A and B; the sun-synchronous ESSA 2 through 6; the High Eccentric Orbiting Satellites, HEOS developed by the European Space Research Organization and the Canadian International Satellite for Ionosheric Studies, ISIS.

In 1966 Delta undertook to adapt the Surveyor spacecraft solid propellant retromotor as a new third stage. The spherical case was modified to mate to a spintable assembly and the motor, designated TE-364-3, was requalified for the Delta spinning environment. The first Delta using this third stage motor, Delta Model J, was launched in July 1968 and carried the Radio Explorer, RAE-A spacecraft.

At about the same time Delta initiated the use of the TE-364-3 motor, the USAF undertook the uprating of the Thor booster by lengthening the liquid oxygen and RP-1 fuel tanks and converting the fuel tank to a constant 2.44 meter diameter. This Long Tank Thor carries about 47 percent more propellants than previous models. In September 1968, Delta launched its first Long Tank Thor with the Improved Delta second stage and TE-364-3 third stage. The Delta Model M carried, among others, the Intelsat II, British Skynet and NATO communications satellite series.

In early 1968, Delta started a redesign and retrofit of the Long Tank Thor engine section to permit the addition of a second set of three thrust augmentation solid motors. The first Delta Model M6 with six solid motors was launched from the Western Test Range in January 1970 and carried the NASA TIROS Operational Satellite, TOS-M into a 1480 km circular sun-synchronous orbit. The two remaining Delta Model M6 vehicles have been used to carry the Interplanetary Monitoring Probe I and the ITOS-A spacecraft.

In 1969, Delta was again uprated in performance capability and also in guidance accuracy. The Delta Model 0903 series introduced the Universal Boattail Thor (UBT), with a new engine section on the first stage that is designed to accommodate attachment of up to nine (9) thrust augmentation solid motors; an uprated second stage propulsion system that incorporates the Titan Transtage engine (AJ10-118F), operating on $N_2O_4$ and Aerozene 50 propellants; and a strapdown inertial guidance system that replaces both the first and second stage autopilot systems and the Western Electric Co. radio guidance system. The first flight of the Model 0903 series successfully launched the Earth Resources Satellite, ERTS-A in July 1972. This series carries also the Nimbus-E and the Improved TIROS Operational Systems series D through F.

Early in 1971, the development of a new spacecraft fairing was initiated together with the incorporation of a higher performance engine into the Delta booster. These changes were phased into Delta in three steps. First, a 2.44 meter diameter spacecraft fairing was integrated with the Model 0913 vehicle by adding a new, constant 2.44 meter diameter interstage and suspending the current 1.52 meter diameter second stage within the interstage by an adapter that also interfaces with the new fairing. Second, to help offset the reduction in controlability due to the 2.44 meter fairing, the UBT booster propellant tanks were extended 3.05 meter thereby increasing the propellant load 13,610 kilograms and reducing the maximum dynamic pressure during transonic flight. Third, the TE-364-3 third stage solid propellant motor was elongated to increase the propellant weight to 1045 kilograms from 653 kilograms. This larger motor is redesignated the TE-364-4. This new launch vehicle configuration, designated the Delta Model 1914 series and known as the "Straight 8" because of its streamlined appearance, was first launched in November 1972 and carried the Canadian Telesat Satellite, ANIK-A. The model 1914 series is assigned to carry the Synchronous Meteorological Satellite (SMS) A, the Radio Astronomy Explorer (RAE) B and the Interplanetary Monitoring Probe IMP-H.

The most recent evolutionary uprating of the Delta launch vehicle ushers in the new Model 2914 series. This configuration is shown in Figure 2. The Model

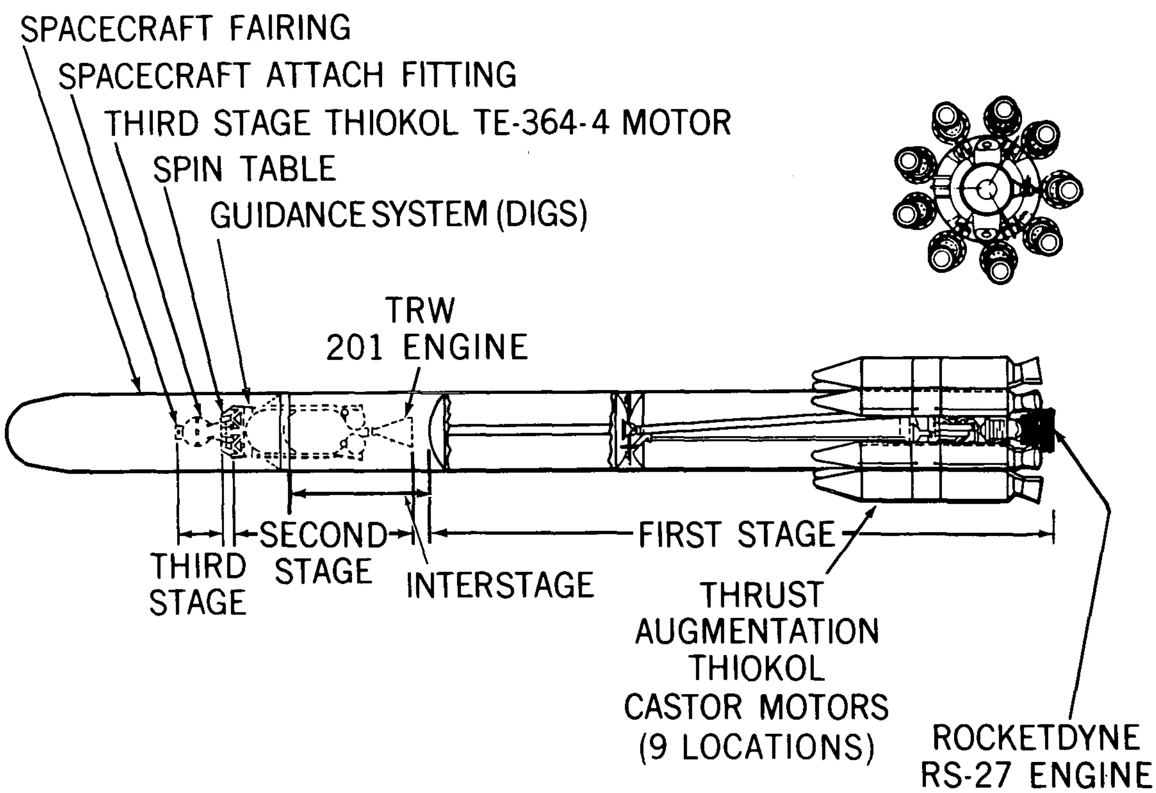



Figure 2. Delta Model 2914

2914 booster, second stage, and third stage are the same as the "Straight 8", except the booster now incorporates a more powerful engine adapted from the man-rated Saturn 1B stage.

The first Delta "Straight 8" Model 2914 is scheduled to launch the British Skynet-IIA satellite late this year. Other missions scheduled to be launched by the 2914 series of vehicles include the following:

Meteorological Satellites

Improved TIROS Operations System (ITOS) series
Nimbus series
Synchronous Meteorological Satellite (SMS) series
ESRO Meteosat
Japanese Geosynchronous Meteorological Satellite (GMS)
TIROS-N series

Applications Satellites

Canadian/U.S. Communication Technology Satellite (CTS)
Italian SIRIO
ESRO Orbital Test Satellite (OTS)
Earth Resources Technology Satellite (ERTS)

U.S. Domestic Communications Satellites

Western Union (WESTAR) series
American Satellite Corporation (ASC) series
CML Satellite Corporation (CML) series
COMSAT Maritime Satellite (MARISAT) series

Foriegn Communication Satellities

Canadian TELESAT series
French/German SYMPHONIE series
NATO III series
Japanese Communication Satellite (JCS)

Scientific Satellites

Aeronomy Explorer (AE) series
Interplanetary Monitoring Platform (IMP) series
ESRO COS-B
LAGEOS
GEOS-C (NASA)

ESRO Geostationary Scientific Satellite (GEOS)
International Ultraviolet Explorer (IUE)
ESRO/NASA Interplanetary Magnetospheric Explorer (IME)
Orbiting-Solar Observatory (OSO)

Early this year another step in the evolutionary development of Delta was started with the initiation of a redesign of the first stage to accommodate up to nine larger thrust augmentation solid motors. The new motors, Castor IV's, provide about two and one half more total impluse than the Castor II motors currently in use. This development marks the first U.S. industry commercially funded improvement of a launch vehicle. The new vehicle, designated the Delta Model 3914 is capable of injecting up to 910 kg into a geosynchronous transfer orbit and is to be launched in late 1975. The use of the Delta Model 3914 is currently restricted to U.S. domestic communications satellite users.

## Vehicle Description

The three stage Delta vehicle, Model 2914, shown in Figure 2 stands 354 meters and weighs 132,000 kilograms at lift-off. The vehicle is designed for ascent through 95% ETR and WTR upper atmosphere annual wind profiles, lift-off in 74 kilometers per hour ground winds, and hold on the launch pad for several hours in readiness for a launch window only seconds wide.

