5-1991

International launch vehicles database with groundtrace capability

Kenol Jules

To the Graduate Council:

I am submitting herewith a thesis written by Kenol Jules entitled "International launch vehicles database with groundtrace capability." I have examined the final electronic copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aerospace Engineering.

F. Shahroki, Major Professor

We have read this thesis and recommend its acceptance:

K. C. Reddy, A. D. Vakili

Accepted for the Council:
Carolyn R. Hodges

Vice Provost and Dean of the Graduate School

(Original signatures are on file with official student records.)

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Vice Provost and Dean of The Graduate School

INTERNATIONAL LAUNCH VEHICLES DATABASE WITH GROUNDTRACE CAPABILITY

A Thesis

Presented for the

Master of Science

Degree

The University of Tennessee, Knoxville

Kenol Jules

May 1991

To My Brother

Jean Max

My Lovely Niece

Slethvana

My good Friend

Rene Noel

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ABSTRACT

The aim of this thesis is three-fold. First, to make available to the space transportation industry a more complete and a more updated informations deck on all the commercial launch vehicles available throughout the world. Second, to use the Orbital Mechanics laws to lay out the process involved in carrying out a groundtrace design. Third, to develop a computer program in support to the database for the launch vehicles and to provide graphics support for the ground trace mapping trajectory. The program was tested for different known cases so as to validate the codes for the ground trace as well as for the database files retrieval system. In all of the cases tested, it works as expected.

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CHAPTER I

INTRODUCTION

The world has become much closer due to the recent development in eastern Europe. As a consequence of that togetherness manifested among the European countries, we might expect a coming down of much of the technological barriers that has divided Europe over the years due to political differences. Once that happens, the level of technology transfer and trade agreement among the Europeans and the rest of the world will exceed anyone's expectation. With that will come greater freedom to use products from the Eastern block by the West and vice versa. One of the benefit of such achievement will be in the arena of launch vehicles. The door will be wide open for the East to commercialize its launch vehicles and compete with the Western market. Also, the West will have a much broader option as to which launch vehicle on the international market is more cost effective for a particular mission. In addition, the third world countries will have lots of freedom in selecting the launch vehicle that best fit their interest since they will no longer have to make a choice based on political constraints. Because of such anticipated needs, this project has become a matter of considerable interest.

This thesis presents most of the technical informations on commercial launch vehicles available worldwide. It also details the groundtrace procedure and explains the user friendly program written to support both the database for the launch vehicles and the trajectory mapping for the ground-trace.

Many articles had been written before in the past on a specific launch vehicle. However, each conveys a very limited amount of technical information on the launch vehicle in question. Only one research work was done in the past to gather most of the known launch vehicles worldwide under one umbrella, Jane's Spaceflight Directory, 1988-89. However, the amount of technical information given in that published work fell short of being significant to anyone wishes to analyze any launch vehicle in that research. However, it remains the only source of information available. The purpose of this thesis is to extend the work done in Jane's Spacefligth Directory and make these technical informations accessible as a database.

The first part of this thesis presents the technical informations on seventeen launch vehicles with different configurations. The data given ranging from the vehicle characteristics, propulsion system description, payload capacity to mission profile.

The second part details all the steps and assumptions involved in carrying out a groundtrace as well as the position of the vehicle in space in a vectorial form.

The third part involves a user friendly program designs to allow a user to select a launch vehicle based on certain requirements imposed by the user. The computer program is made up of the three subroutines.

1. To select a launch vehicle:

This subroutine has a database made up of all the different configurations for each launch vehicles. The database contains information on maximum payload weight, maximum payload bay volume, and orbit. These information enable one to lock up on one launch vehicle for a specified payload.

2. To access the launch vehicle database:

This subroutine is linked to the database that contains all the launch vehicles technical information. All that is required is to input the of the launch vehicle selected by the previous subroutine.

3. To access the ground trace facility:

This subroutine calculates all the six Orbital Mechanics elements and the initial position and velocity of the spacecraft in vectorial form. Also, it accesses plot1 graph subroutine to map the spacecraft trajectory. For this mode the user has to input: latitude and longitude of the launch site, initial velocity of the spacecraft, altitude, heading, date/month and year of the launch (see flow chart figure 1.0). The mapping of the spacecraft trajectory generates by the calculation carried out in the Orbital Mechanics part is good for only one pass due to the two-body assumption made in carrying out the calculation. Therefore, drift causes by drag perturbations was

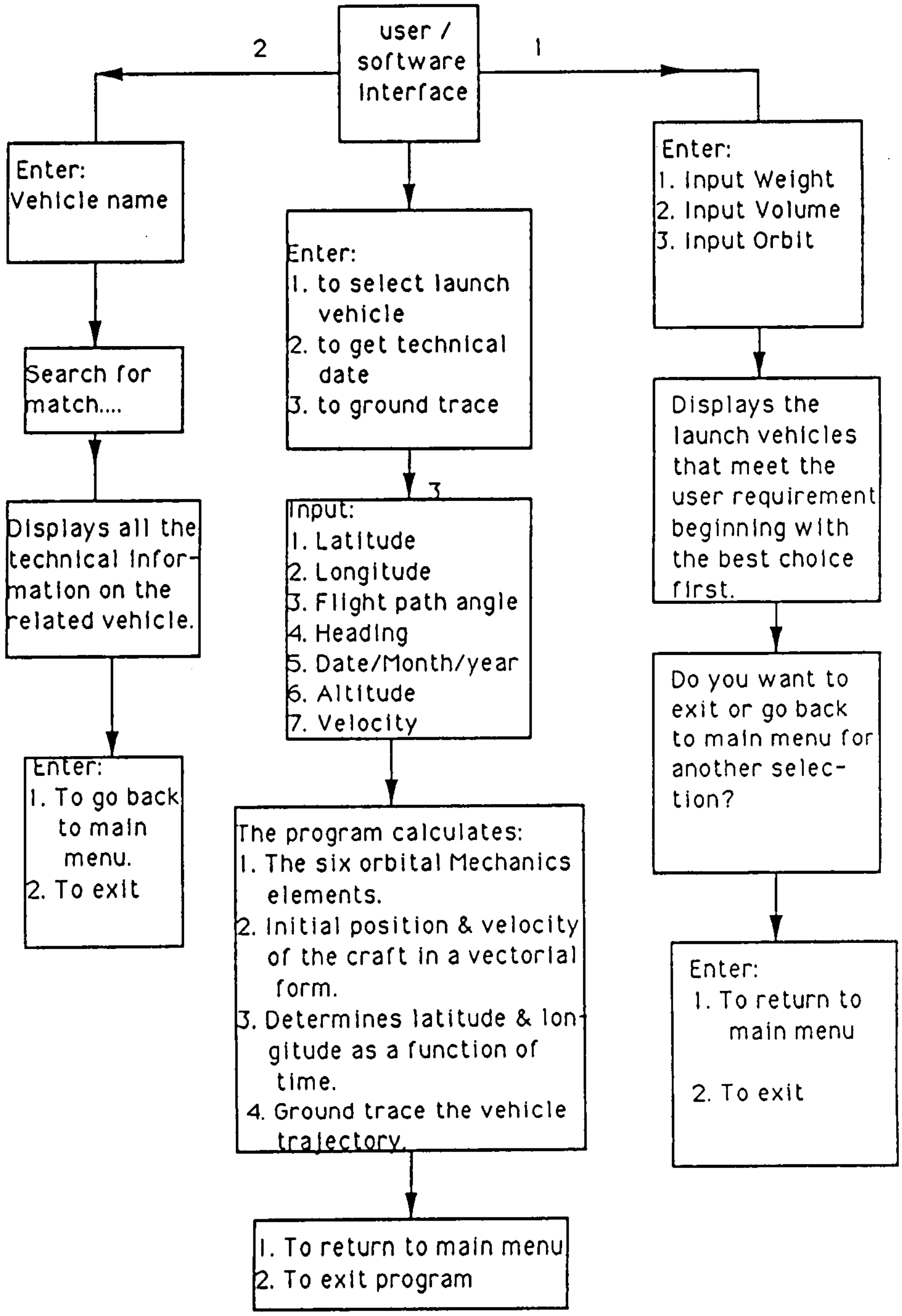

Fig. 1.0 Flow Chart

not taken into consideration. The mapping is a complement to the mission profile sequence of each launch vehicle review in this work. Thus, not only the mission profile sequence technical information is given for each launch vehicle, but also the first pass of the spacecraft can be seen (graph) from the ground trace mapping facility which makes the entire process complete (from launch to orbit). The fortran program was verified with a known ground trace case [1], provided by the National Space Development Agency of Japan (NASDA), of a Broadcasting satellite Bs-3a launch by H-I launch vehicle on August 1990.

The last part presents the discussion, conclusion and recommendations for future work in that area.

DATA COLLECTION AND PROCEDURE:

In order to obtain the technical information necessary for the database, a wish list, containing all the technical information that I feel would be of interest to anyone in the launching / Aerospace business, was designed. That wish list was designed purely based on my interest in the advanced propulsion system and my background in rocket propulsion / turbomachinery. Besides that, a lot of technical papers published on all the vehicles reviewed in this work were read to gain sufficient knowledge in order to create a presentable database. Futhermore, each table was carefully designed to match the need of each launch vehicle so that the technical information can be read with ease. The wish list was sent to the following companies:

General Dynamic Commercial Launch Service (Atlas); Mc Donnell Douglas Commercial Delta, inc. (Delta); Orbital Sciences Corporation (Pegasus and Taurus); National Space Development Agency of Japan (H-I and H-II); Martin Marietta Space Launch System (Titan II and Titan IV); Arianespace, inc. (Ariane-4 and 5); Glavcosmos USSR (Soviet vehicles); Licensintorg (Soviet vehicles); Missiles and Electronic Group Missile Division (Scout); China Great Wall Industry Corp. Space Division (Long March); Indian Satellite Research Organisation (PLSV); NASA Jhonson Space Center, Kennedy Space Center, Marshall Space Flight Center (Space Shuttle).

Whenever enough information was not available from the litterature, the manager in charge for each program or the chief designer of the launch vehicle was contacted by telephone to seek more technical information on the specific launch vehicle or to verify some of them. Futhermore, when enough parameters were known and others were missing and could be calculated, the following rocket equations [2] were used to find them:

$$m = \sqrt{\frac{\gamma g_c w}{R}} \left[\frac{\gamma + 1}{2}\right]^{-\frac{(\gamma+1)}{2(\gamma-1)}} \frac{P_c A}{\sqrt{T_c}}$$

$$u_e = \left[2C_p T_c \{1 - (\frac{P_e}{P_c})^{\frac{\gamma-1}{\gamma}}\}\right]^{\frac{1}{2}}$$

$$f = \frac{m u_e}{g_c} + (P_e - P_a)A_e$$

$$I_{sp} = \frac{u_e}{g} + \frac{(P_e - P_a)A_e g_c}{mg}$$

$$\rho(h) = 0.075 Exp(-7.4 * 10^{-6} h^{1.15})$$

$$g = g_e(\frac{R_e}{R_o + h})^2$$

$$H = u_e * t$$

WISH LIST

The following information was requested from the companies mentioned earlier:

1. Active configuration(s) of the launch vehicle(s).
2. Specification of each configuration: systems description and systems performance.

a. Overall length; total weight of each configuration.

b. Length of each stage

c. Propellant used for each stage and any strap-on boosters

d. Description of type of propellant: static and vacuum thrust for each stage, static and vacuum specific impulse for each stage

e. Volume of the payload bay/configuration

f. Type of orbit service/configuration

g. Maximum payload delivery for each orbit/configuration

h. Launching site facility including latitude, longitude and inclination for the launch site

i. Cost per launch/configuration

j. A comprehensive background (History) on the vehicle

k. Rate of success as of now/configuration

l. Mission profile including performance graph (Weight Vs. orbit)/configuration

m. Dual or Triple launch capability? if yes, Price per payload? and for which orbit?

n. Launch policy in case of failure- relaunch of payload? waiting period (how long)? price for a relaunch? insurance?

and any useful information in that regard

o. Does the price for a launch include site and launch support service? if so what do you offer (give) as launch support?

p. Delta V for each stage (margin of safety for each stage in case of failure).

q. A copy of the payload planners guide (payload users manual) of the launch vehicle(s)

CHAPTER II

U.S LAUNCH VEHICLES

2.1 DELTA II 6925

BACKGROUND ON THE VEHICLE

The Delta II program officially began in February of 1989 with the launch of the first of the United States Air Force (USAF) Navstar II Gloval Positioning System (GPS) satellites. The Delta launch vehicle had its beginnings in 1960, when the first launch of the new, NASA-developed vehicle went aloft with an Echo passive communications satellite. The vehicle rapidly developed into a NASA workhorse, and through the years was continuously uprated to meet the growing needs of the satellite community. Many of those upratings involved changes in the vehicle propulsion tankage capacities, uprating first and second stage engines, increasing size of the third stage solid motors, and adding strap-on solid rocket motors (SRMs) for boost assist [3].

DESCRIPTION OF THE LAUNCH VEHICLE

The 6000 series booster configuration uses the standard RS-27 engine with an 8:1 expansion ratio and the Castor IVA solid rocket strap-ons. The characteristics and capabilities of this launch vehicle are summarized in tables 2.1.1-2.1.4. Delta II 6925 is a three stage launch vehicle. However, using different configurations (see table 2.1.5), it can place 8780

Table 2.1.1 Delta II 6925 Vehicle Characteristics

	Straps-On solid	**First stage**	**Second stage**	**Third stage**
Length (ft)	36.6	85.6	19.6	6.7
Diameter (ft)	3.3	8.0	8.0	4.1
Total weight (lb)	25,822* ; 26,118**	223,843	15,394	4,721
Engine/Motor	Castor IVA	RS-270/B	AJ10-118K	Star-48B
Manufacturer	MTI	Rocketdyne	Aerojet	MTI
Quantity	9	1	1	1
Propellants	Solid	LOX/RP-1	N_2O_4/A-50	Solid
Propellant weight (lb)	22,292 each	211,337	13,367	4,430
Thrust)$_{\text{sea level}}$ (lb)	97,070 each	207,000	----	----
Thrust,)$_{\text{in vacuum}}$ (lb)	108,700 each	231,700	9,645	15,100
Isp)$_{\text{sea level}}$ (sec)	237.3	263.2	----	----
Isp)$_{\text{vacuum}}$ (sec)	265.7	295.0	319.4	292.6
Burning time (sec)	56.2	264.9	439.7	87.1
Expansion ratio	8.29	8:1	65	54.8

* ground lit

** Air lit

Table 2.1.2 Delta II 6925 Launch Vehicle

Program :	Commercial
Prime contrator:	Mc Donnell Douglas
Major subcontractors	Rocketdyne/Morton Thiokol AerojetTechsystems/Delco
Vehicle height:	125.2 ft/38.2m
Liftoff weight:	479,800 lb/217,640 kg
Liftoff thrust (Main engine + six solids)	595,000 lb/2,646 kn
Maximun dynamic pressure	1,180 lb/ft2 / 56,500 n/m2
Maximun acceleration	5.77 g's
Maximun payload to GTO	3,190 lb/1,447 kg
Maximun payload to LEO	8780 lb/3983 kg

Table 2.1.3 Typical Delta II 6925 Mission Profile-GTO

Events	V1 (ft/s)	Acceleration (g's)	Time (sec)
Liftoff	1343	1.31	0
6 SRMS Burnout	2826	0.39	56
3 SMRs ignition	———	———	61
6 SRMs Drop	———	———	63
3 SRMs Drop	———	———	122
MECO	18,701	5.77	265
SECO I	25,870	0.72	680
SECO II	26,307	0.80	1311
Stage III Burnout	33,595	4.24	1488
Spacecraft separation	———	———	1601

NOTE: The data in table 2.1.3 are for an Eastern launch site, flight azimuth 95 degrees; maximum capability to 28.7 degree inclined GTO, 100-nm perigee.