The first stage is composed of a liquid propellant core that is thrust augmented by solid propellent motors. The 2.44 meter diameter core is the UBT booster now elongated to 21.3 meters from 18.3 meters in order to increase the liquid propellant load to 80,300 kilograms from 66,700 kilograms of RP-1 and liquid oxygen. The core is powered by the Rockwell International Rocketdyne RS-27 engine that is derived from the Saturn IB H-1 engine and is now adapted to the Thor. The turbopump fed RS-27 develops 912,000 newtons thrust at lift-off compared to the 756,000 newtons thrust of the MB-3 Thor engine used previously. The core burns to propellant depletion about 224 seconds after lift-off (T + 224) at an altitude of about 95 kilometers. Thrust augmentation solid propellant motors attach at the base of the core on the UBT engine section structure. The UBT is structurally designed and thermally insulated to carry up to nine (9) Thiokol Castor II solid motors (TX-354-5).

Normally, the thrust augmentation motors are build-up in sets of three. Up to six motors can be ignited on the pad and the remainder no sooner than 38 seconds after lift-off in order to hold the vehicle acceleration induced loads on the propellant tank bottoms within allowable limits. The Castor II motors each develop 147,000 newtons thrust at ignition, burn for 40 seconds and are jettisoned from the core at about T + 90. This time is dictated by considerations of combined dynamic pressure angle-of-attack loadings on the jettison mechanism and a Range Safety requirement for an offshore impact of the expended motors. Jettison is effected by firing an explosive bolt holding a clamped ball-socket joint. Acceleration of the core plus aerodynamic drag on the motors eject the empty cases away from the vehicle as shown in Figure 3.

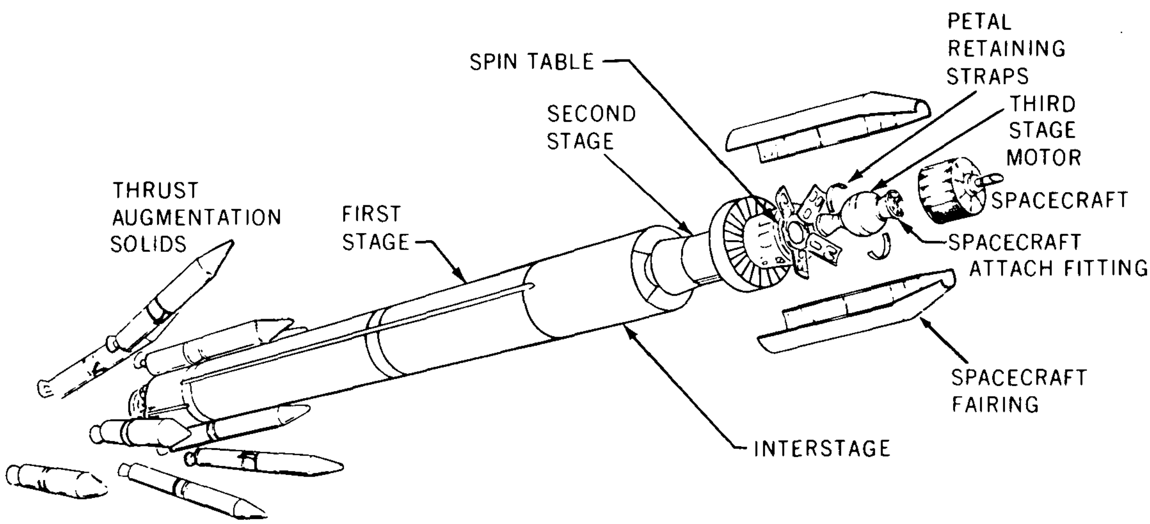



Figure 3. Delta Staging Schematic

During powered flight pitch and yaw steering is exerted by gimballing the core main engine. Roll control is effected by differentially gimballing a pair of small outboard vernier engines. Subsequent to main engine shut-down the verniers continue to operate for about 6 seconds, damping shut-down transients and stabilizing the vehicle for staging of the second stage.

The guidance and control for the booster originates from the second stage. A strap-down inertial guidance system provides guidance and control for the total vehicle from lift-off through attitude orientation and ignition of the spin stabilized third stage solid propellant motor. This strap-down system is composed of a digital computer and an inertial measurement unit (IMU). The 4096 word memory computer performs the navigation, guidance, steering, controls systems stability and shaping, and initiates discretes for both first and second stages. It directs the vehicle through a pre-programmed trajectory navigating on IMU velocity data to determine present position and velocity, which it then predicts forward along a nominal trajectory to determine the final position and velocity at injection. The predicted final terminal state is compared to the desired terminal state to derive the required vehicle steering commands and engine shutdown time to reach the desired terminal injection state. All guidance functions are programmed into the vehicle computer with launch pad computer software rather than hardware adjustments. This permits maximum mission flexibility for the user.

The 2.44 meter diameter interstage section between the first and second stages is provided with a spring separation assembly. Eight seconds after the first stage main engine shutdown, studs with explosive nuts that attach the two stages are fired and the second stage is spring separated from the first stage. Five seconds later the second stage engine is started.

The Delta second stage is 5.2 meters long and approximately 1.52 meters in diameter, except at the 2.44 meter diameter adapter ring that interfaces with interstage and fairing and carries the second stage umbilicals and antennas. At ignition the second stage weighs 5,440 kilograms. The TRW Inc. engine originally developed for the Apollo Lunar Excursion Module vehicle and now adapted by Delta, is a pressure fed, ablative and radiation cooled engine that develops 45,800 newtons thrust and operates for about 345 seconds on Aerozine-50 and $N_2O_4$ storable propellants. The propellant tanks are cylindrical with a hemispherical internal common bulkhead between the fuel and oxidizer tank. The system is pressurized from lift-off to strengthen the structure and suppress oxidizer boiling. The engine, TR 201, is capable of multiple restarts.

During the second stage powered flight, pitch and yaw steering is provided by gimballing the engine and roll is controlled by cold nitrogen gas jets. Cold nitrogen gas jets control the vehicle in all axes during coast and provides propellant settling ullage thrust for restarting the engine. Should there be a failure in a cold gas attitude control valve, this will be sensed by DIGS and a completely redundant set of valves will be switched on-line and the set with the failure deactivated. The control system electrical power and nitrogen gas supply is capable of maintaining second stage attitude for a little over one hour. 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/PCM/FM/FM 45 x 20 "S" band telemetry system, dual command destruct receivers and associated power supplies.

On several Delta missions where the second stage was orbited and the vehicle performance exceeded the requirements of the primary mission, the second stage has been used as a platform for placing secondary satellites into orbit. Table 1 summarizes the secondary satellites that have been carried on the

Table 1

Delta Secondary Experiments/Satellites

| Secondary Experiment/ Satellite | Primary Mission | Date | Experiment/ Satellite Wt. (lbs) | Orbit |
|---|---|---|---|---|
| PAST MISSIONS | | | | |
| TETR-A | PIONEER-C | 12/67 | 55 | 160 x 260 n. mi. x 28.5° |
| TETR-B | PIONEER-D | 11/68 | 55 | 240 x 500 n. mi. x 28.5° |
| PAC | OSO-G | 8/69 | 265 | 300 n. mi. CIRC x 33° |
| OSCAR 5 | TIROS M | 1/70 | 40 | 790 n. mi. CIRC x 101.6° |
| CEP | ITOS-A | 12/70 | 11 | 790 n. mi. CIRC x 101.6° |
| TETR-D | OSO-H | 1971 | 66 | 300 n. mi. CIRC x 33° |
| OSCAR 6 | ITOS-D | 10/72 | 40 | 790 n. mi. CIRC x 101.6° |
| MISSIONS UNDER CONSIDERATION | | | | |
| OSCAR AOB | ITOS-G | 1974 | 50 | 820 n. mi. CIRC x 101.6° |
| INTASAT | ITOS-G | 1974 | 50 | 820 n. mi. CIRC x 101.6° |

Delta second stage and ejected into orbit after either the primary spacecraft or the third stage with the primary spacecraft had been separated from the second stage. Included also in Table 1 are the secondary experiments and satellites currently under active consideration for piggyback flights on Delta. With the introduction of the new 2.44 meter diameter fairing and attendant structural modifications to the second stage to interface with the fairing and interstage, a substantially greater volume is now available to accommodate secondary spacecraft or experiments as shown in Figure 4.

Secondary experiments or satellites can either remain on-board the second stage or be ejected. Support and separation systems have been qualified and flight proven for ejecting satellites. For experiments that remain on-board, an orbiting Delta second stage secondary experiment has demonstrated the feasibility of providing on-board experiments with power, data and command RF links, passive thermal control and earth-oriented attitude pointing for long duration.

The third stage assembly consists of a spin table, the Thiokol TE-364-4 solid propellant motor, spacecraft attach fitting, spacecraft and the spacecraft fairing. The spin table shown in Figure 5 consists of a bearing support structure and a

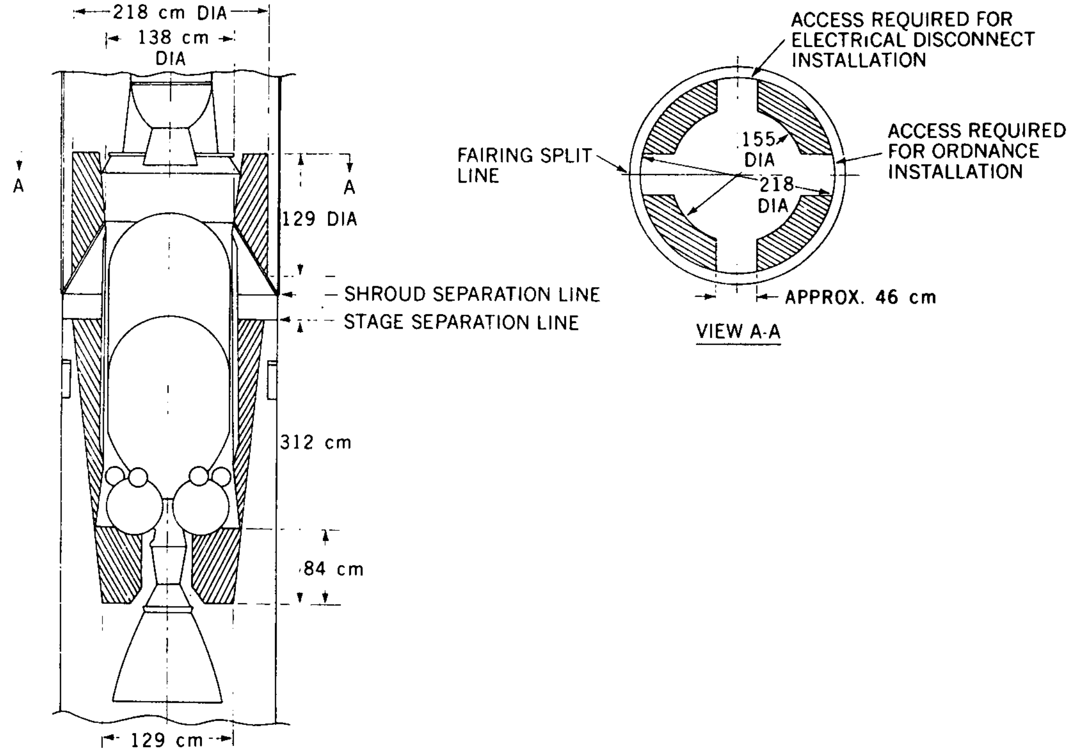



Figure 4. Potential Secondary Experiment Envelope

conical third stage motor pedestal truss that is divided into four petals, hinged at the base and clamped to the equator of the third stage motor by a retaining strap. The retaining strap is held in tension by two explosive bolts that are fired two seconds after the motor and spacecraft are spun up and the 40 second time delay squib that ignites the TE-364-4 motor is started. The released petals fly outward under centrifugal force, releasing the third stage from the spin table (Figure 3). At the same instant the second stage is backed away from the free spinning third stage by venting residual pressurance (helium) overboard through two retrojets. Approximately 38 seconds later the third stage motor is ignited by the time delay squib. The TE-364-4 motor is essentially identical to the -3 model except that a 14 inch cylindrical section is added between the two hemispherical halves of the case. The propellant weight is increased to 1045 kilograms from 653 kilograms, it burns for 44 seconds and develops an average thrust of 66,700 newtons.

Torque to the spin table is imparted by combinations of small solid propellant rocket motors, which provide spin rates from 30 to 150 revolutions per minute (±10 percent) for spacecraft roll moments of inertia ranging from 30 to 230 kilograms meters squared. A lower limit of approximately 30 revolutions per

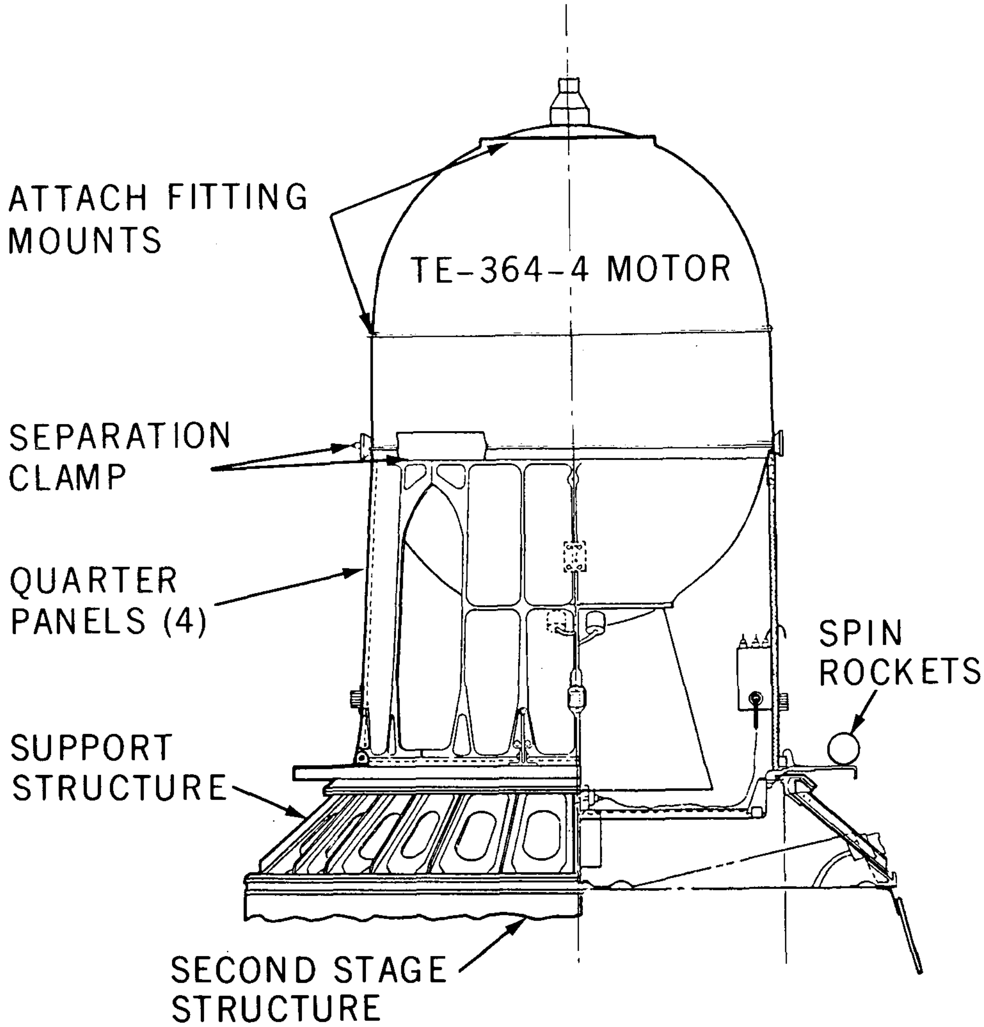
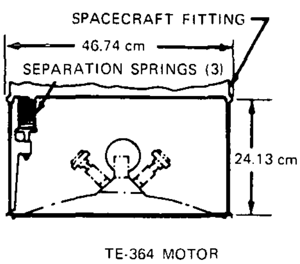



Figure 5. TE-364-4 Third Stage and Spin Table

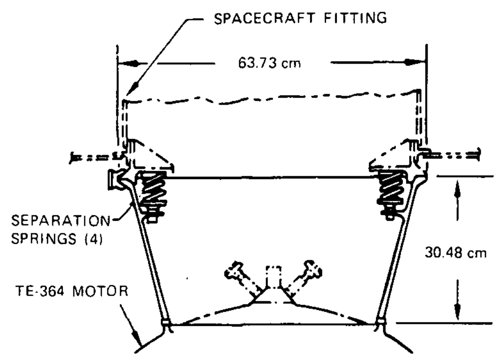
minute (rpm) is dictated by minimum dynamic stability of the third stage/spacecraft during third stage motor burning. If less than 30 rpm is desired the effect upon orbit injection errors must be carefully assessed. The anticipated maximum spin rate users would desire was 150 rpm, consequently the third stage motor is qualified only up to this spin rate.

For those spacecraft that require spin stabilization but the mission does not require use of a third stage, a spacecraft can be spun either by use of the spin table or by placing the combined second stage/spacecraft in a controlled spin with the second stage roll attitude control jets. This technique, which has been used on a number of missions, eliminates the spin table for spacecraft requiring spin rates up to 20 rpm.

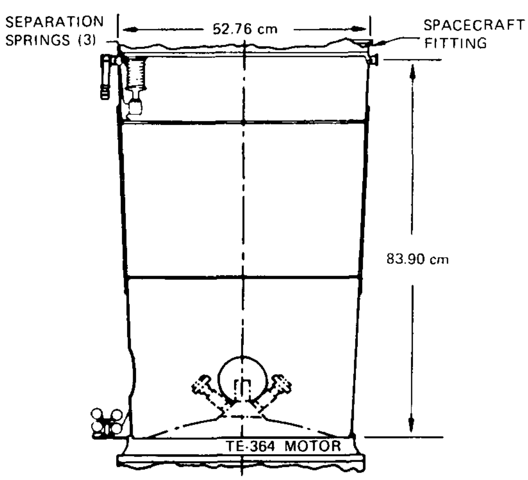
The spacecraft is clamped to the attach fitting by a circular retaining strap assembly that releases by firing two explosive bolt cutters subsequent to third stage motor burn-out. Separation from the expended third stage is then effected by a separation spring, or springs, which provides the spacecraft with a relative separation velocity of 2.0 to 2.5 meters per second 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 shown in Figure 6. 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



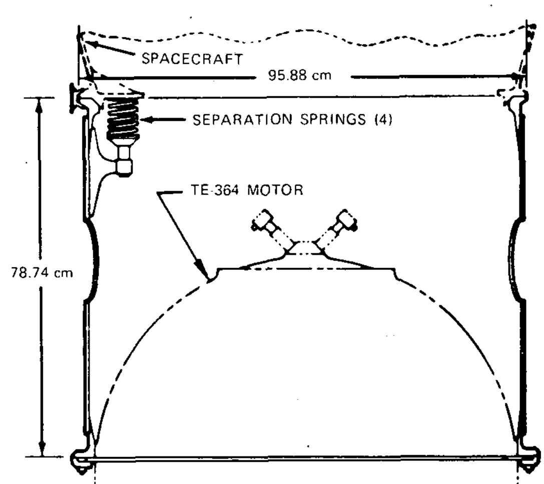

**1809 ATTACH FITTING**
**WT: 11 Kg**




**2512 ATTACH FITTING**
**WT: 14 Kg**




**2030 ATTACH FITTING**
**WT: 15 Kg**




**3731A ATTACH FITTING**
**WT: 28 Kg**

Figure 6. TE-364 Motor/Payload Attach Fittings

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 computer and run on mechanical energy until reaching a predetermined time to fire the spacecraft separation clampband bolt cutters and a pair of squib switches. Two seconds later the squib switches initiate a small rocket or yo weight to tumble the expended third stage motor.