Table 2.1.4 Delta II 6925 Mission Capabilities

Three-stage missions	**Spacecraft Weight (lb/kg)**
GTO: ESMC, i = 28.7 degrees 100 x 19,323 nmi / 185 x 35,786 km	3190/1447
Molniya orbit: WSMC, i = 64.4 degrees 200 X 21,649 nmi / 370 x 40,094 km	2120/962

Table 2.1.5 Delta II 6920 Mission Capabilities

Two-stage missions	**Spacecraft Weight (lb/kg)**
Low earth orbit (LEO): ESMC, i = 28.7 degrees 100 nmi / 185 km circular	8780/3983
Low earth orbit: WSMC, i = 90 degrees 100 nmi / 185 km circular	6490/2943
Sun-synchronous orbit: WSMC, i = 98.7 degrees 450 nmi / 833 km circular	5320/2413

NOTE: Delta II 6920 is the Two-stage configuration for Delta II 6925.

lb for a two stage configuration into a low earth orbit and 3190 lb into a geosynchronous transfer orbit (GTO) for a three stage configuration [3].

Inboard Profile

1. **First Stage**. The first stage has an engine section that houses the Rocketdyne RS-27 main engine, two rocketdyne LR101-NA-11 vernier engines, and provides the aft attachments for the strap-on solid propellant motors. The RS-27 is a single-start, liquid bipropellant rocket engine with a thrust rating of 207,000 lb (921 KN) at sea level. The two vernier engines provide roll control during main-engine burn, and attitude control after cutoff and before second-stage separation. Thrust augmentation is provided by nine unsegmented solid propellant rocket motors, six ignited at liftoff and the remaining three ignited in flight. A rate gyro has been added to the first stage, forward of the center body section, to assure adequate stability margins [4].

1. **Second Stage**. The second stage uses the restartable Aerojet AJ10-118K engine developed for the USAF Improved Transtage Injector program, and uses nitrogen tetroxide and Aerozine-50 storable propellants. Gaseous helium is used for pressurization, and a nitrogen cold gas jet system provides attitude control during coast periods and roll control durint powered flight. The Delta inertial guidance system is a strap-down all-inertial system consisting of a Delta redundant inertial measurement system and a Delco guidance computer. The Delta

redundant inertial measurement system contains three gyros, four accelerometers, and conditioning electronics. Both first and second stages have a battery-supplied DC power system. Separate batteries are used for the guidance and control system, ordnance, engine systems [4].

3. **Third Stage**. The vehicle's third stage is derived from components and concepts used on the Delta third-stage and the USAF SGS-II upper stage. The Star-48B solid rocket motor is supported at the base of the motor on a spin table that mates to the top of the second-stage guidance section. The payload attach fitting is the structure that provides the transition from the top of the solid-rocket motor to the spacecraft interface. The stage also contains a nutation control system to suppress coning, and an S-band T/M system [4].

2.2 DELTA II 7925

The 7000 series booster configuration is almost the same with the 6000 series except that it uses the modified RS-27 engine with a 12:1 expansion ratio (whereas the 6000 series has 8:1 expansion ratio) and the Hercules lightweight, GEM (Graphite Epoxy Motors) solid rocket strap-ons. The features of this launch vehicle are summarized in tables 2.2.1-2.2.5 [4].

Table 2.2.1. Delta II 7925 Vehicle Characteristics

	Straps-On solid	First stage	Second stage	Third stage
Length (ft)	42.5	85.6	19.6	6.7
Diameter (ft)	3.3	8.0	8.0	4.1
Total weight (lb)	28,618* '28,800**	224,239	15,394	4,721
Engine / Motor	GEM	RS-270/C	AJ10-118K	Star-48B
Manufacturer	Hercules	Rocketdyne	Aerojet	MTI
Quantity	9	1	1	1
Propellants	Solid	LOX/RP-1	N_2O_4/A-50	Solid
Propellant weight (lb)	25,800 each	211,147	13,367	4,430
$Thrust_{)\, sea\ level}$ (lb)	98,870 each	201,000	----	----
$Thrust_{)\, in\ vacuum}$ (lb)	110,820 each	237,000	9,645	15,100
$Isp_{)\, sea\ level}$ (sec)	245.7	255.6	----	----
$Isp_{)\, vacuum}$ (sec)	273.8	301.8	319.4	292.6
Burning time (sec)	63	265.4	439.7	87.1
Expansion ratio	8.29	12:1	65	54.8

*Ground lift

** Air lift

NOTE: The data in table 2.2.1 are for an Eastern launch site, flight azimuth 95 degrees; maximum capability to 28.7 degree inclined GTO, 100-nm perigee.

Table 2.2.2 Delta II 7925 Launch Vehicle

Program :	Commercial
Prime contrator:	Mc Donnell Douglas
Major subcontractors	Rocketdyne / Morton Thiokol Aerojet Techsystems/Delco Hercules
Vehicle height:	125.2 ft / 38.2 m
Liftoff weight:	506,470 lb / 229,730 kg
Liftoff thrust (Main engine + six solids)	598,000 lb / 2,660 kn
Maximun dynamic pressure	1,235 lb/ft2 / 59,130 N/m2
Maximun acceleration	5.74 g's
Maximun payload to GTO	4,010 lb / 1,819 kg
Maximun payload to LEO	11,110 lb / 5039 kg

Table 2.2.3 Typical Delta II 7925 Mission Profile GTO

Events	V1 (ft/s)	Acceleration (g's)	Time (sec)
Liftoff	1343	1.33	0
6 SRMs Burnout	3258	0.54	63
3 SRMs ignition	______	______	66
6 SRMs Drop	______	______	68
3 SRMs Drop	______	______	133
MECO	19,962	5.74	265
SECO I	25,645	0.66	625
SECO II	27,186	0.75	1311
Stage III Burnout	33,630	3.56	1488
Spacecraft separation	______	______	1601

Table 2.2.4 Delta II 7925 Mission Capabilities

Three-stage missions	Spacecraft Weight (lb/kg)
GTO: ESMC, i = 28.7 degrees 100 x 19,323 nmi / 185 x 35,786 km	4010 / 1819
Molniya orbit: WSMC, i = 64.4 degrees 200 X 21,649 nmi / 370 x 40,094 km	2810 / 1275

Table 2.2.5 Delta II 7920 Mission Capabilities

Two-stage missions	Spacecraft Weight (lb/kg)
Low earth orbit (LEO): ESMC, i = 28.7 degrees 100 nmi / 185 km circular	11,110 / 5039
Low earth orbit: WSMC, i = 90 degrees 100 nmi / 185 km circular	8420 / 3819
Sun-synchronous orbit: WSMC, i = 98.7 degrees 450 nmi / 833 km circular	7000 / 3175

NOTE: Delta II 7920 is the Two-stage configuration for Delta II 7925.

2.3 ATLAS

BACKGROUND ON THE VEHICLE

With its debut in 1957 as an intercontinental ballistic missile, Atlas was initially designed and constructed under a Defense Department contract and was subsequently transfered to NASA. Atlas has a varied and colorful history, having performed a broad spectrum of precedent-setting missions spanning three decades of service. The world's first communication satellite was launched on an Atlas in 1958. Other historic Atlas events include: the first U.S. manned orbital spaceflight by John Glenn, the first lunar mission, the first launch of a liquid hydrogen stage (Centaur), and the dominant portion of all U.S. lunar and planetary missions [5].

The Centaur upper stage was added in 1962, providing a high-energy cryogenic stage to the basic stage-and-a-half Atlas booster (the Atlas/Centaur combination is typically referred to as Atlas).

Reliability of the Atlas has been verified over a flight history of 494 missions, including 68 Atlas/Centaur launches. Statistical reliability calculations indicate a success rate in excess of 95 percent [5].

DESCRIPTION OF THE LAUNCH VEHICLE

The Atlas launch vehicle is comprised of two-and-one-half stages: the booster half-stage, the first or sustainer stage, and the upper Centaur stage. The Atlas uses the Rocketdyne MA-5 engine system, with two booster engines on the booster section and a sustainer engine mounted on the apex of the conical aft tank bulkhead, all ignited simultaneously at lift-off. The sustainer engine on the first stage continues to provide thrust until propellant depletion. The propellant used for the sustainer and booster stages comes from a common reservoir and is a combination of liquid oxygen and RP-1 (fractioned kerosene) [5].

The Centaur is a cryogenic high-energy upper stage using two Pratt & whitney RL-10 engines fueled by liquid hydrogen and liquid oxygen [5]. Each of the two RL-10A-3-3A engines on the Atlas I has a rated thrust of 16,500 pounds, specific impulse of 444.4 seconds, chamber pressure is 475 psia, with an oxidizer-to-propellant ratio of 5:1. Due to the altitude at which the RL-10 is designed to operate, the 61:1 expansion area ratio is more than twice the Atlas sustainer engine's ratio of 25:1. Twelve Hydrazine thrusters are used to provive Centaur control during coast phases and spin-up prior to payload separation. The Centaur carries all of the launch vehicles's guidance and avionics, which are mounted at the top of the stage. Flexible software allows the Atlas vehicle to be used for a variety of missions, all controlled by the Centaur's commands [5].

ATLAS I

The Atlas I launch vehicle system consists of the Atlas booster (two vernier engines are utilized for dirctional control), the Centaur upper stage vehicle, and the payload fairing. The Atlas I represents the lower end of the family's performance spectrum. Table 2.3.1 shows its characteristics.

Table 2.3.1 Atlas I Vehicle Characteristics

Overall Vehicle	Length (m/ft)	With medium fairing	42.0 / 138
		With large fairing	43.9 / 144
	Diameter (m/ft)		3.1 / 10
	Gross liftoff Weight	With medium fairing (kg/lb)	163,900/361,300
		With large fairing (kg/lb)	164,290/362,200
Centaur	Length (m/ft)		9.1 / 30
	Propellant		LH_2 & LO_2
	Propellant Weight (kg/lb)		13,790 / 30,400
	Total $thrust)_{vacuum}$ (KN/lb)		146.8 / 33,000
	Specific $Isp)_{vacuum}$ (sec)		444.4
Atlas	Length (m/ft)		22.2 / 73
	Propellant		LO_2 & RP-1
	Propellant Weight (kg/lb		137,530/303,200
	Booster engines total thrust* (KN/lb)		1,679/377,500
	Booster engines Isp* (sec)		259.1
	Sustainer engine thrust* (KN/lb)		269 / 60,500
	Sustainer engine Isp* (sec)		220.4
	Vernier engines total thrust* (N/lb)		5,950 / 1,338
	Vernier engines Isp. (sec)		186.7

* @ Sea-Level

Typical Atlas I Mission Profile

There are two basic types of trajectory ascent designs called "direct ascent" (single upper stage burn) and "parking orbit ascent" (multiple upper stage burns)

Direct Ascent A single upper-stage burn, or direct ascent trajectory design, is most appropriate for payloads being delivered to low earth orbits or elliptical transfer orbits with low perigee altitude. Direct ascents with little or no yaw steering can achieve orbit inclinations in the range of 28 to 34 degrees by launching at azimuths between 90 and 108 degrees. Table 2.3.2 illustrates the steps for the direct ascent sequence of events [6].

Table 2.3.2 Direct Ascent Sequence of Events

Events	Time (sec)
Liftoff	0.0
Booster engine cutof @ 5.5 g's (BECO)	169
Jettison booster package (BECO + 3.1)	172
Jettison payload fairing	205
Sustainer engine cutoff (SECO)	288
Atlas/Centaur separation (SECO + 2.0)	290
Upper stage main engine start (MES)	301
Upper stage main engine cutoff (MECO)	774
Start hydrazine settling (MECO + .06)	774
Orientation for separation and spin (MECO + 2.0)	776
End hydrazine settling (MECO + 30)	
Separate spacecraft (MECO + 135)	804
Start collision & contamination	909
avoidance maneuver (MECO + 150)	924

Parking Orbit Ascent The parking orbit ascent, used primarily for geostationary missions, is the most common Atlas mission design. Two upper stage burns generally are sufficient to maximize performance and satisfy mission requirements using this ascent mode. Tables 2.3.3-2.3.4 illustrates the parking orbit ascent sequence of events and mission capabilities of Atlas I [6]. The performance shown in table 2.3.4 is for system weight of spacecraft, payload adapter, and mission-peculiar hardware. Typical payload and mission-peculiar hardware weight is 60 kg (130 lb).

ATLAS II

The Atlas II vehicle is an uprated version of the General Dynamics Atlas I launch vehicle, with capabilities for delivering heavier payloads to orbit. The Atlas II buils on the Atlas I configuration to provide increased performance capability. Atlas upgrades include increased booster engine performance and a 9-foot (2.7 m) lengthening of the propellant tanks. Atlas propulsion is provided by the uprated Rocketdyne MA-5A engine system, which develops a total sea level rated thrust of 474,500 lbs (2109 KN). In addition, the vernier engines which are used in Atlas I for roll control are replaced with a hydrazine roll control system. Centaur performance upgrades include a 3-foot (.09 m) longer tank and a change in engine mixture ratio from 5.0:1 to 5.5:1. The current jettisonable fiberglass Centaur insulation panels are replaced with a fixed foam insulation system. Table 2.3.5 summarizes the characteristics of the Atlas II launch vehicle [6].

Table 2.3.3 Parking Orbit Ascent Sequence of Events

Events	Time (sec)
Liftoff	0.0
Booster engine cutof @ 5.5 g's (BECO)	169
Jettison booster package (BECO + 3.1)	172
Jettison payload fairing	227
Sustainer engine cutoff (SECO)	285
Atlas/Centaur separation (SECO + 2.0)	287
Upper stage 1st main engine start (MES1)	298
Upper stage 1st main engine cutoff (MECO1)	670
Upper stage 2nd main engine start (MES2)	1470
Upper stage 2nd main engine cutoff (MECO2)	1578
Start hydrazine settling (MECO2 + .06)	1578
Orientation for separation and spin (MECO2 + 2.0)	1580
End hydrazine settling (MECO2 + 30)	
Separate spacecraft (MECO2 + 135)	1608
Start collision & contamination	1713
avoidance maneuver (MECO2 + 150)	1728

Table 2.3.4 Atlas I Mission Capabilities

Typical Missions	Medium Fairing (kg/lb)	Large Fairing (kg/lb)
Geosynchronous transfer orbit: 167 x 35,788 km (90 x 19,324 nmi) 28.5-deg transfer orbit inclination	2,340 / 5,150	2,250 / 4,950
Low earth orbit: 185 x 185 km (100 x 100 nmi) 28.5-deg orbit inclination	5,900 / 13,000	5,700 / 12,550
Earth escape: C3 = 0	1,520 / 3,350	1,400 / 3,100

Table 2.3.5 Atlas II Vehicle Characteristics

Overall Vehicle	Length (m/ft)	With medium fairing	45.6 / 150
		With large fairing	47.5 / 156
	Diameter (m/ft)		3.1 / 10
	Gross liftoff Weight	With medium fairing (kg/lb)	187,170 / 412,600
		With large fairing (kg/lb)	187,56 0 / 413,500
Centaur	Length (m/ft)		10.1 / 33
	Propellant		LH_2 & LO_2
	Propellant Weight (kg/lb)		16,780 / 37,000
	Total thrust$_{vacuum}$ (KN/lb)		148.1 / 33,300
	Specific Isp$_{vacuum}$ (sec)		442.3
Atlas	Length (m/ft)		24.9 / 82
	Propellant		LO_2 & RP-1
	Propellant Weight (kg/lb		156,260 / 344,500
	Booster engines total thrust* (KN/lb)		1,815 / 408,000
	Booster engines Isp* (sec)		263.1
	Sustainer engine thrust* (KN/lb)		269 / 60,500
	Sustainer engine Isp* (sec)		220.4

* @ Sea-Level

Mission Profile for Atlas II

Atlas II flight profile is basically the same as those for Atlas I except the staging times are slightly later due to increased propellant weights. Atlas II can perform a two-burn or parking orbit ascent trajectory to achieve circular orbits or a one-burn upper-stage direct ascent to circular orbits. Tables 2.3.6-2.3.7 show the Atlas II nominal sequence of events and mission capabilities [7]. The performance shown in table 2.3.7 is for system weight of spacecraft, payload adapter, and mission-peculiar hardware. Typical payload and mission-pecu-liar hardware weight is 60 kg (130 lb).