For users requiring real time third stage motor performance, environmental or velocity increment information an "S" band telemetry system and a "C" band tracking beacon are developed and flight proven. These are carried on either the spacecraft attach fitting or on the third stage motor.

The new 2.44 meter diameter spacecraft fairing is aluminum and constructed in two half-shells that are brought up around the spacecraft laterally. A contamination-free thrusting joint between the fairing halves is used to thrust the two shells laterally and clear of the spacecraft and vehicle (Figure 3). Normally, the fairing is jettisoned within 15 to 30 seconds after second stage engine 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 1,135 watts per square meter or about equivalent to the solar heating rate to the spacecraft. Aerodynamic heating of the fairing is controlled, if required for the spacecraft, by application of ablative materials to the external surface of the fairing.

Access ports through the fairing and R. F. transparent windows are provided at the locations that meet the needs of the vehicle user. The available fairing internal envelope is shown in Figure 7.

## Flight Sequence and Performance

The Delta flight profile and sequence of events for a three stage geosynchronous transfer mission having a perigee altitude of 185 kilometers, an apogee altitude of 35,790 kilometers and an inclination of 28.7 degrees is shown in Figure 8. The vehicle is launched from ETR on an azimuth of 95 degrees. The third stage assembly is placed into a 167 by 260 kilometer parking orbit with the booster and first burn of second stage. The second stage and third stage assembly then coast to a point just short of the Equator where the second stage is reburned and then the third stage is spun-up, separated, and ignited. The third stage burns out directly over the Equator at an altitude of 185 kilometers, an inertial flight path angle of zero degrees, and with sufficient velocity to coast the spacecraft to an altitude of 35,790 kilometers 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.

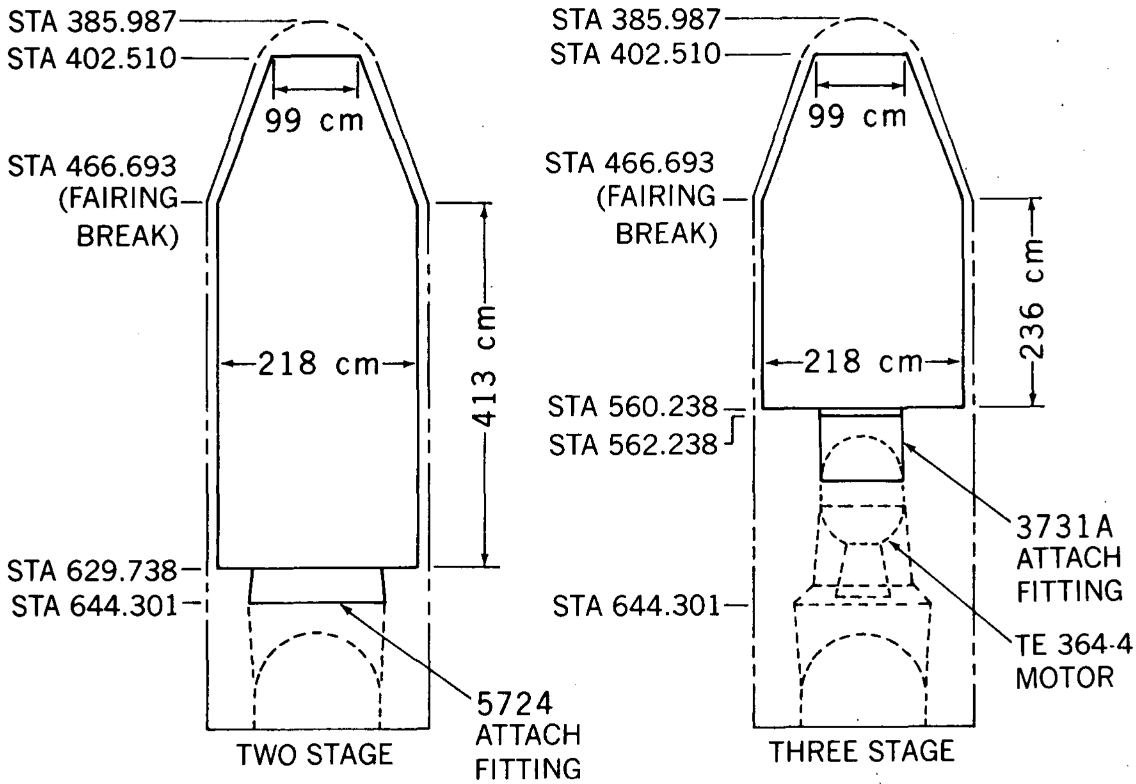



Figure 7. Delta Payload Envelope

Payload weight versus characteristic inertial velocity for Delta from the Eastern Test Range in Florida and the Western Test Range in California is shown in Figures 9 and 10. The performance capability for a number of scientific and applications missions carried on Delta is summarized in Table 2. These Delta performance capabilities are the useful load that can be carried above the last powered stage and thus includes the spacecraft weight, its attach fitting, and the third stage telemetry and tracking system weight, if one is provided. The definition of the Delta model number nomenclatures noted on Figures 9 and 10 for the Delta Model 2914 series described here is as follows: the first digit (2) designates the extended long tank UBT with the new RS-27 engine; the second digit (3, 6 or 9) designates the number of Castor II thrust augmentation solid motors used; the third digit (1) that the second stage incorporates the TR-201 engine, $N_2O_4$/Aerozine 50 propellants, DIGS, and the new 2.44 meter diameter metal fairing; and the fourth digit (0 or 4) that no (0) third stage is used or the TE-364-4 (4) third stage solid motor is used.

The injection accuracy of Delta is strongly dependent on whether the vehicle is two or three stages and on the trajectory profile. The orbital accuracy achieved