ATLAS IIA

Atlas IIA is an uprated version of Atlas II. Everything that was said about Atlas II is true for Atlas IIA except that the Centaur upper stage for Atlas IIA was totally modified.

Centaur For Atlas IIA:

The Centaur upper stage utilizes two uprated Pratt & Whitney RL10A-4 engines rated at 20,500 lb (91.1 KN) thrust each without an extendible nozzle, and at 20,800 lbf (92.5 KN) with an extendible nozzle. The engine is a gimbaled, turbo-pump-fed, regeneratively cooled, single-chamber rocket engine consisting of a fixed primary nozzle and an optional secondary extendible nozzle. Without the nozzle, it delivers a nominal specific impulse of 443.0 seconds and has an operating life of 3,000 seconds. With the nozzle, it delivers a nominal

Table 2.3.6 Parking Orbit Ascent Sequence of Events

Events	Time (sec)
Liftoff	0.0
Booster engine cutof @ 5.5 g's (BECO)	172.4
Jettison booster package (BECO + 3.1)	175.5
Jettison payload fairing	226
Sustainer engine cutoff (SECO)	278.2
Atlas sustainer jettison	280.2
Upper stage 1st main engine start (MES1)	290.7
Upper stage 1st main engine cutoff (MECO1)	673.3
Upper stage 2nd main engine start (MES2)	1472.5
Upper stage 2nd main engine cutoff (MECO2)	1577.3
Orientation for separation and spin	1579.3
Separate spacecraft	1712.3
Start collision & contamination avoidance maneuver	2312.3
Satellite first apogee arrival	5.7 hr.

Table 2.3.7 Atlas II Mission Capabilities

Typical Missions	Medium Fairing (kg/lb)	Large Fairing (kg/lb)
Geosynchronous transfer orbit: 167 x 35,788 km (90 x 19,324 nmi) 28.5-deg transfer orbit inclination	2,770 / 6,100	2,680 / 5,900
Low earth orbit: 185 x 185 km (100 x 100 nmi) 28.5-deg orbit inclination	6,780 / 14,950	6,580 / 14,500
Earth escape: C3 = 0	1,940 / 4,270	1,820 / 4,020

specific impulse of 449.5 seconds and has an operating life of 2,000 seconds. In both cases, the engine has multiple start capability [6].

Mission Profile for Atlas IIA

Atlas IIA flight profile is basically the same as those for Atlas II except the staging times are slightly later due to increased propellant weights. Atlas IIA can perform a two-burn or parking orbit ascent trajectory to achieve circular orbits or a one-burn upper-stage direct ascent to circular orbits. Tables 2.3.8-2.3.10 show the Atlas IIA characteristics, nominal sequence of events and mission capabilities [6].

The performance shown in table 2.3.9 is for system weight of spacecraft, payload adapter, and mission-peculiar hardware. Typical payload and mission-peculiar hardware weight is 60 kg (130 lb).

ATLAS IIAS

The Atlas IIAS vehicle is an uprated version of the General Dynamics Atlas IIA launch vehicle, with capabilities for meeting even heavier payload requirements. The main changes are Atlas tanks beef-up, thrust section redesign/strengthening, and attachment/avionic provisions for the four Castor IVAs.

Solid Rocket Booster (SRB) Addition:

The four SRBs used are Thiokol's Castor IVA. The Castor IV is an "off-the-shelf" motor that has been successfully

Table 2.3.8 Atlas IIA Vehicle Characteristics

Overall Vehicle	Length (m/ft)	With medium fairing	45.6 / 150
		With large fairing	47.5 / 156
	Diameter (m/ft)		3.1 / 10
	Gross liftoff Weight	With medium fairing (kg/lb)	187,310 / 412,900
		With large fairing (kg/lb)	187,700 / 413,800
Centaur	Length (m/ft)		10.1 / 33
	Propellant		LH_2 & LO_2
	Propellant Weight (kg/lb)		16,780 / 37,000
	Total thrust, $_{vacuum}$ (KN/lb)		180.2 / 40,500
	Specific Isp, $_{vacuum}$ (sec)		448.9
Atlas	Length (m/ft)		24.9 / 82
	Propellant		LO_2 & RP-1
	Propellant Weight (kg/lb		156,260 / 344,500
	Booster engines total thrust* (KN/lb)		1,815 / 408,000
	Booster engines Isp* (sec)		263.1
	Sustainer engine thrust* (KN/lb)		269 / 60,500
	Sustainer engine Isp* (sec)		220.4

* @ Sea-Level

Table 2.3.9 Parking Orbit Ascent Sequence of Events

Events	Time (sec)
Liftoff	0.0
Booster engine cutof @ 5.5 g's (BECO)	168.7
Jettison booster package (BECO + 3.1)	171.8
Jettison payload fairing	232.4
Sustainer engine cutoff (SECO)	278.9
Atlas sustainer jettison	280.9
Upper stage 1st main engine start (MES1)	291.4
Upper stage 1st main engine cutoff (MECO1)	595.1
Upper stage 2nd main engine start (MES2)	1449.4
Upper stage 2nd main engine cutoff (MECO2)	1539.5
Orientation for separation and spin	1541.5
Separate spacecraft	1674.5
Start collision & contamination avoidance maneuver	2274.5
Satellite first apogee arrival	5.7 hr.

Table 2.3.10 Atlas IIA Mission Capabilities

Typical Missions	Medium Fairing (kg/lb)	Large Fairing (kg/lb)
Geosynchronous transfer orbit: 167 x 35,788 km (90 x 19,324 nmi) 28.5-deg transfer orbit inclination	2,900 / 6,400	2,810 / 6,200
Low earth orbit: 185 x 185 km (100 x 100 nmi) 28.5-deg orbit inclination	7,120 / 15,700	6,920 / 15,250
Earth escape: C3 = 0	2,100 / 4,620	1,980 / 4,370

flight proven 117 times on the Delta II launch vehicle. Each Castor IVA SRB is 37 feet long, 40 inches in diameter, weights 25,500 pounds, and provides a sea level average thrust of 97,500 pounds over a 53 seconds burn time. Tables 2.3.11-2.3.13 present all the features of this configuration [5,6].

The performance shown in table 2.3.12 is for system weight of spacecraft, payload adapter, and mission-peculiar hardware. Typical payload and mission-peculiar hardware weight is 60 kg (130 lb).

Centaur_ The Atlas IIAS upper stage is identical to the Centaur flown on Atlas IIA.

Mission Profile for Atlas IIAS

Atlas IIAS flight profile is essentially the same as for Atlas IIA except that four Castor IVA SRBs are used to provide added thrust. Two SRBs are ignited on the ground, while ignition of the second pair is delayed until burnout of the first two. This ignition sequence is to provide sufficient thrust/ weight for liftoffwhile minimizing the dynamic pressure the vehicle experience during the ascent phase. The Atlas IIAs can perform a two-burn or parking orbit ascent for circular orbits or a single upper-stage burn direct ascent to circular orbits.

Table 2.3.11 Atlas IIAS Vehicle Characteristics

Overall Vehicle	Length (m/ft)	With medium fairing	45.6 / 150
		With large fairing	47.5 / 156
	Diameter (m/ft)		3.1 / 10
	Gross liftoff Weight	With medium fairing (kg/lb)	233,600 / 515,000
		With large fairing (kg/lb)	234,010 / 515,900
Centaur	Length (m/ft)		10.1 / 33
	Propellant		LH_2 & LO_2
	Propellant Weight (kg/lb)		16,780 / 37,000
	Total thrust$_{,vacuum}$ (KN/lb)		180.2 / 40,500
	Specific Isp$_{,vacuum}$ (sec)		448.9
Atlas	Length (m/ft)		24.9 / 82
	Propellant		LO_2 & RP-1
	Propellant Weight (kg/lb		156,260 / 344,500
	Booster engines total thrust* (KN/lb)		1,815 / 408,000
	Booster engines Isp* (sec)		263.1
	Sustainer engine thrust* (KN/lb)		269 / 60,500
	Sustainer engine Isp* (sec)		220.4

Solid Rocket Motors (4 Used)	Propellant	HTPB
	Propellant Weight per motor	10,230 / 22,560
	Average thrust per motor	433 KN / 97,520
	Specific Impulse*	229 sec.

* @ sea level

Table 2.3.12 Parking Orbit Ascent Sequence of Events

Events	Time (sec)
Liftoff/SRM ignition (first pair)	0.0
SRM burnout (first pair)	56.0
SRM ignition (second pair)	66.2
SRM jettison (first pair)	91.7
SRM burnout (second pair)	122.5
SRM jettison (second pair)	126.2
Booster engine cutof @ 5.5 g's (BECO)	167.8
Jettison booster package (BECO + 3.1)	170.9
Jettison payload fairing	217.5
Sustainer engine cutoff (SECO)	283.4
Atlas sustainer jettison	285.4
Upper stage 1st main engine start (MES1)	295.9
Upper stage 1st main engine cutoff (MECO1)	588.1
Upper stage 2nd main engine start (MES2)	1420.4
Upper stage 2nd main engine cutoff (MECO2)	1521.1
Orientation for separation and spin	1523.1
Separate spacecraft	1656.1
Start collision & contamination avoidance maneuver	2256.1
Satellite first apogee arrival	5.7 hr.

Table 2.3.13 Atlas IIAS Mission Capabilities

Typical Missions	Medium Fairing (kg/lb)	Large Fairing (kg/lb)
Geosynchronous transfer orbit: 167 x 35,788 km (90 x 19,324 nmi) 28.5-deg transfer orbit inclination	3,630 / 8,000	3,490 / 7,700
Low earth orbit: 185 x 185 km (100 x 100 nmi) 28.5-deg orbit inclination	8,610 / 19,000	8,390 / 18,500
Earth escape: C3 = 0	2,670 / 5,890	2,550 / 5,640

2.4 Titan IV

BACKGROUND ON THE VEHICLE

Thirty years ago, beginning with the Titan I Intercontinental Ballistic Missile, the Titan heritage was begun. From that start in 1959, a steady progression of successful missiles and launch vehicles has grown into a family of rockets which have earned Titan the reputation of "the world's most reliable launch vehicle" [8]. Titan launch vehicles have been contributing to the United States of America national space objectives for more than 25 years. As the American space program has grown, the Titan family has expanded to meet the changing requirements. The dependability and versatility of Titan vehicle have been demonstrated by their selection for missions ranging from strategic intercontinental ballistic missile weapon systems, to the manned Gemini space flights, to NASA interplanetary missions, to Viking Mars missions, to critical national security programs [9].

The Titan IV is designed to complement the National Space Transportation System (shuttle, orbiter). It is an independent launch vehicle system to assist in assuring Department Of Defense access to space.

The Titan IV is essentially a derivative of the 34D with structural extensions and other modifications to the core vehicle design to accommodate two seven-segment SRMs or two three-segment Solid Rocket Motor Upgrades (SRMUs). The Titan

IV provides STS-equivalent and greater payload-to-orbit performance, thus enabling it to meet DOD unique requirements.

The major elements that comprise the Titan IV launch vehicle are: the Titan IV core with its two liquid fueled stages, engines and forward skirts; the SRMs; the PLFS; and the Upper Stages and No Upper Stage (NUS) [9].

Titan IV Stage I/II Propulsion System

Stage I Engine

The Booster vehicle has two Liquid Rocket Engine (LRE) assemblies, one for each stage. The Stage I engine Aerojet LR87-AJ-11 two subassemblies that operate simultaneously and independently, but under a single/common control system. Each assembly contains fuel and oxidizer pumps that are geared together and are turbine driven. The turbine is powered by a gas generator which uses the same liquid propellants as the main combustion chambers. Hydraulic actuators, in response to vehicle guidance commands, vector the thrust of the two combustion chambers to provide pitch, yaw and roll control [9].

Stage II Engine

The Stage II engine, Aerojet LR91-AJ-11, operates in the same manner as the Stage I engine, except that it has only one combustion chamber. This chamber is hydraulically vectored to provide pitch and yaw control.

Propellants

Both Stage I and Stage II engines use liquid hypergolic propellants that can remain aboard in a launch ready state for extended periods of time. The fuel (Aerozine-50) is nomimally a 50-50 weight mixture of hydrazine and unsymmetrical dimethylhydrazine (UDMH) and the oxidizer is nitrogen tetroxide (N_2O_4). these propellants are storable at ambient temperature and pressure, thus eliminating temperature conditioning equipment such as is required when handling cryogenic propellants. The hypergolic property (spontaneous ignition upon fuel oxidizer contact) eliminates the need for an ignition system and related checkout and support equipment. Tables 2.4.1 - 2.4.7 summarize all the features of the Titan IV vehicles [9].

Payload Fairing

The PLF is available in lengths from 56 ft to 86 ft in 10 ft increments and is 16.66 ft in outside diameter. Useable internal diameter for payloads is 14.99 ft. Each payload fairing is built to accommodate specific satellite vehicle requirements.

Centaur Upper stage

The Centaur G Prime vehicle overall length with the engines is 29.45 ft and its largest diameter is 14.17 ft. The propulsion is supplied from a single-stage, liquid-fueled, dual engine system. The Centaur is powered by two Pratt and whitney RL 10A-3-3A engines. The propellant feed and main engine system provides the Centaur vehicle with the thrust generated by the combustion of LO_2 and LH_2 propellants. The engines are capable of multiple restarts and are capable of fulfilling the requirement of a variety of missions.

Table 2.4.1 Titan IV with 7 Segment Solid rocket Motors Performance Capability:

Titan IV Type I (7 SRMs)			**Payload Capability (lb)**	
Reference Launch	Vehicle Description	Final Mission	Minimum Guaranteed	Current Status
ESMC	TIV/Centaur TIV/Centaur TIV/Centaur	GSO 12-hr inclined 24-hr inclined	10,000 11,500 N/A	10,030 15,270 10,272
ESMC	TIV/IUS	GSO	38,780	5,078
WSMC	TIV/NUS	100 X 100 x 90	30,100	30,892
ESMC	TIV/NUS	80 x 95 x 28.6	39,100	38,803

Table 2.4.2 Titan IV Type II With 3 Segment SRM

Titan IV type II with 3 Segment Solid Rocket Motor Upgrade			**Payload Capability (lb)**	
Reference Launch	Vehicle Description	Final Mission	Minimum Guaranteed	Today Status
ESMC	TIV/SRMU/Ct	GSO	12,700	12,597
ESMC	TIV/SRMU/NUS	80x95x28.6	47,000	47,893
WSMC	TIV/SRMU/NUS	100x100x90	38,800	38,524
ESMC	TIV/SRMU/NUS	80x95x28.6	47,800	48,247

Table 2.4.3 Solid Rocket Motors-Seven Segment Characteristics

Solid Rocket Motor Length	(ft)	114.8
Solid Rocket Motor Diameter	(ft)	10
Dry Weight $)_{each}$	(lb)	94,941
Loaded Weight $)_{each}$	(lb)	687,052
Propellant Weight $)_{each}$	(lb)	591,111
Average Vacuum Thrust	(lb)	1.394 million
Vacuum Specific Impulse	(sec.)	272.0

Table 2.4.4 Solid Rocket Motors Upgrade-Three Segment Characteristics

Solid Rocket Motor Length	(ft)	112.385
Solid Rocket Motor Diameter	(ft)	10.499
Dry Weight $)_{each}$	(lb)	81,820
Loaded Weight $)_{each}$	(lb)	770,673
Propellant Weight $)_{each}$	(lb)	688,853
Average Vacuum Thrust	(lb)	1,783,080
Vacuum Specific Impulse	(sec.)	284.33

Table 2.4.5 TITAN IV TYPE I/CENTAUR CHARACTERISTICS

VEHICLE CHARACTERISTICS		
Average Stage I Nozzle Centerline Thrust	(lbf)	544,414
Average Stage I Nozzle Centerline Isp	(sec)	301.23
Average Stage II Vacuum Thrust	(lbf)	106,224
Average Stage II Vacuum Isp	(sec)	317.7
PAYLOAD FAIRING:		
Dimensions (diameter and length)	(ft)	16.7x86
Weight	(lbm)	14,115
Average Centaur Vacuum Thrust	(lbf)	33,030
Average Centaur Vacuum Isp	(sec)	443.80
MISSION PROFILE:		
Launch Azimuth		
GSO Mission	(degree)	93.0
12-hr & 24-hr inclined Mission	(degree)	37.9

Table 2.4.6 TITAN IV TYPE I/CENTAUR TYPICAL SEQUENCE OF MISSIONS

	GSO (sec)	12-hr* (sec)	24-hr** (sec)
SRM Ignition	0.0	0.0	0.0
Roll to Flight Azimuth	6.0	6.0	6.0
End of roll Maneuver	9.0	12.0	12.0
Begin Pitch Maneuver	10.0	14.0	14.0
Maximum Dynamic Pressure	57.0	57.0	57.0
Stage I Ignition	115.0	115.9	115.9
SRM Separation	126.0	126.0	126.0
Jettison Payload Fairing	233.3	237.7	232.7
Step 1/Stage II Separation	304.0	304.1	304.0
Step 2/Centaur Separation	541.8	541.8	541.8
Centaur First Burn Ignition (MES1)***	553.8	554.8	553.8
End Centaur First Burn (MECO1)****	767.9	788.4	789.5
Park Orbit Injection	1017.9	793.4	794.5
Centaur Second Burn Ignition (MES2)	1412.3	3582.2	2218.2
End Centaur Second Burn (MECO2)	1678.8	3839.6	2477.8
Centaur Third Burn Ignition (MES3)	20376.9	7824.9	20999.6
End Centaur Third Burn (MECO3)	20498.1	7874.0	21100.8
Final Orbit Injection	20528.1	8118.0	21130.

* 12-Hour Elliptical Inclined Orbit

** 24-Hour Circular Inclined Orbit

*** Main Engine Stard

**** Main Engine Cutoff

Table 2.4.7 TITAN IV TYPE I/CENTAUR TYPICAL PERFORMANCE

	GSO	12-HOUR	24-HOUR
Payload Weight (lbm)	10,030.0	15,270.0	10,272.0
Mission Required Propellant Margin (lbm)	602.0	750.0	643.0
Launch Azimuth (deg)	93.0	37.0	37.9
Park Orbit:			
Perigee, nmi	85.0	98.0	86.0
Apogee, nmi	250.0	101.0	156.0
Inclination, (deg)	28.6	55.0	55.0
Transfer Orbit:			
Perigee, nmi	96.0	112.0	110.0
Apogee, nmi	19,413.0	17,488.0	19,396.0
Inclination, (deg)	26.6	55.0	55.6
Final Orbit:			
Perigee, nmi	19,323.0	509.0	19,323.0
Apogee, nmi	19,323.0	21,298.0	19,323.0
Inclination, (deg)	0.0	63.4	65.0
Argument of perigee, (degree)	N/A	270.0	N/A

TITAN IV/INERTIAL UPPER STAGE (IUS)

The Inertial Upper Stage (IUS) is designed and produced by the Boeing Aerospace Company. The IUS vehicle is 17.0 feet long, 7.6 feet in diameter in the cylindrical section, flaring to 9.5 feet at the forward end. It consists of an aft skirt, a relatively large First Stage Rocket Motor (SRM-1), an interstage, a smaller second stage Solid Rocket Motor (SRM-2) and an Equipment Support Section (ESS) that interfaces with the Spacecraft.

The IUS propulsion system consists of one large and one small United Technologies SRM and an Reaction Control System (RCS). The RCS is a mono-propellant hydrazine system which uses blowdown pressurization. Its activation occurs approximatively 30 secondes after IUS separation from Titan. Its operational requirements include: orienting the vehicle prior to SRM firing, roll control during the SRM firings, vernier corrections for motor impulse variations, attitude control and maneuvering during transfer orbit coast period. Tables 2.4.8-2.4.10 summarize the features of the Titan IV Type I/IUS configuration [9].

TITAN IV/NUS (No UPPER Stage):

The Titan IV/NUS system utilizes the basic Titan IV/Core Vehicle, appropriate skirts, PLF, SRMs, associated facilities and Ground Support Equipment for the purpose of placing a spacecraft into low earth orbit missions from VAFB Space Launch Complex. Tables 2.4.11 - 2.4.16 show all the features [9].

Table 2.4.8 TITAN IV TYPE I/IUS CHARACTERISTICS

VEHICLE CHARACTERISTICS		
Average Stage I Nozzle Centerline Thrust	(lbf)	546,267
Average Stage I Nozzle Centerline Isp	(sec)	301.18
Average Stage II Vacuum Thrust	(lbf)	106,392
Average Stage II Vacuum Isp	(sec)	317.66
PAYLOAD FAIRING:		
Dimensions (diameter and length)	(ft)	16.7 x 56
Weight	(lbm)	8,884
INETIAL UPPER STAGE:		
Stage I Average Vacuum Thrust	(lbf)	41,780
Stage I Average Effective Isp	(sec)	294.1
Stage II Average Vacuum Thrust	(lbf)	17,222
Stage II Average Effective ISP	(sec)	302.2
MISSION PROFILE:		
Launch Azimuth		
GSO Mission	(degree)	93.0

Table 2.4.9 TITAN IV TYPE I/IUS TYPICAL SEQUENCE OF MISSIONS

	GSO MISSION (sec)
SRM Ignition	0.0
Begin Roll to Flight Azimuth	6.0
End of roll Maneuver	9.0
Begin Pitch Over	10.0
Maximum Dynamic Pressure	56.3
Stage I Ignition	116.3
SRM Separation	126.2
Jettison Payload Fairing	229.7
Step 1/Stage II Separation	302.4
Step 1/IUS Separation, Park Orbit Inject	534.9
IUS First Stage Ignition	4,137.5
IUS First Stage Shutdown, Begin Tailoff	4,290.2
End Tailoff, Transfer Orbit Inject	4,646.6
IUS Second Stage Ignition	23,068.7
IUS Second Stage Shutdown, Begin Tailoff	23,178.1
Final Orbit Inject	23,582.8
End of Mission	26,000.0

Table 2.4.10 TITAN IV TYPE I/IUS TYPICAL PERFORMANCE

	TYPICAL PAYLOAD CAPABILITY (lbm)	
	Park Orbit Inject at Perigee	Park Orbit Inject Off-Perigee
GSO Mission (First Crossing)	5,201	5,201
GSO Mission (Second Crossing)	5,077	5,152

NOTE: Payload Weight mentioned in table 2.4.10 includes the spacecraft adapter.

Table 2.4.11 TITAN IV TYPE I/NO UPPER STAGE CHARACTERISTICS

VEHICLE CHARACTERISTICS		
Average Stage I Nozzle Centerline Thrust	(lbf)	545,894
Average Stage I Nozzle Centerline Isp	(sec)	300.4
Average Stage II Vacuum Thrust	(lbf)	106,312
Average Stage II Vacuum Isp	(sec)	317.5
PAYLOAD FAIRING:		
Dimensions (diameter and length)	(ft)	16.7 x 56
Weight	(lbm)	8,884
MISSION PROFILE:		
Launch Azimuth	(degree)	182
Inclination	(degree)	90.0

Table 2.4.12 TITAN IV TYPE I/NUS Sequence of Events From WSMC

	LOW EARTH ORBIT MISSION (sec)
SRM Ignition	0.0
Begin Roll to Launch Azimuth	6.0
End Roll Maneuver	9.0
Begin Pitch Maneuver	10.0
Maximum Dynamic Pressure	55.2
Stage I Ignition	117.9
SRM Separation	127.8
Jettison Payload Fairing	234.0
Step I/Stage II Separation	307.0
Step II/NUS Separation, Orbit Inject	543.1

Table 2.4.13 TITAN IV TYPE I/NUS TYPICAL PERFORMANCE FOR A WSMC LAUNCH

	TYPICAL PAYLOAD CAPABILITY (lbm)		
Orbit (nmi x nmi)	99 Degree Orbital Inclination	90 Degree Orbital Inclination	63.5 Degree Orbital Inclination
100 x 100	30,000	31,200	34,900
200 x 200	24,000	25,000	27,500
300 x 300	16,000	16,600	19,000

TITAN IV TYPE I/NUS (WITH KICK MOTOR FOR FINAL CIRCULARIZATION) TYPICAL PERFORMANCE	
	PAYLOAD CAPABILITY (lbm)
12-hour period elliptical orbit inclined 63.5 degrees and with 270 degrees Argument of perigee: (180 nmi perigee and 21,616 nm apogee)	12,300*
12-hour period circular orbit (10,898 nmi apogee and perigee inclined 55.0 degrees)	7,000**

* The Upper Stage kick motor required for this mission has a thrust of 15,000 lbf, a specific impulse of 290 sec and a mass fration (propellant weight divided by total kick motor weight) equal to 0.9. The final orbit P/L weight is 12,300 lbm.

** The Upper Stage kick motor required for this mission has a thrust of 15,000 lbf, a specific impulse of 290 sec and a mass fraction equal to 0.9 (propellant weight of 5415 lbm and a jettisonable case weight of 602 lbm).

Table 2.4.14 TITAN IV TYPE I/NUS CHARACTERISTICS FOR A ESMC LAUNCH

VEHICLE CHARACTERISTICS		
Average Stage I Nozzle Centerline Thrust	(lbf)	547,005
Average Stage I Nozzle Centerline Isp	(sec)	301.16
Average Stage II Vacuum Thrust	(lbf)	106,460
Average Stage II Vacuum Isp	(sec)	317.7
PAYLOAD FAIRING:		
Dimensions (diameter and length)	(ft)	16.7 x 56
Weight	(lbm)	8,950
MISSION PROFILE:		
Launch Azimuth	(degree)	93

Table 2.4.15 TITAN IV TYPE I/NUS FOR A ESMC LAUNCH SEQUENCE OF EVENTS

	LOW EARTH MISSION (sec)
SRM Ignition	0.0
Begin Roll to Launch Azimuth	6.0
End Roll Maneuver	9.0
Begin Pitch Maneuver	10.0
Maximum Dynamic Pressure	56.4
Stage I Ignition	116.5
SRM Separation	126.4
Jettison Payload Fairing	231.0
Step 1/Stage II Separation	302.0
Step 2/NUS Separation, Orbit Inject	533.7

Table 2.4.16 TITAN IV TYPE I/NUS TYPICAL PERFORMANCE FOR A ESMC LAUNCH

	TYPICAL PAYLOAD CAPABILITY (lbm)	
Orbit (nmi x nmi)	55 Degree Orbital Inclination	28.6 Degree Orbital Inclination
100 x 100	36,000	38,500
200 x 200	28,500	29,500
300 x 300	19,500	19,600

NOTE: These payload capabilities in table 2.4.16 are for the 16.7 x 56 ft (9,950 lbm) payload fairing.

2.5 TITAN II

BACKGROUND ON THE VEHICLE

The Titan II was Martin Marietta's second ICBM program. Development began in 1960 with the first launch in March 1962. Titan II was the first strategic missile that used storable hypergolic propellants and an inertial guidance system. This weapon system was deployed in 1962 with deactivation completed in 1986. Of the 163 Titan built, 81, were launched for development and training purposes [10].

The Titan II ICBM was also converted into the Titan/Gemini space launch vehicle by manrating its critical systems. The success of this program was a significant milestone in the evolution of the Apollo spaceflight program. Twelve successful Gemini launches occurred between April 1964 and November 1966 [10].

The Titan II ICBM weapon system is being deactivated by the U.S. Government and is undergoing conversion into a fleet of space launch vehicles. The goal of this program is to maximixe the use of Titan II weapon system materials and to minimize launch site modification.

A total of 55 Titan II ICBMs is available to converted into space launch vehicles as part of this program. Forty-two vehicles are currently available (as of Jan 1991). All refurbishment activities are performed at Martin Marietta Denver Aerospace facilities [10].

The Titan II Space Launch Vehicle (SLV) configuration consists of the same two-stage liquid-fueled booster used for the ICBM configuration. A payload fairing is added which is 10 feet in diameter and currently available in lengths of 20, 25 and 30 feet. Longer payload fairings are potentially available in five-foot increments up to 50 feet long. The ICBM vernier motors on Stage II are removed and replaced with retrorocket motors to assit in payload separation.

STAGE I AND II PROPULSION SYSTEMS

Both Stages I and II use storable liquid hypergolic (combust spontaneously when mixed) propellants that can remain aboard in launch-ready state for extended periods of time. The fuel, Aerozine 50 (A50), is a blend of 50 percent hydrazine and 50 percent unsymmetrical dimethylhydrazine (UDMH). The oxidizer used is nitrogen tetroxide (N_2O_4). The liquid rocket engines are hydraulically gimbaled and fed by a turbine pump. The Stage I engine has two subassemblies that provide pitch, yaw, and roll control. The Stage II engine has one subassembly that provides pitch and yaw control. Stage II roll control is provided by ducting turbine exhaust through a roll control nozzle that is swiveled. Each subassembly has a regeneratively cooled thrust chamber, gas generator start cartridges and turbine pumps. The Stage II engine has an ablative nozzle extension [10].

The Stage II retro rocket configuration consists of four retros. The retro rockets consist of a solid propellant which, when activated, burns for 2.92 seconds producing a delta velocity approximately 10 feet per second away from the Titan II/ payload interface. Tables 2.5.1 - 2.5.5 present the features of both Titan II and Titan II SLV configurations [10].