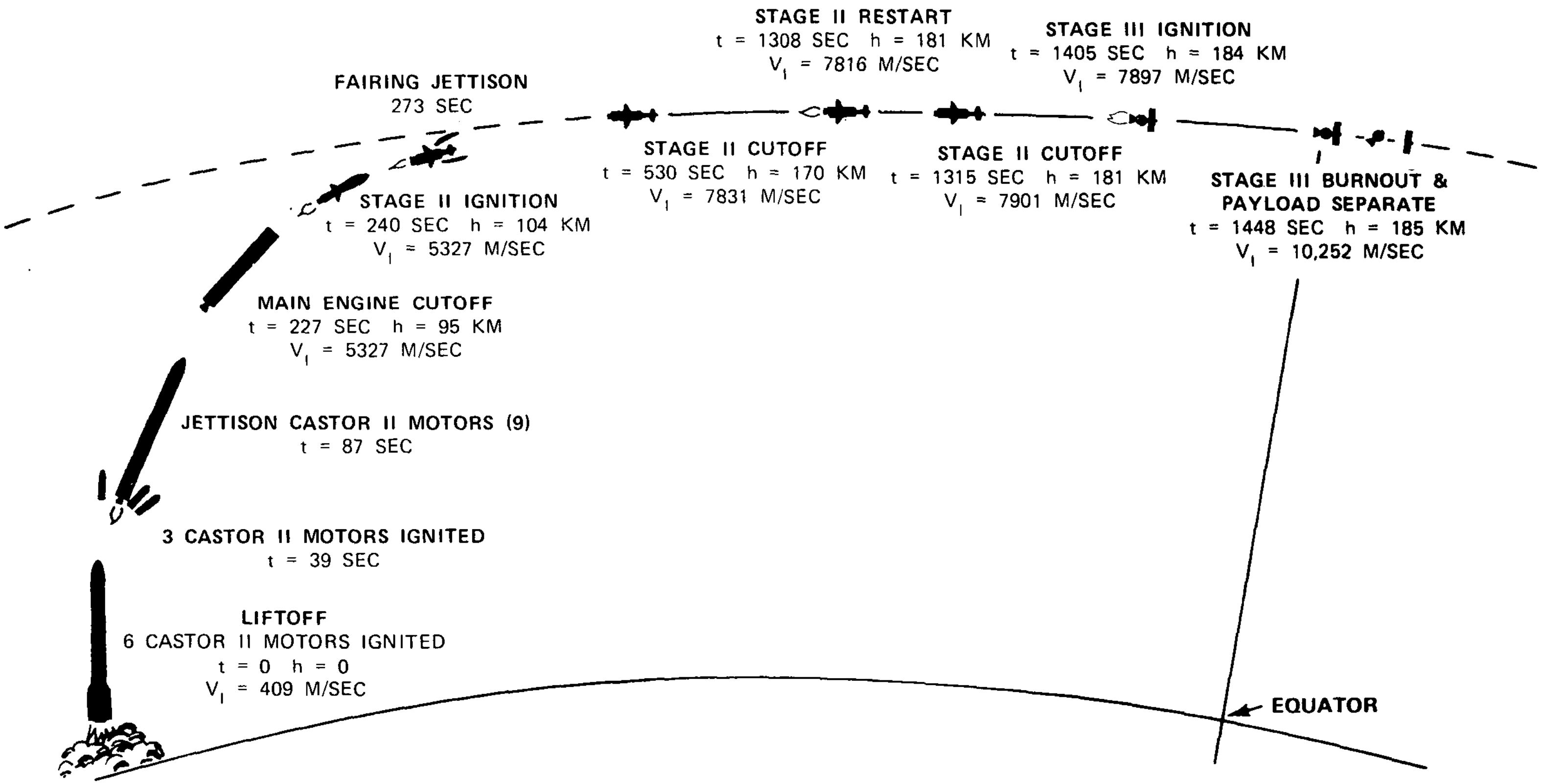



Figure 8. Typical Three Stage Geosynchronous Mission Profile

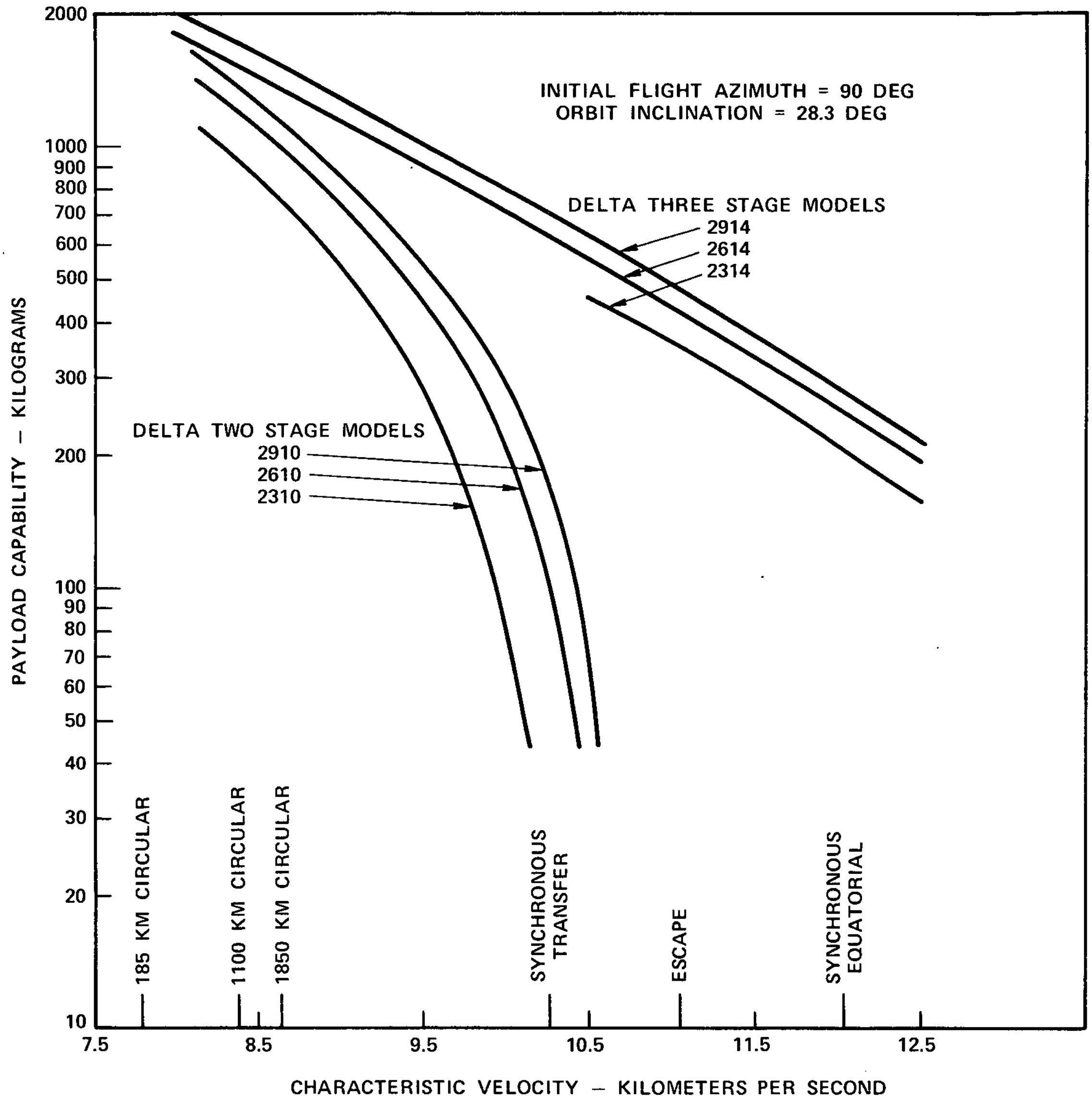



Figure 9. Delta Payload Capability - Eastern Test Range

on those flights since the introduction of the DIGS is shown in Table 3. The difference between the predicted and achieved orbital parameters are compared with the predicted three sigma ($3\sigma$) deviations in orbital parameters. The two stage missions are inertially guided and controlled up through injection. For three stage missions, inertial guidance and control is maintained up until the unguided, spin-stabilized third stage is separated from the second stage. Nearly two thirds of the errors in injection velocity and attitude on a three stage mission 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.

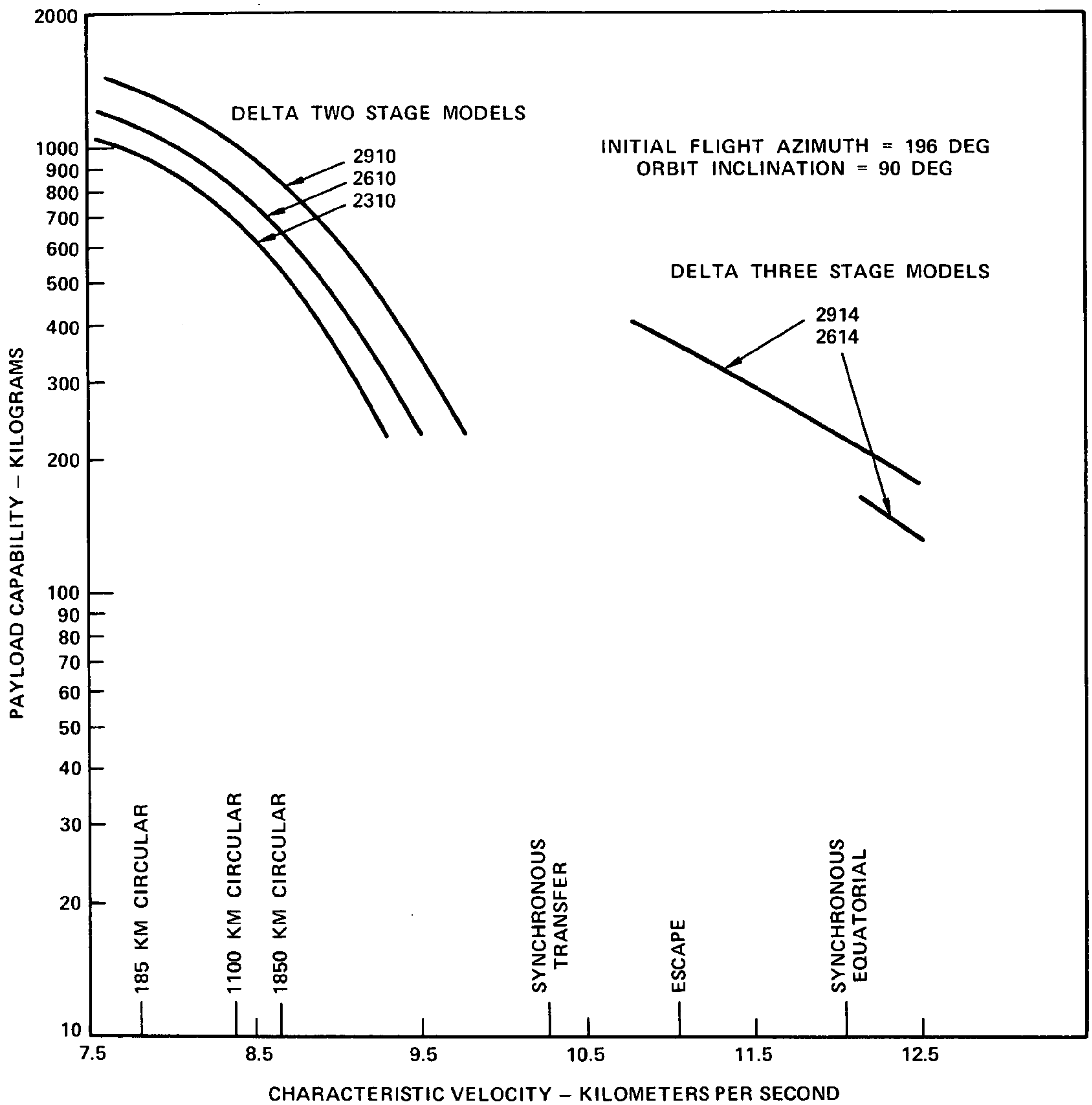



Figure 10. Delta Payload Capability - Western Test Range

Though the three stage dispersions are moderately large, it should be remembered that typically this class of missions, for example, the synchronous communications satellites, carry a propulsion system for velocity adjustment or station keeping and hence the penalty paid in additional propellant to trim out injection errors is quite small.