PERFORMANCE CAPABILITIES

All the performance capabilities data presented in the tables here are for nominal conditions for the Western Space and Missile Center. Also, all the data are based on a 20-foot paylod fairing. Currently, the Western Space and Missile Center (WSMC) is the only launch site scheduled for Titan II SLV launches. Titan II SLV can deliver payloads to ballistic, circular, or elliptical polar orbits.

Table 2.5.1 TITAN II SLV CHARACTERISTICS

System Height	Up to 140.8 feet
Payload Fairing Height	Up to 30 feet
Payload Fairing Diameter	10 feet
STAGE II	
Height	40.1 feet
Diameter	10 feet
Thrust	100,000 lb (Vacm)
Burn Time	182 seconds
STAGE I	
Height	70.7 feet
Diameter	10 feet
Thrust	474,000 lb (Vacm)
Burn Time	
GUIDANCE : Inertial Guidance System	

Table 2.5.2 TYPICAL LOW EARTH ORBIT MISSION PROFILE

Low Earth Mission Sequence of Events (with ACS*)			
EVENTS	**TIME (sec)**	**ALTITUDE (nmi)**	**VELOCITY (fps)**
Go Inertial	- 5.0	0.0	
Stage I Ignition	- 2.0	0.0	
Liftoff	0.0	0.0	
Roll To Flight Azimuth	9.0	____	
End Stage I Roll Maneuver	19.0	____	
Stage I End Steady State	156.0	____	
Stage I/II Separation	157.7	35.0	8865
Payload Fairing Separation	210.9	64.0	11123
Stage II Shutdown	332.8	102.0	25524
Begin Velocity Trim	378.0	____	
End Velocity Trim	406.0	____	
Start Maneuver to SV Sep.	434.0	____	
SV Separation	519.0	100.0	25677

* Attitude Control System (ACS), if ACS is used, it can performed the following functions. After Stage II shutdown: recover from attitude transients associated with shutdown, velocity trim burn, and reorient to payload separation attitude. When the ACS is used, Stage II is separated and maneuvered from the payload sometime after Stage II shutdown by use of retro rockets.

Table 2.5.3 TYPICAL SUN SYNCHRONOUS MISSION PROFILE

Sun-Synchronous Sequence of Events (without ACS*)			
EVENTS	**TIME (sec)**	**ALTITUDE (nmi)**	**VELOCITY (fps)**
Go Inertial	- 5.0	0.0	
Stage I Ignition	- 2.0	0.0	
Liftoff	0.0	0.0	
Roll To Flight Azimuth	9.0	____	
End Stage I Roll Maneuver	19.0	____	
Stage I End Steady State	156.9	____	
Stage I/II Separation	157.7	35.0	8447
Payload Fairing Separation	210.1	63.0	10611
Stage II Shutdown	324.4	134.0	22587
SV Separation	341.0	150.0	22625
AKM** Ignition	816.8	451.0	20296
AKM Burnout	860.6	458.0	24325
Begin SV Velocity Trim	870.3	____	
End SV Velocity Trim	884.7	____	
Orbit Inject	885.0	458.0	24366

* Without ACS, Stage II is separated and maneuvered from the payload by means of retro rockets approximately 16 seconds after shutdown command.

** Apogee Kick Motor (AKM), it is used to establish the desired orbit.

Table 2.5.4 PERFORMANCE CAPABILITY FOR SELECTED BALLISTIC TRAJECTORIES

INCLINATION (degree)	PERIGEE (nmi)	APOGEE (nmi)	PAYLOAD (lb)
63.4	-300	600	4540
63.4	-600	100	6570
63.4	-600	400	5770
63.4	-600	600	5150
63.4	-2000	100	16,060
63.4	-2000	400	11,720
63.4	-2000	600	9451
99*	-100	270	5660
99*	-1000	270	8580
99*	-2000	270	15,081

* Trajectories reflect capabilities of Titan II No Upper Stage with 4 Solid Rocket Motors.

Table 2.5.5 TITAN II PAYLOAD PERFORMANCE

TYPICAL MISSIONS	INCLINATION (deg)	PAYLOAD (lb)
Low Earth Orbit:	28.5	18,000*
Circular Polar Orbit:	63.4	16,000*
Polar Orbit:	90.0	7,800**
Polar Orbit:	99.0	4,200
Sun-Synchronous Orbit:	98.789	5,000

* Using Two Liquid Strap-on Rockets.

** Using 2-10 Solid Rocket Motors Castor IVa.

2.6 TAURUS

BACKGROUND ON THE VEHICLE

Taurus, being developed by Orbital Sciences Corporation for the DARPA Standard Small Launch Vehicle program and commercial applications, is a four-stage, solid-propellant, inertially guided, ground-mobile and ground-launched vehicle. This 90 ft long, 150,000 lb vehicle is capable of delivering 3,600 lb of payload to a low inclination, 150 nmi Earth orbit [11].

The Taurus launch vehicle evolved in early 1989 in response to the Defense Advanced Research Projects Agency's (DARPA) requirements for a Standard Small Launch Vehicle (SSLV). The Taurus launch vehicle was designed to meet the following requirements: full launch system ground transportability; launch from a dry concrete pad; launch system site; launch within 72 hours of a command to launch after set up; and launch of a 1000 lb payload into a 400 nmi circular polar orbit from Vandenburg AFB, CA. The Taurus launch vehicle can satisfy also Department of Defense, NASA and commercial payloads requirement using less demanding launch site and time requirements [11].

The Taurus launch vehicle uses current state-of-the-art technology throughout and achieves reliability and cost effectiveness through the high use of in-production components with demonstrated launch and missile system heritage. The launch vehicle design is based on avionics and propulsion systems proven during the Pegasus Air Launched Space Booster and Peacekeeper strategic missile program.

PROPULSION SYSTEM

Taurus is a four stage all solid propellant. The vehicle uses class 1.3 propellants in all stages for maximum safety. The propulsion system is based on proven motors currently in production for Peacekeeper (MX) and Pegasus.

First Stage

First stage propulsion is provided by a Peakeeper Stage 1 solid rocket motor. The motor is being produced by Morton Thiokol, inc. for the Air Force Peacekeeper ballistic missile program, and the Air Force has authorized production and procurement of the motor for SSLV use. The design features an omni-axis movable carbon-phenolic nozzle with a 3D carbon-carbon integral throat/entry. Thrust vector control for pitch and yaw is provided by a flexseal nozzle with a turbo-hydraulic actuation system. Motor ignition is provided by a proven first stage pyrogen igniter. The motor case uses kelvar/epoxy composite material [10].

Stage 2 to Stage 4

Stage 2 to Stage 4 propulsion system is provided by three new graphite-epoxy composite case solid-propellant rocket motors developed by Hercules. All three motors use IM7/55A graphite-epoxy composite cases with aramid-filled ethylene propylene diene monomer (EPDM) rubber internal insulator and integral skirts. Each nozzle consists of a carbon phenolic exit cone with 3D carbon-carbon integral throat/entry (ITE). The propellant is hydroxy terminated polybutadiene (HTPB) with 88% solids, and the propellant grains are designed for low burn rates [10].

Second Stage

The second stage has a core-burning grain design with the igniter mounted on the forward dome. The design includes a fixed nozzle, an aluminum wing mounting saddle, and an extended forward skirt. It also uses Government surplus, flight proven Pershing 1A jet vane assemblies in the top section of the Stage 1/Stage 2 interstage to provide thrust vector control in roll, pitch and yaw.

Third Stage

The third stage is also a core-burning design and uses a silicon elastomer flexseal nozzle and electromechanical actuators for thrust vector control (TVC).

Fourth Stage

The fourth stage motor has a head-end grain design to maximize propellant density. This motor also uses a flexseal nozzle and electromechanical TVC and employs an aft-mounted toroidal igniter. Tables 2.6.1 - 2.6.3 summarize the features of the taurus launch vehicle.

Table 2.6.1 Taurus Vehicle Characteristics

Motors Characteristics*		2nd Stage	3rd Stage	4nd Stage
Inert Weight	(lb)	2,780	800	277
Gross Weight	(lb)	30,908	7,500	2,172
Propellant Weight	(lb)	26,790	6,670	1,725
Burn Time	(sec)	72.3	71.4	64.6
Max. Pressure	(psia)	1,088	1,003	749
Avg Pressure	(psia)	797	793	637
Max Vax Thrust	(lbf)	131,244	30,912	9,065
Avg Vac Thrust	(lbf)	109,419	27,605	7,772
Vac Impulse	(lbf-sec)	7,911,000	1,971,000	502,100
Isp Vac	(sec)	295.3	295.5	291.1
Vehicle Total Length	(ft)			90.25
Vehicle Total Gross Weight	(lb)			150,000**
Payload Fairing Dimension	(ft)			4.0 x 8.0

* Performance data for stage I are classified; ** No payload;

Table 2.6.2 Taurus Typical Flight Operation Sequence

EVENTS	Time (sec)	Altitude (nmi)	Range* (nmi)
Stage 1 Ignition/Liftoff	0.0	0.0	0.0
Stage 1 Burnout	56.01	18.0	19.0
Stage 1/2 Separation	57.01	19.0	20.0
Stage 2 Ignition	57.07	19.0	20.0
Stage 2 Burnout	133.01	73.0	137.0
Fairing Separation	138.0	77.0	150.0
Stage 3 Ignition & 2/3 Sep.	143.01	80.0	162.0
Stage 3 Burnout	217.74	138.0	376.0
Stage 4 Ignition & 3/4 sep.	817.74	399.0	2260.0
Stage 4 Burnout & Orbit Insertion	883.23	400.0	2476.0
Payload Separation (Typical)	900.0	400.0	

* Distance Downrange

Table 2.6.3 TAURUS MISSION CAPABILITIES

TYPICAL MISSIONS	PAYLOAD (lb)	INCLINATION (deg)
Circular Polar Orbit	1,000	90
Geosynchronous Tranfer Orbit (ETR)*	830	28.5
Molnyia Orbit (WTR)**	515	90

* Eastern Test Range

** Western Test Range

2.7 PEGASUS

BACKGROUND ON THE VEHICLE

The historical motivation for the development of the air-launch Pegasus booster were purely economical. The goal was to provide a dedicated launch for a "useful" payload at the lowest possible total mission cost, rather than achieve a certain payload capacity or cost per pound. Pegasus incorporates state-of-the-art technologies in the first all new U.S. launch vehicle design since the 1970's.

The Pegasus Air-Launch Space Booster is a new launch vehicle developed in a privately-funded joint venture by Orbital Sciencee Corporation (OSC) and Hercules Aerospace Company. Pegasus is a three-stage, solid-propellant, inertially-guided, all-composite, winged launch vehicle. The vehicle is approximately 50 ft long and 50 in. in diameter, and has a gross weight (excluding payload) of approximately 41,000 lb. It is carried aloft by a conventional transport-or bomber-class aircraft to level-flight conditions of approximately 40,000 ft and Mach 0.8 [12]. After release from the aircraft and ignition of the Stage 1 motor, the autonomous flight contro system provides all guidance necessary to produce a wide range of suborbital and orbital trajectories. The Pegasus vehicle is available for both government and commercial users. Pegasus made its first flight on April 5, 1990.

SYSTEM DESCRIPTION

The Pegasus launch system comprises four elements: flight vehicle, carrier aircraft, airborne support equipment (ASE), and ground support equipment (GSE) and facilities [13].

PEGASUS FLIGHT VEHICLE

The vehicle consists of three graphite-epoxy composite case solid-propellant rocket motors, a fixed delta planform composite wing, an aft skirt assembly including three composite control fins, an avionics section forward of the Stage 3 motor, and a two-piece composite payload fairing.

Pegasus main propulsion is provided by three new graphite-epoxy composite case solid-propellant rocket motors developed by Hercules. All three motors use IM7/55A graphite-epoxy composite cases with aramid-filled ethylene propylene diene monomer (EPDM) rubber internal insulator and integral skirts. Each nozzle consists of a carbon phenolic exit cone with 3-D carbon-carbon integral throat/entry (ITE). The propellant is hydroxy terminated polybutadiene (HTPB) with 88% solids, and the propellant grains are designed for low burn rates. The Stage 1 motor has a core-burning grain design with the igniter mounted on the forward dome. The design includes a fixed nozzle, an aluminum wing mounting saddle, and an extended forward skirt. The Stage 2 motor is also a core-burning design and uses a silicon elastomer flexseal nozzle and electromechanical actuators for thrust vector control (TVC). The Stage 3 motor has a head-end grain design to maximize propellant density.

This motor also uses a flexseal nozzle and electromechanical TVC and employs an aft-mounted toroidal igniter [13]. The specific impulse of each stage (most particularly the first stage) benefits from airborne launch in two ways. The lower outside static pressure at 40,000 ft (approximately one quarter of sea level pressure) yields a higher specific impulse for a given nozzle expansion ratio. In addition, the expansion ratio of the first stage motor (40:1) has been chosen to optimize performance over the altitude range of the first stage burn (from 40,000 ft to 200,000 ft). The resulting improvement in specific impulse relative to a ground-launched first stage with a typical 8:1 expansion ratio nozzle is better than 20 secondes [11]. Tables 2.7.1 - 2.7.3 show all the relevant features of the pegasus launch vehicle.

Table 2.7.1 Pegasus Vehicle Characteristics

Vehicle Total Length (ft)	50.0
Vehicle Diameter (ft)	4.166
Vehicle Total Gross Weight (lb)	41,000*
Vehicle Dropping Altitude (ft)	40,000
Payload Fairing Weight (lb)	244
PLF Dimension (ft)	6.0 x 3.833

	1st Stage	**2nd Stage**	**3nd Stage**
Inert Weight (lb)	2,780	800	277
Gross Weight (lb)	30,908	7,500	2,172*
Propellant Weight (lb)	26,790	6,670	1,725
Burn Time (sec)	72.3	71.4	64.6
Max. Pressure (psia)	1,088	1,003	749
Avg Pressure (psia)	797	793	637
Max Vac Thrust (lbf)	131,244	30,912	9,065
Avg Vac Thrust (lbf)	109,419	27,605	7,772
Vac Impulse (lbf-sec)	7,911,000	1,971,000	502,100
Isp Vac (sec)	295.3	295.5	291.1

* Excluding payload.

Table 2.7.2 Pegasus Typical Flight Operations Sequence

Events	Time (sec)	Altitude (ft)	Velocity (fps)
Release of Pegasus	0.0	40,000	____
Stage 1 Motor Ignition	5.0	39,700	____
Stage 1 Motor Burnout	81.8	199,000	____
Stage 2 Motor ignition	85.3	213,000	____
Payload Fairing Separation	124.0	369,000	11,900
Stage 2 Motor Burnout	160.0	556,000	17,500
Stage 2/3 Coast	163.0	____	____
Stage 3 Motor Ignition	466.0	1,506,872	15,800
Stage 3 Burnout / Orbital Insertion	531.5	1,519,025	25,043.2

Table 2.7.3 PEGASUS MISSION CAPABILITIES

TYPICAL MISSIONS	PAYLOAD (lb)	INCLINATION (deg)
Circular Polar Orbit	600	90
Low Earth Orbit	900	0-20
Suborbital*	2000	

* Non satellite payloads (attached or deployed): depressed suborbital trajectory.