Table 2

Delta Performance Capabilities

| Two Stage Configurations | | Delta Performance Capability Kilograms | | |
|---|---|---|---|---|
| | | Model Number | | |
| Missions | Flight Mode | 2310 | 2610 | 2910 |
| Biosatellite<br>370 KM Circular<br>Incl. = 28 Degrees | Two Stage Restart<br>185 x 370 KM<br>Hohmann Transfer | 1540 | 1815 | 1950 |
| Earth Resources<br>925 KM Circular<br>Sun-Synchronous | Two Stage Restart<br>185 x 925 KM<br>Hohmann Transfer | 885 | 1065 | 1180 |
| Improved TIROS<br>Operational Satellite<br>(ITOS) 1480 KM<br>Circular Sun-Synchronous<br>Incl. = 102 Degrees | Two Stage Restart<br>185 x 1480 KM<br>Hohmann Transfer | 700 | 840 | 950 |
| Three Stage Configurations | | Model Number | | |
| Missions | Flight Mode | 2314 | 2614 | 2914 |
| SKYNET II<br>Synch Transfer<br>185 x 35,790 KM<br>Incl. = 28.5 Degrees | Three Stage<br>Direct Ascent<br>Second Stage Placed in 185 KM Parking Orbit | 555 | 635 | 705 |
| Planetary Explorer<br>VENUS TYPE II<br>$C_3 = 8.231\ KM^2/Sec^2$ | Three Stage<br>Direct Ascent<br>Second Stage Placed in 185 KM Parking Orbit | 360 | 410 | 440 |

# Table 3

## Delta Orbital Accuracy

| Mission | Orbital Parameters | Predicted Orbit | Achieved Orbit | Orbit Deviation | Predicted 3 Sigma Deviation |
|---|---|---|---|---|---|
| | Sun-Synchronous | | | | |
| ERTS-A | $h_a$ (km) | 919.19 | 918.6 | -0.59 | -5.9 to +6.5 |
| DELTA 89 | $h_p$ (km) | 908.35 | 905.5 | -2.85 | -13.1 to +10.5 |
| 7/23/73 | i (deg.) | 99.095 | 99.118 | +0.023 | -0.045 to +0.45 |
| | Highly Eliptical | | | | |
| IMP-H | $h_a$ (km) | 242,357 | 247,342 | +4,985 | -28,263 to +31,607 |
| DELTA 90 | $h_p$ (km) | 248.3 | 249.1 | +0.8 | -10.0 to 10.0 |
| 9/22/72 | i (deg.) | 28.81 | 28.65 | +0.16 | -0.63 to +0.63 |
| | Sun-Synchronous | | | | |
| ITOS-D | $h_a$ (km) | 148.0 | 1,468.2 | +0.2 | -6.3 to +12.9 |
| DELTA 91 | $h_p$ (km) | 1,465.2 | 1,457.6 | -7.6 | -16.3 to +6.5 |
| 10/15/72 | i (deg.) | 101.76 | 101.74 | -0.02 | +0.043 |
| | Geo-Synchronous Transfer | | | | |
| TELESAT-A | $h_a$ (km) | 36,158.4 | 36,482.5 | +324.1 | -886 to +842 |
| DELTA 92 | $h_p$ (km) | 194.5 | 194.5 | 0 | -18.6 to +18.6 |
| 11/9/72 | i (deg.) | 27.0 | 27.0 | 0 | -0.35 to +0.35 |
| | Sun-Synchronous | | | | |
| NIMBUS-E | $h_a$ (km) | 1,108.3 | 1,104.4 | -3.9 | -4.6 to +16 |
| DELTA 93 | $h_p$ (km) | 1,105.7 | 1,103.4 | -2.2 | -4.6 to +12.5 |
| 12/10/72 | i (deg.) | 99.96 | 99.94 | -.02 | +0.044 |
| | Geo-Synchronous Transfer | | | | |
| TELESAT-B | $h_a$ (km) | 36,183.3 | 36,484.4 | +301.1 | -886 to +842 |
| DELTA 94 | $h_p$ (km) | 213.0 | 212.517 | -0.5 | -18.6 to 18.6 |
| 4/20/73 | i (deg.) | 26.8 | 26.73 | -0.07 | -0.35 to 0.35 |
| | Lunar Transfer | | | | |
| RAE-B | $h_a$ (km) | 401,601.5 | 389,696.2 | -11,905.3 | -31,900 to 32,930 |
| DELTA 95 | $h_p$ (km) | 179.8 | 180.3 | +0.5 | +2 to 1.6 |
| 6/9/73 | i (deg.) | 29.22 | 29.10 | -0.12 | -0.48 to 0.50 |

## Flight Environment

The environment imposed on the spacecraft by the vehicle is estimated from previous flight measurements. A summary of the expected environment for both the two and three stage Delta vehicle is provided in Table 4 for use in preliminary studies by spacecraft mission planners.

At liftoff the spacecraft is subjected to both lateral and longitudinal sinusoidal vibration that load the spacecraft structure dynamically. At the time the three stage Delta lifts off the launch pins and the umbilicals are simultaneously retracted, a 455 kilogram spacecraft can experience a maximum of ±1.5 g, zero-to-peak (0-P), in the vehicle lateral modal frequencies, the most significant of which are below 35 Hz. Superimposed at this time is a ±1.5 g (0-P) longitudinal oscillation. These combined liftoff oscillations typically last for two to five seconds with the peak acceleration lasting one to two cycles. During the last twenty seconds of first stage flight, the Thor exhibits a 20 Hz "pogo" longitudinal oscillation that builds up to ±4.5 g (0-P). For conservatism it is assumed here that the Delta booster with the RS-27 engine will exhibit the same phenomenon and oscillatory acceleration levels. The maximum first stage steady state acceleration of 8.5 g's is the highest imposed by the two stage Delta. For three stage Delta, the maximum steady state acceleration is dictated by the TE-364-4 third stage and reaches 21.5 g's for a 225 kilogram spacecraft; or 9.5 g's for a 680 kilogram spacecraft.

Random vibration measured at the third stage attach fitting and spacecraft show power spectrum densities that range from 20 Hz to 2000 Hz in both lateral and longitudinal axes. The principal source of random vibration is acoustic excitation with secondary levels caused by boundary layer turbulence over the fairing and the second stage that excites the structure and feeds up through the third stage assembly to the base of the spacecraft.

At liftoff and transonic the overall acoustical level inside the fairing is approximately 145 db (referenced to 0.0002 dynes/cm$^2$) from 37.5 to 9600 Hz. These levels are present for about 5 seconds at liftoff and are about 6 db down for 10 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 one-third millisecond, 1400 g terminal peak saw tooth input.

Table 4

Delta Critical Flight Environment

| Excitation | Flight Event | Duration Seconds | Two Stage (Model 2910) | | Three Stage (Model 2914) | |
|---|---|---|---|---|---|---|
| | | | Frequency, Hz | Level | Frequency, Hz | Level |
| Sinusoidal Vibration<br>Thrust Axis | Lift Off<br>T + 210 sec. | 2 to 5<br>5 to 7 | 5-15<br>15-21<br>21-100 | 1.5 g (0-P)[1]<br>4.0<br>1.5 | 5-15<br>15-21<br>21-100 | 1.5 g (0-P)[1]<br>4.5<br>1.5 |
| Lateral Axis | Lift Off | 2 to 5 | 5-14<br>14-100 | 1.5[1,2]<br>1.0 | 5-14<br>14-100 | 1.5[1,2]<br>1.0 |
| Random Vibration<br>Three Axes | Transonic &<br>Max. Q. | 10 to 15 | 20-300<br>300-700<br>700-2000 | +4 db/Octave<br>$0.07\ g^2/Hz$<br>-3 db/Octave | 20-300<br>300-700<br>700-2000 | +4 db/Octave<br>$0.07\ g^2/Hz$<br>-3 db/Octave |
| Shock | Spacecraft Separation | 0.001 | 1600 g at 0.8 Milliseconds<br>Terminal Peak Saw Tooth | | 1400 g at 0.3 Milliseconds<br>Terminal Peak Saw Tooth | |
| Steady State<br>Acceleration | First Stage Burnout | | | 8.5 g | 7.7 g | |
| | Third Stage Burnout | | | | 21.5 g for 225 kg spacecraft<br>9.5 g for 680 kg spacecraft | |
| Acoustic | Lift Off and Transonic | 10 to 15 | 145 db 37 to 9600 Hz<br>(Peak Level 500 Hz) | | | |

[1] Because of shaker limitations acceleration can be limited to 0.2 cm D.A. displacement of the armature.

[2] For spacecraft weight of 455 kg.

## Cost

The projected recurring cost for a Delta Model 2914 reimbursable launch in 1974-75 from ETR is about $9 million dollars. This cost includes all hardware for the three stage mission; the mission peculiarization and configuration of the hardware; spacecraft integration to the vehicle; sustaining engineering for the analytical programs used to develop a mission trajectory; the checkout of the vehicle in the production area and at the launch site; the NASA Delta Project management and engineering support at Goddard Space Flight Center (GSFC) and Kennedy Space Center (KSC); a pro rata share of ongoing vehicle reliability, support equipment and facility improvements; NASA global tracking and data acquisition support; Air Force launch range support and contract administration and a pro rata share of the depreciation and rental of the facilities and support equipment and tools. The breakdown of these costs shown in Table 5 are based on actual or estimated expenses billed to outside agency users such as ESSA, Comsat, Telesat and ESRO for reimbursement to NASA and a projection of these costs into the 1974 time frame when the Delta Model 2914 series shall be operational. Actual charges for any given mission will, of course, vary to reflect the specific mission requirements.

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.

In addition to the recurring costs shown in Table 5 is a one time, first of a mission series, nonrecurring cost that historically has averaged about $300,000 for the initial mission analyses, spacecraft/vehicle interface drawings, spacecraft blockhouse wiring drawings, spacecraft fairing thermal analyses and development of the mission trajectory.

# ORGANIZATION AND INTERFACES

Delta users interface organizationally with three elements within NASA. This is best illustrated by the relationship that existed between Telesat Canada and the NASA Delta Project on the Delta 92 launch as is shown in Figure 11.

Agreement between NASA and a foreign space organization for a launch and associated services is established at NASA Headquarters level. This is normally done with a Memorandum of Understanding outlining the principles under which such arrangements are to be made, followed by a specific contract for each

Table 5

Delta Launch Costs (1974-75)

| MODEL 2914 | COSTS (Thousand Dollars) |
|---|---|
| ● HARDWARE | |
| · First Stage Core | $1,600,000 |
| · Thrust Augmentation Solid Motors (9) | 700,000 |
| · Second Stage and Fairing | 2,000,000 |
| · Third Stage and Attach Fitting | 150,000 |
| ● TRANSPORTATION AND PROPELLANTS | 40,000 |
| ● LAUNCH SERVICES - CONTRACTOR | |
| · Analysis/Software/Sustaining Support | 800,000 |
| · Vehicle Checkout | |
| · Production Area | 300,000 |
| · Launch Site | 700,000 |
| ● SUPPORTING SERVICES - GOVERNMENT | |
| · NASA | |
| · Project Management, Engineering & Overhead | 400,000 |
| · Amortization of Program Improvements | 650,000 |
| · Tracking & Data Acquisition | 100,000* |
| · US Air Force | |
| · Range Costs | 1,200,000 |
| · Contract Administration | 60,000 |
| ● DEPRECIATION OF FACILITIES AND SUPPORT EQUIPMENT | 200,000 |
| TOTAL | $8,900,000 |

*Nominal support, as Tracking & Data Acquisition is highly dependent on mission requirements.

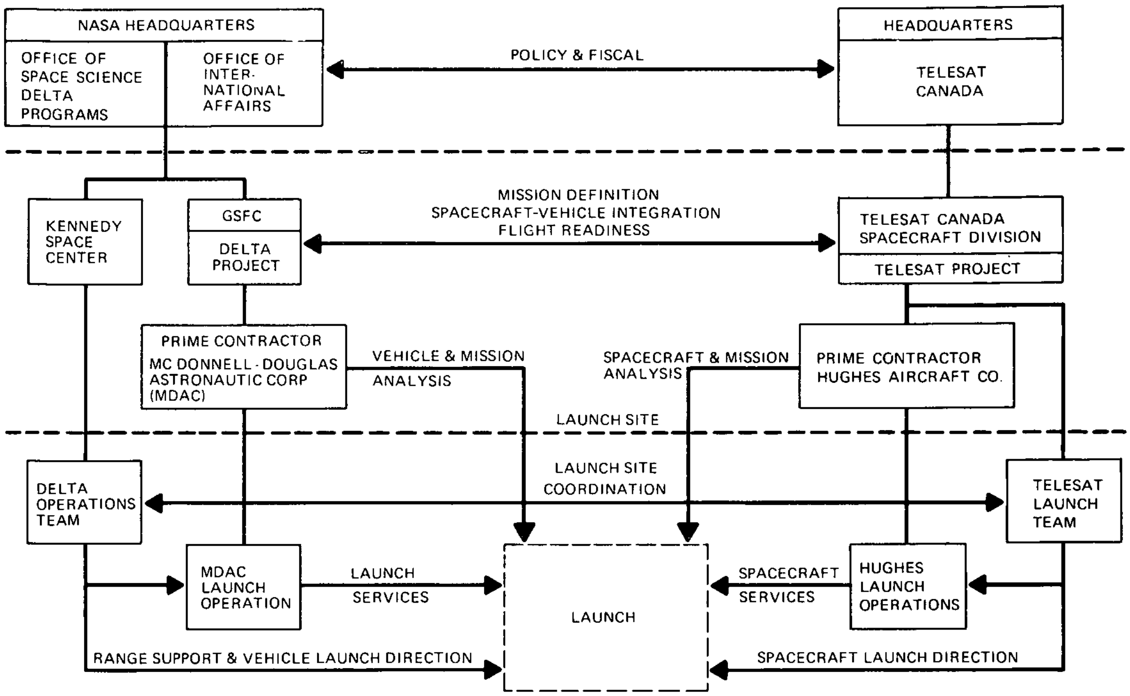



Figure 11. Delta Organization and Interfaces

mission. The agreed-to policies and fiscal arrangements are then passed through the NASA Office of Space Science, Delta Programs Office to the Goddard Space Flight Center (GSFC) Delta Project Office for implementation. The Delta Project Office is vested with the authority and responsibility for carrying out all aspects of a Delta vehicle mission. The Delta Project works directly with the Spacecraft Project to develop and define the spacecraft/vehicle mission requirements, integrate the spacecraft to the vehicle, establish schedules, and determine the final flight readiness of the vehicle. The Delta Project contracts with and directs a single industrial contractor, MDAC, for vehicle hardware, mission analysis, and launch support services. Direction of the launch support services furnished by MDAC at the launch site is delegated to the NASA Kennedy Space Center (KSC). The KSC Delta Operations Team works directly with the Spacecraft Project at the launch site to insure required Range and contractor services are provided and to coordinate the launch site vehicle and spacecraft activities.

This simple organizational structure with short and direct authority and communications lines is a significant factor in the flexibility and responsiveness Delta provides its users.

To request a Delta launch from NASA, the organization seeking launch services should first contact the Delta Project, Code 470, Goddard Space Flight Center, Greenbelt, Maryland 20771 to establish an informal meeting with the program and technical representatives of Delta. At this meeting, the organization seeking launch services would advise NASA of the preliminary design details of the spacecraft and of the mission requirements. NASA in turn would advise the organization of those services it believes can be provided or are required to meet the desired mission objectives and launch schedule. Also, at this time NASA will provide preliminary cost estimates and copies of a typical launch services contract. This informal exchange then would be followed by a formal request addressed to the NASA Administrator (Dr. James C. Fletcher) by the head of the organization requesting the launch services. The letter would describe the launch services required, the Delta vehicle model needed, the desired launch date, by month or quarter within a calendar year and a brief description of the type spacecraft to be launched and its purpose.

In response to the request for launch services, the NASA Administrator will confirm that NASA is prepared to provide the launch services required or if the desired launch schedule cannot be fully met, an indication of the schedule NASA can guarantee, or would attempt to meet on a best-effort basis. These commitments will be embodied in a Memorandum of Understanding between NASA and the organization seeking launch services which will be consummated before negotiations of the launch services contract underway. In addition, NASA will request that, pending satisfactory conclusion of the launch services contract agreement, NASA be authorized to incur costs toward the launch and an immediate initial payment of $100,000 be made to cover costs of preliminary mission studies. Further, if the desired first launch date is less than two years in the future from the date of the Administrator's letter then the initial payment requested will be the amount to bring the progress payment into phase with the payment schedule that is based on a 24 month lead time period as specified in Table 6. The Administrator's letter will also identify the NASA office to which the initial payment is to be forwarded and the NASA representatives who will be points of contact for technical and contractual matters. Upon receipt by NASA of official notification that the conditions stated in the Administrator's letter are acceptable and that initial payment will be promptly made, contract negotiations will commence as soon as is practical; however, the official exchange of letters that constitute a Memorandum of Understanding provides sufficient legal basis for NASA to proceed with the preliminary phases of the launch program.

The Delta Progress Payment Schedule in Table 6 is designed to conform with the approximate rate of expenditures incurred during the 24-month period prior to a launch date. It is a typical schedule which past experience has shown to be practical. However, it is not totally fixed and some variation is possible. For example, it could be arranged for a launch customer to start progress payments 36 or 30 months before launch, instead of 24 months, with appropriate adjustments to the number and size of progress payments. Within the period leading

Table 6

Delta Progress Payment Schedule

| Months to Launch | Percent of Estimated Cost |
|---|---|
| 24 | 5 |
| 21 | 10 |
| 18 | 15 |
| 15 | 15 |
| 12 | 15 |
| 9 | 15 |
| 6 | 10 |
| 3 | 10 |
| 0 | 5* |

*Final cost adjustment and settlement to be made within 2 years from launch.

up to 12 months before launch, however, approximately 60 percent of total costs must be paid to offset vehicle manufacture costs.

Reimbursable launch contracts ordinarily provide for readying of a second vehicle to launch a back-up satellite if required. Normally, however, such back-up launchings are not required.

Availability of a back-up launch vehicle is guaranteed when provided for in the launch contract, in which case the requesting agency would also be expected to make progress payments on the second vehicle, usually being readied six months behind the original vehicle. If the first launch is successful and no back-up launch is desired, virtually all progress payments made on the second vehicle would be refundable, provided that there exists at that time an alternative mission for the vehicle, and except to the extent that mission peculiar modifications may already have been made to the vehicle before the back-up launch is cancelled. If such modifications, which normally are begun about six months before launch, have been made, the requesting agency is then liable for the costs of returning the vehicle to the standard production configuration.

If no provision is made for a back-up vehicle in the launch contract, and a back-up launch becomes necessary due to failure of the initial mission, the customer agency may at that time request NASA to determine if another launch vehicle could be made available during the time period desired. If such a vehicle were available, the requesting agency would be expected to make an immediate progress payment in an amount sufficient to bring its payment into phase with the normal 24 months lead time preceding a launch. If no such vehicle is available, then NASA would not normally be able to provide a back-up launching within a shorter time than the 24 months usually required for preparation of a new vehicle.