2.8 SCOUT

BACKGROUND ON THE VEHICLE

Designed by Vought as a low-cost launcher for sub-orbital orbital and re-entry reseach, Scout was the 1st US vehicle to use all-solid propellant. First successful orbital launch was Explorer 9 on Feb 1961 [14]. Since the Scout launch, in 1960, the Scout launch vehicle has demonstrated an increasingly important role as a highly reliable, cost-effective vehicle within the US national booster inventory. This vehicle has achieved a flight reliability record in excess of 95% in launching a variety of United States and international payloads into orbital, probe and re-entry missions. These missions have encompassed inclined, equatorial and polar orbits.

The standard Scout launch vehicle is a solid propellant, four stage booster system approximately 23 meters in length with a launch weight of 21,500 kilograms and lift-off thrust of 588,240 newtons. LTV Missiles and Electronics Group_ Missile Division is the prime vehicle contractor to the NASA for the Scout launch vehicle [15].

PROPULSION SYSTEM

Stage I

The Algol IIIA, Scout first stage propulsion unit, is manufactured by the United Technologies Corporation/Chemical System Division. The Algol IIIA combines a steel motor case with a lightweight reinforced plastic nozzle.

Stage II

Castor IIA motor is the Scout second stage propulsion unit. It is manufactured by Morton-Thiokol Corp.

Stage III

The Antares IIIA rocket motor, Scout third stage propulsion unit, is manufactured by the Morton-Thiokol Corp. The motor case is kevlar and epoxy composite. The nozzle uses a 4D carbon/carbon throat insert, a carbon-phenolic exit cone and titanium housing.

Stage IV

The Altair IIIA rocket motor, scout foruth stage propulsion unit, is manufactured by Morton-Thiokol Corp. The motor is filament wound of fiberglass. Tables 2.8.1 - 2.8.7 show all the relevant features of the Scout launch vehicle [15].

The Scout vehicle can be used for a wide variety of reentry conditions. Typically the first two stages are fired, the vehicle coasts beyond the peak altitude and then the upper stages are fired to drive the payload back into the atmosphere. Typical re-entry performance is presented in table 2.8.5 for Vandenberg Air Force base Westerly launches.

As mentionned earlier, the Scout launch vehicle can be used as a probe booster as well. For that configuration, all the stages of the vehicle are fired in sequence and the payload coasts to apogee (see table 2.8.6).

Table 2.8.1 SCOUT MOTOR PERFORMANCE CHARACTERISTICS

Stage	Motor Name	Total Impulse Vacuum (N-S)	Avg Web Thrust Vacuum (N)	Web Burn Time (sec)	Total Weight (kg Mass)
I	ALGOL 3A	32,325324	464,332	58.44	14,220
II	CASTOR 2A	10,254806	267,162	37.71	4,421
III	ANTARES IIIA	3,721,804	80,764	44.90	1,395
IV	ALTAIR IIIA	772,826	25,821	29.06	301

Table 2.8.2 SCOUT STANDARD VEHICLE WEIGHT TABLE

VEHICLE DIMENSIONS:	
Height overall	22.92 m
Maximum body diameter	1.14 m
Launch Weight	21,400 kg
Lift-off Thrust	588,240 N

STAGE WEIGHT (LESS PAYLOAD)	KILOGRAMS
Payload	0.0
Fourth Stage-Inert	41.0
Fourth Stage Burnout	41.0
Fourth Stage -Consumed	275.0
Fourth Stage Ignition	317.0
Third Stage-Inert	335.0
Third Stage Burnout	653.0
Third Stage-Consumed	1,299.0
Third Stage Ignition	1,953.0
Second Stage-Inert	1,064.0
Second Stage Burnout	3,017.0
Second Stage-Consumed	3,768.0
Second Stage Ignition	6,786.0
First Stage-Inert	1,939.0
First Stage Burnout	8,725.0
First Stage-Comsumed	12,817.0
First Stage Ignition	21,543.0

Table 2.8.3 SCOUT TYPICAL SEQUENCE OF EVENTS

EVENTS	TIME (sec)
Stage 1 Ignition	-0.13
Lift-off	0.00
Stage 1 Burnout	81.45
Stage 2 Ignition	89.76
Stage 2 Burnout	128.49
Stage 3 Ignition	133.49
Stage 3 Burnout	179.14
Activate "c" Coast Controls	184.14
Explosive Bolt Ignition, Separate 3rd stage	611.76
Retro Force Command	612.26
Stage 4 Ignition	616.61
Stage 4 Burnout	650.75

Table 2.8.4 SCOUT PERFORMANCE CAPABILITIES

MISSIONS	PAYLOAD (KG)	INCLINATION	LAUNCH SITE
Circular Orbit	202	37.7	Wallops
Circular Orbit	230	2.9	San Marco
Circular Orbit	175	90.0	Vandenberg

TABLE 2.8.5 Scout Trajectory And Re-entry Value

Trajectory* Parameters	Nominal Re-entry Value at Fourth Stage Burnout		
Range	1200 Km	3700 Km	7780 Km
Altitude	91 Km	122 Km	122 Km
Relative Velocity	5.4 Km/sec	6.1 Km/sec	6.9 Km/sec
Relative Flight Path Angle	- 20 deg	- 20 deg	- 20 deg

* For a payload of 270 kg

Table 2.8.6 SCOUT PROBE CONFIGURATION

Launch Site	Inclination	Performance 4[nd] Stage	Apogee Altitude	0-G Time
Wallops Flight Facility	37.7	91 Kg	14631 Km	263*
Wallops Flight Facility	37.7	181 Kg	3111 Km	116*

* Minutes

Table 2.8.7 WALLOP FLIGHT FACILITY 90 degree LAUNCH AZIMUTH

STAGES	PAYLOAD WEIGHT (lbs/kg)	APOGEE ALTITUDE (nm/km)	RANGE @IMPACT (nm/km)
3 Stages	1000 / 450 1000 / 450	1280 / 2370 200 / 370	1290 / 2390 2450 / 4540
2 Stages	1000 / 450 1000 / 450	710 / 1315 110 / 205	525 / 970 1200 / 2220
1 Stage	1000 / 450 1000 / 450	380 / 705 300 / 555	160 / 295 610 / 1130

2.9 SPACE SHUTTLE

BACKGROUND ON THE VEHICLE

When first conceived in 1969 the space transportation system was to be recoverable. But a proposed manned launch vehicle somewhat similar to the currently proposed United German Sanger system was abandoned because of its high development cost in favour of the present system. This involves the Solid Rocket Boosters parachuting into the sea for recovery, and the external tank being irrecoverably jettisoned. Shuttle made its maiden fligh in 1981.

The Space Shuttle is composed of the Orbiter, an External Tank that contains the propellant to be used by the Orbiter's three main engines, and two Solid Rocket Boosters. The Orbiter and the boosters are reusable; the External Tank is expended on each launch.

ORBITER

The Orbiter carries the crew and the payloads for the Space Shuttle. About the size of a commercial D.C-9 jet airlined, the Orbiter can deliver to orbit single or multiunit payloads up to 29,000 kg (65,000 lb) in its huge 4.5 by 18 meter (15 by 60 ft) cargo bay, and bring back payloads weiging up to 14,515 kg (32,000 lb) [16].

The Orbiter's main structural elements, constructed primarily of aluminum, are the forward fuselage containing the crew module, the mid fuselage including the payload bay doors, the aft fuselage including the engine thrust structure, the wing, and the vertical tail. The Orbiter's exterior is covered with thermal protective materials to protect the spacecraft from solar radiation and the extreme heat of atmospheric reentry. Two types of reusable insulation, coated silica tiles and coated flexible sheets, cover the top and sides of the Orbiter. The tiles protect the Orbiter surfaces up to 649 degrees Celcius (1,200 degrees F), and the flexible insulation protects the Orbiter up to 371 degrees C (700 degrees F). The coating on both types of insulation gives the Orbiter a nearly white color and has optical propeties that reflect the solar radiation [16].

On the bottom of the Orbiter and on the leading edge of the tail, a high temperature reusable surface insulation, made of coated silica tiles, is used to protect the aluminum structure up to 1,260 degrees C (2,300 degrees F). The high temperature coating gives a glossy black appearance.

A reinforced carbon-carbon material is used for the nose cap and the wing leading edges where the temperatures exceed 1,200 degrees C (2,300 degrees F) during reentry [16].

The Orbiter carries a crew of three and as many as four additional payload and technical personnel, occupying a two-level cabin within the crew module at the forward end of the vehicle.

PROPULSION SYSTEM

Three main propulsion rocket engines used during launch are in the aft fuselage. Two orbital maneuvering rockets in external pods on the aft fuselage provide thrust for orbit insertion, orbit changes, rendez-vous, and return to Earth. Reaction control thrusters in the two orbital maneuvering system pods and in a module in thee nose section of the forward fuselage provide attitude control in space and precision velocity changes for the final phases of rendez-vous and docking or orbit modification.

The Space Shuttle Main Engine, designed to power the Space, along with two Solid Rocket Boosters, is the most advanced liquid-fueled rocket engine ever built. With variable thrust permitting the engine thrust to be tailored to the mission needs, it can operate effectively at both high and low altitudes. It has the highest thrust for its weight of any engine yet developed and can operate up to 7.5 hours of accumulated firing time before maintenance or overhaul. The Main Engine is reusable for up to 55 separate Shuttle missions [16].

The three Main Engines are mounted on the Orbiter aft fuselage in a triangular pattern. The engines are so spaced that they can be gimballed during flight and, in conjunction with the two Solid Rocket Boosters, are used to steer the Shuttle during flight as well as provide thrust for launch.

The Space Shuttle Main Engine (SSME) is a stage combustion cycle engine which operates at extremely high pressures to reduce engine envelope while attaining efficiencies previouly unknown in rocket engine technology. Burning liquid hydrogen and liquid oxygen, the three engines in the shuttle orbiter provide the majority of the total impulse (thrust times duration) required to attain orbital velocity. Solid Rocket Boosters, which provide lift-off thrust are dropped after two minutes operation [17]. Fuel for the engines, liquid hydrogen and liquid oxygen, is carried in the External Tank, the largest element of the Space Shuttle. Fuel can be supplied from the tank at a rate of about 171,396 liters (45,283 gallons) per minute of hydrogen and about 63,588 liters (16,800 gallons) per minute of oxygen. Each engine has three primary levels of thrust or power - minimum, rated and full power. Engine thrust, however, can be varied throughout the range from minimum to full power level depending upon mission needs. At rated power, each engine develops 2,090,560 Newtons (470,000 lb) of thrust at altitude or 1,668,000 Newtons (375,000 lb) at sea level [16]. They can burn for about eight minutes, while drawing 242,240 liters (64,000 gallons) of propellants each minutes, and are used to take the orbiter to the edge of space and a near-orbit velocity. The Orbital Maneuvering System (OMS) engines, burning nitrogen tetroxide for the oxidizer and monomethyl hydrazine for the fuel, supply the last increment of velocity to reach orbit, and the thrust for in-orbit and de-orbit maneuvers. These propellants are from tanks carried in two pods at the upper rear of the orbiter [18].

Among the key components of the engine are its four turbopumps (two low pressure and two high pressure), two pre-burners, the main injector, the main combustion chamber, the nozzle, and the hot-gas manifold. The nozzle area ratio is 77.5:1.

SOLID ROCKET BOOSTER

Two Solid Rocket Boosters are used for each Space Shuttle flight to provide, along with the Orbiter main engines, the initial ascent thrust to lift the Shuttle with its payload from the launch pad to an altitude of about 44 kilometers (27.5 miles). Prior to launch, the entire Shuttle weight is supported by the two boosters.

The Solid Rocket Booster is made up of six subsystems: the solid rocket motor, structures, thrust vector control, separation, recovery and electrical and instrumentation subsystems. The heart of each booster is the motor, the largest solid rocket motor ever flown and the first designed for reuse [16].

The two solid rocket boosters are each 45.4 meters (149.1 feet) high and 3.7 meters (12.2 feet) in diameter. Each weighs 589,670 kg (1,300,000 lb). Each motor, when assemble, contains about 503,600 kg (1.1 million lbs) of propellant. Their solid propellant consists of aluminum powder, aluminum perchlorate powder, and a dash of iron oxide catalyst, held together with a polymer binder. They produce over 13,300,000 Newtons (3,000,000 lbs) thrust each at liftoff. Together with the three main engines on the orbiter, this provides a total liftoff thrust of over 31,000,000 Newtons (7,000,000 lbs) [18].

At burnout, the two Solid Rocket Boosters are separated from the External Tank by pyrotechnic (explosive) devices and moved away from the Shuttle vehicle by eight separation motors, four housed in the forward nose frustum and four on the aft skirt. Each of the eight separation motors, fired at solid rocket motor burnout, develops a thrust of 97,856 N (22,000 lbs) for a duration of a little more than one-half second, just enough to move the boosters away from the still accelerating Orbiter and External Tank. The two Solid Rocket Boosters are recovered for reuse [16].

EXTERNAL TANK

The external Tank has two major roles in the Space Shuttle program - to contain and deliver quality propellants, liquid hydrogen and liquid oxygen to the engines and to serve as the structural backbone of the Space Shuttle during launch operations.

The External Tank is composed essentially of two tanks, a large hydrogen tank and a smaller oxygen tank, joined together to form one large propellant storage container 47 meters (154.2 feet) long and 8.4 meters (27.5 feet) in diameter. It weighs a total of 756,441 kg (1,667,677 lbs) at liftoff.

The oxygen tank is the forward portion of the External Tank and, when loaded, contains 603,983 kg (1,331,555 lbs) of liquid oxygen. (As a comparison, the oxygen tank has 552 cu. meters or 19,500 cubic feet - more volume than that of a 186 sq m or 2,000 sq ft. house) [16].

The liquid hydrogen tank, aft of the oxygen tank, is about two and one-half times larger than the oxygen tank. In this tank is stored 101,503 kg (223,775 lbs) of cold liquid hydrogen (about minus 251 degrees C or minus 420 degrees F) [16].

For launch, the External Tank supports the Orbiter and Solid Rocket Boosters at attach points on the tank. Since thrust is generated by the Orbiter main engines and the Solid Rocket Boosters, the External Tank must absorb the thrust loads for the Shuttle during launch.

Much of the outer surfaces of the tank are protected thermally. Spray-on foam insulation is applied over the forward portion of the oxygen tank, and the sides of the hydrogen tank. The foam insulation is needed to reduce ice or frost formation on the tank during launch preparation, thus protecting the Orbiter insulation from free-falling ice during flight, to minimize heat leaks into the tank which would cause excessive boiling of the liquid propellants, and to prevent liquification and solidification of the air next to the tank.

An ablating material, a material that chars away, is applied to the External Tank bulges and projections to protect them from aerodynamic heating during flight through the atmosphere. Sometimes a combination of foam and ablator is used where both heating and insulation protection is needed. Protection is also needed in areas where exhausts of launch engines provide high radiant energy to the tank and where separation motors' exhaust plumes may strike the tank [16].

Near the end of the launch phase of the Shuttle mission when the Orbiter is just short of orbital velocity the main engines are cut off and 10 seconds later, the External Tank is severed from its attachment to the Orbiter, playing a totally passive role in the separation sequence. The tank breaks up upon reentry and fall within the designated ocean impact area. The External Tank is the only major Shuttle component expended on each launch.

The rotation of the Earth has a significant bearing on the payload capabilities of the Space Shuttle. A due east launch from the Kennedy Space Center (KSC) in Florida uses the Earth's rotation as a launch assist, since the ground is turning to the east at that point at a speed of 1,473 km (915 miles) per hour. This permits payloads on the fully developed Shuttle to weigh as much as 65,000 lbs when launched from KSC.