If a back-up vehicle is contracted for and not required and an alternative mission exists for it, NASA would refund the progress payments (less any costs incurred to return the vehicle to the standard configuration if it had already been modified) and the vehicle in question would come under NASA control for disposition. It would not be possible for the customer agency to determine the disposition of an unused back-up vehicle. Storage and disposition of an unused back-up spacecraft would, of course, remain the responsibility of the customer agency.

## SPACECRAFT INTEGRATION AND LAUNCH OPERATION

Delta vehicle interface constraints together with performance and accuracy estimates are provided to potential vehicle users as soon as the concept of the mission is outlined to the NASA, Goddard Space Flight Center Delta Project Office. The Delta Project welcomes and encourages early definition of prospective missions by potential users. In some instances, mission definition and integration planning has preceded actual mission commitment by two and three years. Experience has demonstrated that this advance and continuous coordination between the user and the Delta Project during the period of developing mission requirements, enhances the visibility of both parties and reveals problem areas before final definition of the spacecraft/vehicle interface and trajectory parameters. In general, spacecraft/vehicle planning for new missions follow the pattern and time frame outlined in Figure 12 and starts about one year (T-52 weeks) before launch when the Spacecraft Project provides the Preliminary Mission Definition and Requirements to the Delta Project Office. This definition encompasses the preliminary spacecraft configuration, mass properties, trajectory, and orbital requirements necessary for preliminary vehicle performance evaluation and analysis. A preliminary trajectory with attendant injection error studies and thermal studies is completed within ten weeks. With this visibility, the Delta Project and the Spacecraft Project jointly develop a Final Mission Requirement specification (T-40 to T-26 weeks) 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 the Final Mission Analysis.

The Spacecraft Project reviews the final mission trajectory about T-35 weeks and the final injection and orbital error analysis about T-23 weeks. The trajectory includes all 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.

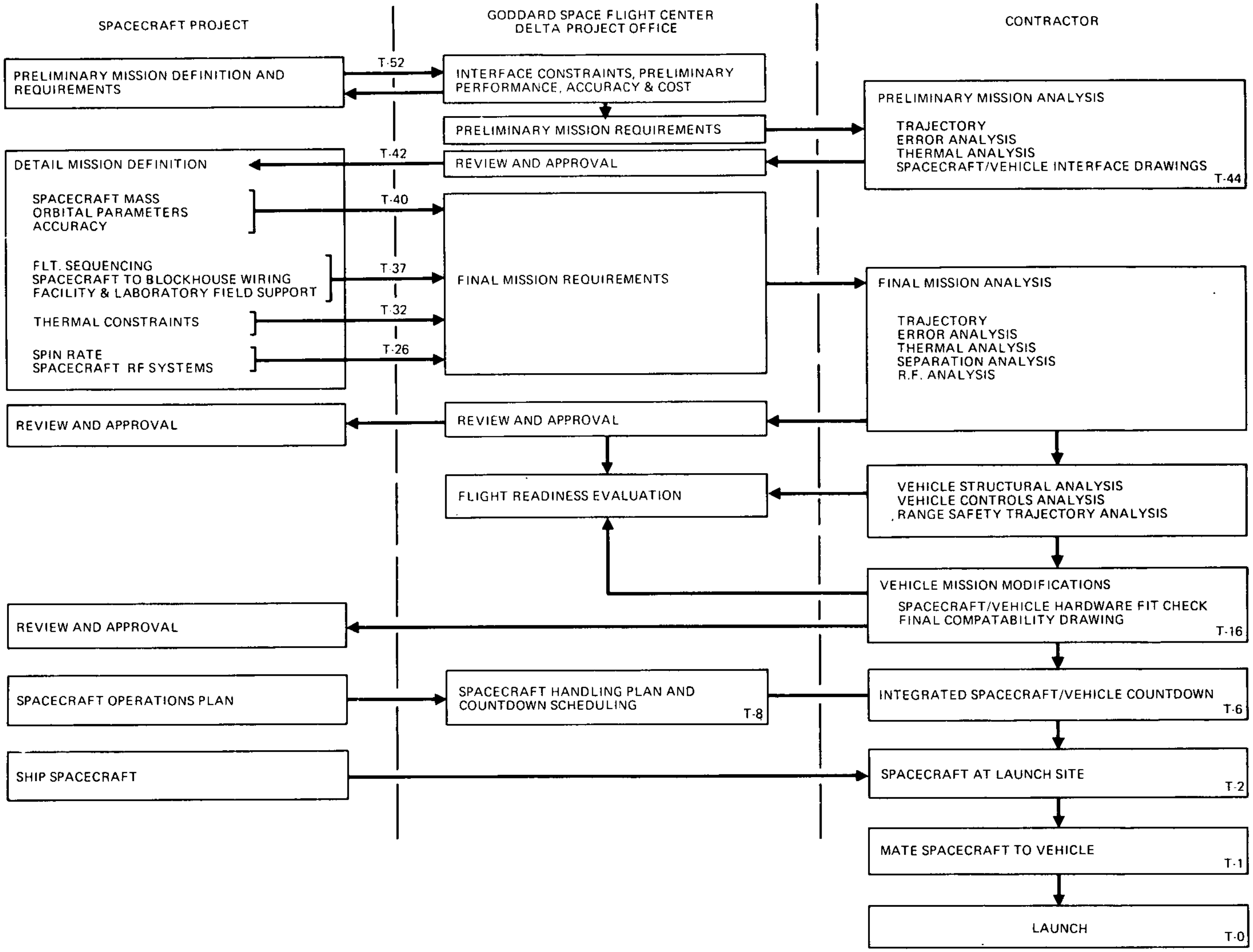



Figure 12. Delta Mission Analysis and Integration

Final definition of the maximum and minimum allowable spin rate, spacecraft RF systems, and permissible inflight thermal inputs are provided to the Delta Project by T-26 weeks. A full scale compatibility drawing based on the Spacecraft Project's final configuration drawings is normally prepared at T-16 weeks. This drawing is primarily to show all clearances 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 Spacecraft Handling Plan is jointly developed and finalized about T-8 weeks and describes all hazardous systems, spacecraft test procedures, and details pre-launch work schedules. Typically the spacecraft arrives at the launch-site two weeks before launch (T-2), is built up on the third stage motor assembly and mated to the vehicle on the pad one week before launch for RFI testing with the vehicle and Range RF systems. Final weights are inputed to trim the final trajectory parameters in the inertial guidance computer the week of launch.

For three stage missions the spacecraft must be statically and dynamically balanced prior to receipt at the launch site. The allowable spacecraft center-of-gravity offset and principle axis misalignment is 0.050 inches and 0.003 radians, respectively. For missions where injection attitude is extremely critical for mission success, a third stage assembly composite spin balance can be conducted.

The Delta Project conducts launches from both ETR and WTR. Prograde missions with orbital inclinations of 30 degrees or less are normally launched from ETR and near-polar or retro-grade missions from WTR, though near polar missions have been launched from ETR.

Facilities for the Spacecraft Project use at the launch site include spacecraft assembly and checkout laboratories, telemetry, fabrication and cryogenic laboratories, clean rooms, shops, storage, and offices.

The first and second stage mission modifications to the vehicle are made in the contractor production area. The first and second stages are delivered directly to the launch pad, erected, and again undergo systems testing. The thrust augmentation solid motors and third stage solid motors are stored and prepared at the launch site. The thrust augmentation solid motors are mated to the first stage on the launch pad about three weeks before launch. The third stage motor is built up on the spin table and the spacecraft mated with the assembly at about T-2 weeks. The spacecraft/third stage assembly is transported in an environmentally controlled canister to an environmental room on top of the mobile service tower around the vehicle and there the assembly is mated to the vehicle. While the spacecraft is mated to the vehicle, spacecraft and vehicle checkout and testing is interspersed whenever possible to accommodate the spacecraft requirements.

On pad checkout of the vehicle culminates in a pre-countdown simulated flight without propellant on-board, wherein all systems of the vehicle are exercised as they are during the mission. The simulated flight test takes place one week beforelaunch and is followed by final preparation of the vehicle for launch and then a three day countdown to lift-off. If necessary, complete access to the spacecraft can be provided up to four hours prior to liftoff, 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 are provided through the vehicle wiring. While the spacecraft is on the vehicle, thermally and hermetically conditioned, filtered air is provided to the spacecraft right up to lift-off.

Launches off the same pad at ETR have been conducted at two week intervals, though four weeks is the normal operation. For follow-on spacecraft in a mission series a called-up launch can be made on 120 days notice at no increase in launch costs provided it is an identical mission, the mission peculiar hardware (attach fitting, etc.) have been provisioned and the launch does not impact another scheduled mission. Call-up time may be reduced to 60 days at a cost of about $200,000 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 $300,000 of non-recoverable funds if call-up is not exercised.

Based on 12 years of experience at ETR, the probability of launching in a window 15 seconds wide on a given day is about 70 percent. The probability for a thirty minute launch window, typical for most missions, is about 90 percent. Recently, when spacecraft development delays impacted a number of Delta launches of lower priority and a call-up launch option was suddenly exercised, the Delta Project launched three spacecraft at one-week intervals to relieve a congestion of six spacecraft awaiting launch. Normally, however, the scheduled launch date desired by the Spacecraft Project can be met.