A launch to the south into polar orbit from Vandenberg AFB in California, where the Earth's rotation neither assists nor hinders the Shuttle's capabilities, permits a payload of up to 40,000 lbs to be carried into orbit. The most Westerly launch from VAFB allows a payload up to only 32,000 lbs to be transported to orbit since the Earth's rotation is counter to the Westerly launch azimuth (heading) [16].

SPACE MISSION PROFILE

The Space Shuttle is launched with all three SSME's operating in parallel with the SRB's. The orbiter's main engines ignite first and build to full power before the huge solid rockets ignite and liftoff occurs. The solid rockets burn out after about two minutes - about 44 km (27.5 miles) high [18]. After SRB separation, the Orbiter and ET continue ascent, using the three main engines - a total of about eight minutes of burn-time on the three main engines. Main engine cutoff (MECO) occurs about 8 minutes after liftoff. Then the ET is separated from the Orbiter. After a short coasting period, the Orbiter two OMS engines then burn to inject the orbiter alone into low Earth orbit. Later OMS burns raise or adjust the orrbit as necessary. A typical Shuttle mission lasts from two to ten days. Upon completion of on-orbit operations, the payload bay doors are closed, and the Orbiter is configured for return to Earth.

The Orbiter returns to Earth by firing the OMS engines to reduce velocity. After reentering the Earth's atmosphere, the Orbiter glides to an unpowered landing. Unlike prior manned spacecraft, which followed a ballistic trajectory, the Orbiter has a crossrange capability (can move to the right or the left off the straight line of its entry path) of about 2,045 km (1,270 miles). The landing speed is from about 341 to 364 km per hour (212 to 226 miles per hour) [18]. Tables 2.9.1 - 2.9.3 present a summary of all the relevant features of the Space Shuttle.

Table 2.9.1 SPACE SHUTTLE MAIN ENGINE PERFORMANCE

SPACE SHUTTLE MAIN ENGINE PERFORMANCE (FULL POWER LEVEL)			
Maximum Thrust: (109% power level)		Power:	
At Sea Level	408,750 lb	High Pressure Pumps	
In Vacuum	512,300 lb	Hydrogen	74,928 hpower
Pressures:		Oxygen	28,229 hpower
Hydrogen Pump Discharge	6,872 psia	Weight:	7,000 lb
Oxygen Pump Discharge	7,936 psia	Length:	14 ft
Combustion Pressure	3,277 psia	Diameter:	7.5 ft
Flow rates:		Propellants:	
Total	1,130 lb/sec	Fuel	Liquid Hydrogen
Hydrogen	160 lb/sec	Oxidizer	Liquid Oxygen
Oxygen	970 lb/sec		

Table 2.9.2 PAYLOAD PERFORMANCE OF THE SPACE SHUTTLE

LAUNCH SITE	INCLINATION (deg)	ALTITUDE (n.mi)	WEIGHT CAPABILITY (lb)
Kennedy Space Center	28.5	160.0	65,000*
" " " "	57.0	160.0	56,000*
" " " "	55.0	431.0	25,000*
Vandenberg AF Base	104.0	160.0	32,000**1
" " " "	90.0	130.0	40,000**2
Landing Site (KSC/VAFB) With Payload:			32,000

* : For a Due East launch; **1: For a Due West launch (Sun-synchronous orbit); **2: For a Due South launch (Polar orbit).

Table 2.9.3 SPACE SHUTTLE CHARACTERISTICS

SPACE SHUTTLE SYSTEM	
OVERALL LENGTH	184.2 ft / 56.1 m
HEIGHT	76.6 ft / 23.3 m
SYSTEM WEIGHT	
-DUE EAST	4,490,800 lb / 2037 Mg
- 104 degrees	4,449,000 lb / 2018 Mg
PAYLOAD WEIGHT	
-DUE EAST	65,000 lb / 29,483 kg
-104 degrees	32,000 lb / 14,515 kg
EXTERNAL TANK	
DIAMETER	27.8 ft / 8.5 m
LENGTH	154.4 ft / 47.1 m
WEIGHT	
-LAUNCH	1,649,600 lb / 748,242 kg
-INERT	71,000 lb / 32,205 kg
SOLID ROCKET BOOSTERS	
DIAMETER	12.2 ft / 3.7 m
HEIGHT	149.1 ft / 45.4 m
WEIGHT (each)	
-LAUNCH	1,292,600 lb / 586,310 kg
-INERT	183,800 lb / 83,370 kg
THRUST (each)	
-LAUNCH	2,700,000 lb / 12,010,140 N
SEPARATION MOTORS (each SRB)	
-4 AFT 4 FORWARD	
-THRUST (each)	22,000 lb / 97,860 N
ORBITER	

TABLE 2.9.3 SPACE SHUTTLE CHARACTERISTICS (CONT..)

SPACE SHUTTLE SYSTEM	
LENGTH	122.2 ft / 37.2 m
WINGSPAN	78.1 ft / 23.8 m
TAXI HEIGHT	57.0 ft / 17.0 m
PAYLOAD BAY (size)	15 ft x 60 ft long / 4.6 m x 18.3 m long
CROSS RANGE	1100 n.mi / 2037 kmm
MAIN ENGINES (3)	
-VACUUM THRUST$_{each}$	470,000 lb / 2090.7 kN
OMS ENGINE (2)	
-VACUUM THRUST$_{each}$	6,000 lb / 26.7 kN
RCS	
38 ENGINES	
-VACUUM THRUST$_{each}$	870 lb / 3869.9 N
-6 VERNIER ENGINES	
-VACUUM THRUST$_{each}$	25 lb / 111.2 N
WEIGHT	
-INERT	162,000 lb / 73,482 kg
-LANDING	
WITH PAYLOAD	211,000 lb / 95,707 kg
WITHOUT PAYLOAD	179,000 lb / 81,193 kg

CHAPTER III

EUROPEAN LAUNCH VEHICLES

3.1 ARIANE-4

BACKGROUND ON THE VEHICLE

The development of different versions of the Ariane launcher (ARIANE 1 to ARIANE 5) is carried out under the aegis of the European Space Agency (ESA). ESA is, in particular, responsible for the overall management of the Ariane development programs with the French "Centre National d'Etudes Spatiales" (CNES) acting as prime contractor.

The Ariane program was iniated in December 1973. The first launch took place in December 1979 with Ariane I. Since the first launch in 1979, the Ariane family has grown, and there are already four versions of the launcher. Three of the versions are already phased out. Ariane flew its last mission in February 1986. Ariane II and III were phased out in july 1989.

The European Space Agency (ESA) has developed a more powerful version which entered service on June 15^{th}, 1988. The Ariane 4 development program started in January 1982 when the ESA decided to develop a more powerful version of the Ariane launcher. Ariane 4 is a futher development of the basic Ariane launcher and has six different versions with a capability of lifting between 1900 kg (4190 lb) and 4200 kg (9260 lb) into geostationary transfer orbit. All these versions have a lengthened first stage containing 226 tons of propellant, powered by four Viking V engines identical to those of Ariane 3, a second and third stage also identical to Ariane 3 except for the fact that the structures are reinforced, a new vehicle equipment bay to which it is possible to fit a payload bearing structure for multiple launches, called SPELDA (Structure Porteuse Extreme pour Lancement Double Ariane -Ariane Dual Launch External Bearing Structure), and a large-diameter (4m) fairing available in two lengths. The six Ariane 4 versions are differentiated by the number (four, two or none) and type (solid or liquid propellant) of strap-on boosters used for the first stage [19].

The choice of these various Ariane 4 configurations makes the launcher extremely flexible and its lift capability can be adjusted to payloads, with the possibility of multiple launches keeping launch costs down.

ARIANE 4 POSSIBLE CONFIGURATIONS:

ARIANE 40 : no strap-on boosters
42P : 2 solid strap-on boosters
44P : 4 solid strap-on boosters
42L : 2 liquid strap-on boosters
44LP : 2 liquid + 2 solid strap-on boosters
44L : 4 liquid strap-on boosters

The Ariane launches are made from the Ariane launch site (ELA) at the Guiana Space Center (CSG) near Kourou. Close to the equator (latitude 5.23 deg North), the Center is geographically well located, with a launch sector over the Atlantic Ocean extending from North to East (-10.5 deg to + 93.5 deg), which makes it ideally suited for launching satellites into geostationary orbit [19].

ARIANE PROPULSION SYSTEM

FIRST STAGE The propulsion system comprises four Viking V engines developing a total thrust of 2708 kN at liftoff. Each engine is an independent assembly supplied with propellant and water via its own valves. The propellants used are UH25 (a mixture of unsymmetrical dimethylhydrazine and hydrazine hydrate) and N_2O_4 (nitrogen tetroxide). Ariane 4's first stage carries 226 tons of these propellants. During the propulsion phase the first stage uses about one ton of propellant per second. Each Viking engine has a gas generator, supplied with propellant and water for cooling the gases. Burning time is 205 seconds [19].

The liquid propellant boosters (PAL): are a completely new element. They use the same propulsion system including the Viking engine as the first stage. They carry 39 metric tons of (N2O4 + UDMH + Hydrazine Hydrate). Each has a total thrust of of 666 kN and a burning time of 135 seconds [19]. They essentially comprise a Viking VI engine attached to a thrustframe at a fixed angle, two identical separate steel tanks, an intertank skirt, a forward skirt an a nose cone. Water for the Viking engine is supplied from the central water tank of the first stage. After burn-out the boosters are separated at the attachment ponts by pyrotechnic cutting devices, the removing forces being provided by small rockets installed on the boosters.

The solid propellant boosters (PAP): the solid propellant boosters are attached symmetrically to the rear part of the first stage. Each weighs 11.9 tons, of which 9.5 tons is grain. They develop a thrust of about 700 kN each, over a burn-time of 42 seconds. The nozzle, of the semi-hooded type, is inclined about 14 degrees to the booster axis. The boosters are ignited in flight at about 11 meters from the lift-off level and are jettisoned by a spring system after 32 seconds [20].

SECOND STAGE the second stage (L33) when empty weighs 3.6 tons (including the second-third stage interstage skirt) and measures 11.6 m in height and 2.6 m in diameter. Its Viking IV engine functions at a chamber pressure of 58.5 bars and develops 786 kN of thrust in vacuum. The integral tank with a common bulkhead is made in aluminium light alloy. Both tank compartments are pressurized by Helium gas stored in spherical bottles at a pressure of 300 bars at ambient temperature. The engine swivelled about two axes allows yaw and pitch control; roll control is provided by two tangential hot gas jets (50 N of thrust) [19].

THIRD STAGE the third stage (H 10) weighs 1.2 tons when empty and has a diameter of 2.6 m for a height of 9.9 m. Its engine, HM7B, develops a thrust of 62 kN in vacuum. It operates at a chamber pressure of 35 bars and a burn time of 725 seconds. Two aluminum tanks contain 10.7 tons of cryogenic propellants (Liquid Oxigen and Liquid Hydrogen). Both propellant compartments are separated by a vacuum double-walled common bulkhead. The engine is linked to the conical thrustframe through a gimbal joint allowing swivelling of the engine for pitch and yaw attitude control. Gaseous Hydrogen thrusters provide rotational momentum for roll control. After engine cut-off these thrusters together with additional hydrogen thrusters perform three axes attitude control of the stage and attached payload [19]. Tables 3.1.1 - 3.1.3 show all the relevant features of the Ariane-4 configuration.

Table 3.1.1 ARIANE-4 VEHICLE CHARACTERISTICS

	Height/Diameter/Drymass	Propulsion system mass-propellant/thrust
FAIRING	13.5 m / 4 m / 816 kg	________
SPELDA	2.8 m / 4 m / 410 kg	________
VEB	1.0 m / 4 m / 530 kg	________
3nd Stage	9.90 m / 2.6 m / 1.20 t	10.7t -LH_2&LO_2/ 63 KN
2nd Stage	11.6 m / 2.6 m / 4.2 t	34t-UH25&N_2O_4/ 800 KN
1st Stage	23.2 m / 3.8 m / 20.0 t	228t-UH25&N_2O_4/3000 KN
PAL	19 m / 2.2 m/ 4.5 t	39t-UH25&N_2O_4/ 750 KN
VIKING IV, V, VI	________	Biliquid Propulsion
HM 7B	________	Cryotechnic propulsion

VEHICLE TOTAL HEIGHT TOTAL LIFT-OFF MASS	58.4 meters 418 tons

Table 3.1.2 ARIANE-4 TYPICAL FLIGHT OPERATION SEQUENCE

EVENTS	Time (sec)	Altitude (km)	Velocity (m/sec)
First Stage Ignition	0.0	0.0	
Ignition of solid boosters	3.0	0.0	
Lift-off	4.4	0.0	
End of vertical ascent	12.0		
Solid booster jettisoned	67.0	7.5	241.0
Liquid propellant booster jettisoned	146.0	37.0	1,272.0
First Stage Separation	214.30	74.0	2,786.0
2nd Stage Ignition	215.80		
Fairing jettisoned	284.80	115.0	3,863.0
2nd Stage Separation	344.30	147.0	5,379.0
3nd Stage Ignition	347.80		
2nd Stage Burnout	1060.30		
Injection into target Orbit	1061.70	225.0	9,740.0
Separation of upper satellite	1212.30		
Separation of upper part of SPELDA	1360		
Separation of lower satellite	1457.30		
3nd Stage avoidance procedure	1487.30		
End of Ariane mission	1835.30		

Table 3.1.3 ARIANE-4 PAYLOAD CAPABILITY

	ARIANE-4 : SIX VERSIONS					
PAYLOAD MASS (kg)	A40	A42P	A44P	A42L	A44LP	A44L
Single launch into GTO	1900	2600	3000	3200	3700	4200
Dual launch into GTO	1700	2400	2600	2800	3300	3800
Sun-synchronous orbit	2700	3400	4100	4500	5000	6000
Low circular orbit	4600	6000	6500	7000	7000	7000

3.2 ARIANE-5

This version represents a departure from the evolutionary step-by-step growth pattern of the Ariane 1-4 series, since it involves a completely different design philosophy for the main core stage. In order to keep the economic advantage of double launches, Europe needs a more powerful launcher which is capable of performing dual launches with satellite masses up to 3000 kg each. Ariane 5 development started in November 1987, and it will be operational by mid-1990s.

The Ariane 5 launch system will be called upon to carry out three types of mission [21], namely to:

- launch commercial geostationary and sun-synchronous satellites, and scientific and experimental application satellites,
- launch the Hermes manned spaceplane,
- launch elements of the Columbus system.

The Ariane 5 launcher consists of a lower, two-stage composite, identical for Hermes launches and for automatic payload, comprising a cryogenic stage, fuelled by hydrogen and liquid oxygen, and two solid boosters. The cryogenic stage has a single engine, the VULGAIN, ignited before lift-off so that its operation can be checked before the solid boosters are fired. For HERMES launches, the lower composite directly supports the unit constituted by the spaceplane and its resource module. For the launch of automatic payloads, the lower composite is topped by an upper composite, the configuration of which depends on the particular mission. The simultaneous launch of several satellites into geostationary transfer orbit uses an upper composite made up as follows: a single-engine storable proppellant upper stage, a Vehicle Equipment Bay (VEB), one or two bearing structures (SPELTRA) designed to enable two or three satellites respectively to be carried, and a short fairing. Launch of a single satellite involves an upper stage with vehicle equipment bayy and "stretched fairing. The capacity of the stretched fairing is sufficient to permit the launch of COLUMBUS elements (the COLUMBUS free flying laboratory and the polar platform) [22].

PROPULSION SYSTEM

LOWER COMPOSITE

The core vehicle consists of a cryogenic first stage. It measures 5.4 meters in diameter and 29 meters high. It is the main element of the launcher. Its VULGAIN engine provides most of the kinetic energy needed to place the payloads into orbit. The engine is ignited on the ground before lift-off and produces a thrust of 110 tons in vacuum, burning 24 tons of hydrogen and 126 tons of oxygen stored in liquid staate at low temperatures in the fuel tanks in ten minutes [22].

Two three-segmented P230 solid boosters on Ariane 5 consist essentially of the motors using solid propellants. They have a diameter of 3.03 meters, a height oof 30 meters and a mass at lift-off of 265 tons. The solid propellant motor delivers a thrust of 600 tons at lift-off, falling off after 15 seconds to reduce the aerodynamic loads on the launcher and increasing again to give a total burn time of 120 seconds for 230 tons of solid propellant. The motor consists of 3 propellant-filles segments, joined together. The steel case is made of 7 cylindrical sections and a nozzle fitted with a flexible bearing so that the direction of thrust may be varied by up to 6 degrees [22].

For unmanned automatic flights launching satellite into GTO, a L7 second stage, similar to the Ariane 4 second stage and using 7.2 metric tons of the same UH25/N_2O_4 hypergolic propellant combination, provides 27.5 kN of vacuum thrust. The configuration will be able too place up to 6.8 tons (single payload) into GTO, and 5.9 tons for a dual launch using the SPELTRA (Ariane triple launch external bearing structure) system. Vehicle diameter has been increased to 5.4 meters, which, with the constant diameter fairing configuration, results in an interior clearance envelope of 4.6 meters, equivalent to that of the U.S Shuttle payload bay. The fairing and SPELTRA configuration will enable many combinations of single, double and triple launches to be accommodated. All vehicle command and sequencing is designed to handle the separation and maneuvering requirements for three different payloads [23].

UPPER COMPOSITE

For the launch of conventional payloads the uppe composite is fixed to the main cryogenic stage. Apart from the payloads, it consists of:

- Storable propellant stage,
- Vehicle equipment bay,
- 0, 1, 2 SPELTRAS (Ariane triple launch external bearing structure) depending on whether 1, 2, or 3 payloads are being launched,
- the "short" fairing for conventional launches and the "stretched" fairing for certain single launches of "COLUMBUS" elements,
- and payload adaptors.

The upper composite has an external diameter of 5.4 meters and a height of 23 meters [22].

For manned missions the upper composite is made up of the Hermes spaceplanee mounted on its adaptor (fitted with an additional propulsion unit).

PERFORMANCE

Ariane 5 will place into geostationaryy transfer orbit one or more satellites with a guaranteed total mass of 6800 kg (including the adapter or multiple-launch devices), using a medium-power upper stage. This equates, in the dual-launch configuration, to a net lift performance of 5900 kg.

Ariane 5 will place space station modules or platforms with a guaranteed mass of 18000 kg into a circular Low-Earth orbit of 550 x 550 km x 28.5 degree and platforms with a guaranteed mass of 12000 kg into a circular Low-Earth orbit of 800 x 800 km x 98.6 degree.

Ariane 5 will place the Hermes spaceplace into a transfer orbit preparatory to reaching a final circular orbit of 500 x 500 km x 28.5 degree, in such a way that when the correct circular orbit is obtained by Hermes, the spaceplane mass will be 21,000 kg [21].

ARIANE 5 FLIGHT SEQUENCE

The launcher's powered flight divides into two main phases:

First Phase (Lower Composite)

It starts with ignition on the ground of the Vulcain engine in the cryogenic main stage. Following a check to ensure that this engine is functionning correctly, the command is given to ignite the solid-propellant boosters and this initiates lift-off. The boosters are jettisoned when they have burnt out (@ Time = 2 minutes and H = 60 km) and propulsion continues under the power of the Vulcain engine alone. Fairing jettisonning occurs at an altitude of 110 km. H155 / L5 separation occurs at an altitude of 145 km and t = 9 minutes [21].

Second Phase (Upper Composite)

For geostationary missions, the storable-propellant stage (L5) is ignited immediately after separation from the cryogenic main stage. L5 burn-out occurs at t = 22 minutes, follows by payload pointing and separation.

In Low-Earth orbit missions, on completion of the first phase the storable-propellant stage, with the vehicle equipment bay and payloads, is placed in its final orbit. With a Hermes launch, the end of the first phase sees the separation of the spaceplane from the H155 stage. Hermes is then placed in a transfer orbit by the propulsion unit in its adaptor and it moves into a circular orbit at the first apogee and commences the manoeuvres required for its mission.

First flight of Ariane 5 is currently scheduled for mid-1995, and commercial operation by Arianespace will begin in 1996. The man-rated version to be employed with the Hermes spaceplane will fly for the first time in 1997. It is expected that the gradual buildup in Ariane 5 flight rate near the end of the decade will result in the phasing out of the Ariane 4 in about 1998. Tables 3.2.1 - 3.2.2 show the characteristics and performance of the Ariane-5 configuration.

Table 3.2.1 TECHNICAL CHARACTERISTICS OF THE ARIANE-5 GTO

	SOLID PROPELLANT BOOSTER (P230)	CRYOGENIC MAIN STAGE (H155)	STORABLE PROPELLANT STAGE (L5)	VEB	FAIRING	COMPLETE LAUNCHER
DIAMETER (m)	3.0	5.4	5.4	5.4	5.4	——
HEIGHT (m)	30	30	4.5	2.2	20/11	50.0
TOTAL MASS (tons)	269 each unit	170	6.0	1.1	2.4/1.4	725.0
MASS OF PROPELLANT @ LIFT-OFF (tons)	230 each unit	LOX: 130 LH_2: 25 TOTAL: 155	MMH: 1,7 N_2O_4: 3,5 TOTAL: 5,2	HYDRAZINE 60 Kg		620
BURN TIME (s)		615	800			1415
THRUST (tons)	650 each unit	104 (Vac.)	2.0			1590

Table 3.2.2 ARIANE-5 PERFORMANCE CAPABILITIES

Single Launch Payload into GTO	6800 kg
Dual Launch into GTO	5900 kg
Payload into LEO (800 x 800 x 98.6 deg.) (Unmanned Mission)	12,000 kg
LEO (550 x 550 km x 28.5 deg.) (Unmanned Mission)	18,000 kg
LEO (550 x 550 km x 28.5 deg.) (Manned Mission, Hermes)	21,000 kg
Design Reliability (Unmanned Missions)	98%
Design Safetyy (Manned Missions)	99.9%

3.3 PROTON

BACKGROUND ON THE VEHICLE

The Soviet Union's Proton Launch Vehicle is a versatile, medium heavylift launch vehicle capable of placing Atlas and Ariane class payloads into geostationary orbit, and heavier class payloads into Low Earth orbit. Proton has been in operation since 1965. According to Western and Soviet data, Proton has 133 launch successes out of 155 attempts to its credit through 1987 [24].

The Soviet Proton launch vehicle development started in early 1960's under the direction of the Academician V. N. Chelomey design bureau. This development effort culminated in July, 1965, with the first operational test launch of the new launch vehicle [24].

Series production of the new vehicle started in June, 1966, with modest runs of about five vehicles per year. As more experience was gained with the new vehicle, production rates increased to eight boosters per year by the 1980's. Because the Proton is a versatile, reliable vehicle production increased to about 13 boosters per year as of 1985, even though only eight boosters were launched in 1986 [24]. Only two or three Proton launch vehicles are available per year for commercial launches, primarily due to the five year plan batch production as contingencies in case of launch failures and time constraints at the cosmodrome.

Proton has launched all Soviet space stations, from Salyut-1 in 1971 to Mir in 1986. More recently, Proton was the launcher for the two VEGA spacecraft, which flew by Halley's Comet, and the two Phobos spacecraft to Mars. Proton has been used to launch the Zond, Luna, Venera, and other Mars, lunar and planetary exploration spacecraft [24].

There are two basic versions of the launch vehicle; the three-stage (D-1, SL-13) and the four-stage (D-1-e, SL-12). Both are designed to insert payloads weighing up to 20 tons into Low Earth orbit an inclination of 51.6 degrees and a near circular orbital altitude of 180 by 200. The launch vehicle with its fourth stages is used to insert the spacecraft into geostationary orbit, as well as into interplanetary trajectories.

Proton launch vehicle is made up of three and four tandem stages with lateral stage separation. The total length of the basic D-1 three stage launch vehicle, without payload, is 44.3 m; its lateral base diameter is 7.4 m. Proton is constructed of an all-aluminum alloy and is covered by an external thermal finishing paint. Proton's propellant tanks are internally coated with an anti-corrosive coating as are the oxidizer tanks. The oxidizer also has a passive abator mixed into it to reduce its corrosiveness [24].

PROPULSION SYSTEM

All stage have high-efficiency, reliable, small, single-chamber ZHRD (liquid propellant rocket engine), which operate with afterburning of the gas-generation products after the turbine in the high-pressure combustion chambers. The first three stages utilize a high-boiling bipropellant: oxidizer-nitrogen tetroxide (AT), fuel - unsymmetrical dimethylhydrazine (NDMG) [25].

FIRST STAGE

The first stage has six RD-253 single-chamber rocket engines burning nitrogen tetroxide and unsymmetrical dimethylhydrazine (UDMH). Each engine delivers a thrust of 150 tons at sea level and 166 tons in vacuum; consequently total lift-off thrust is 900 tons. Burn time is 130 seconds. The engines operate on the pre-burner closed cycle principle. These engines, characterized by high specific impulse and efficiency, have been used since 1965 on all stages of the Proton booster [26].

Their mounts ensure the rotation of the combustion chamber in the vertical plane to control the thrust vector. The reliability of the RD-253 has been structurally ensured by extensive application of welding: there are only 11 joints in its main lines. The on and off operations are executed by nine explosive valves of simple design. Heat shields are used to protect the engine units from the effect of the reactive jet [25].

SECOND STAGE

The second stage has four single high pressure combustion chamber liquid propellant rocket engines . Each develops about 60 tons in vacuum thrust. All the four engines use the same propellant used for the first stage.

THIRD STAGE

The third stage has a single, similar 60 tons thrust engine, plus four steerable vernier engines for course correction that together develop about three tons thrust. The 60 tons engine is also of the pre-burner type and it operates without afterburning [26]. The third stage is equipped with a triple redundant guidance system for the first three stages of the launch vehicle control. The third stage can not be restarted or throttled, but like all stages on Proton, the burn time can be varied according to the mission [24].

FOURTH STAGE

The fourth stage is a booster unit adapted for a lengthy stay in space with a capability of multiple starts of the sustainer ZhRD having a thrust of 8.5 tons and a total operating time of 600 seconds. The fuel in the fourth stage is oxygen and kerosene. Furthermore, is has an independent power plant for control during coasting. Adapter modules are used for mating the booster stage with the carrier: conical, which separate together with the third stage and cylindrical which is jettisoned after separation from the third stage. The adapter for docking the booster stage with a space vehicle is interchangeable [25].

The fourth stage is used to transfer payload from Low Earth orbit into geostationary orbit or interplanetary trajectory. This escape mission fourth stage is designed for a long operational life in the space environment and is equipped for up to seven engine restarts within a 48 hours of launch. Its fuel weight is 17.3 tons, and it has a maximum length of 5.5 m. The booster launch vehicle escape stage has ann interface diameter of 4 m. The single engine has a specif impulse of 3,450 m/sec or 351.8 seconds. Hypergolic propellants of nitrogen tetroxide and unsymmetrical dimethylhydrazine are used by the self-contained rocket thruster engines for three axis attitude control during the inactive coast periods of the fourth stage during the mission [24]. Tables 3.3.1 - 3.3.3 present the features of the proton launch vehicle.

PAYLOAD FAIRING SIZE

- 3.3 meters diameter (old shroud) x 4.2 meters long
- 3.8 meters diameter (new shroud) x 7.9 meters long

TABLE 3.3.1 PROTON RD-253 ROCKET ENGINE CHARACTERISTICS

CHARACTERISTICS OF THE GDL-OKB RD-253 ROCKET ENGINE	
THRUST CHAMBER WEIGHT	400 kg
CHAMBER PRESSURE @ 3127 C	150 atm.
OXIDIZER	Nitrogen tetroxide N_2O_4
FUEL	Unsymmetrical dimethylhydrazine UDMH
$THRUST)_{sea\ level}$	150,348 Kg
$THRUST)_{vacuum}$	166,770 kg
$SPECIFIC\ IMPULSE)_{sea\ level}$	285 sec / (2,795 m/sec)
$SPECIFIC\ IMPULSE)_{vacuum}$	316 sec / (3,100 m/sec)
LENGTH/DIAMETER OF CHAMBER	2.72 / 1.5 m
ENGINE BURN TIME	130 sec.
FLIGHT BURN TIME	120 sec.
CHAMBER PRESSURE	14.7 Mpa = 150-160 atm
EXIT PRESSURE	61 kpa @ 60 C (degree)
TURBINE POWER	18,740 K Watts
TURBINE REVOLUTIONS	231/ sec.
TEMPERATURE GAS GENERATOR	780 K
GAS GENERATOR PRESSURE	240 bar
MAXIMUM ENGINE MASS/CONSTRUCTION	1,280 / 1,460 kg (dry)/(fueled)
INSTALLED MASS	1,460 kg
MAXIMUM DIAMETER AT NOZZLE	1.5 m

Table 3.3.2 PROTON GENERAL CHARACTERISTICS

	1st STAGE	2nd STAGE	3rd STAGE	4th STAGE
Dry Weight	45,400 kg	15,600 kg	5,600 kg	2,650 kg
Fuel Weight	410,200 kg	150,000 kg	50,000 kg	17,300 kg
Number of Engines	6.0	4.0	1 + 4 verniers	1.0
Thrust by Stage	918,000*/1,000,620** kg	1,068,000** kg	64,260** kg	8,760** kg
Thrust per Engine	153,000* /166,770** kg	61,200 kg	61,200 & 765 kg	8,665 kg
Fuel by Stage	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	LO_2/Kerosene
Specific Impulse	285* / 316** sec	316 sec	316 sec	351.8 sec
Length (m)	20.2	13.7	6.4	5.5
Diameter (m)	7.4	4.15	4.15	3.7
Wet mass (t)	45.4	15.6	6.10	17.3
Exhaust velocity	3.10 km/sec	3.27 km/sec	3.37 km/sec	8.6 km/sec
Total Dry Weight				60,000 kg
Total Fuel Weight				694,000 kg
Liftoff Thrust				918,000*1 kg
Burn Time	127 sec	212 sec	250 sec	600 sec
PAYLOAD:				
-Low Earth orbit				20 tons
-Moonwards				5.7 tons
-Venuswards				5.3 tons
-Marswards				4.6 tons
-Geotransfer				over 5.5 tons
-Geostationary				2.2 tons
-Geosynchronous				2.0 tons

*: @ sea level; **: In vacuum; *1: Total thrust at liftoff

Table 3.3.3 PROTON MISSION PROFILE

EVENTS	TIME (sec)	EVENTS	TIME (sec)
Lifoff Contact.	0.0	3rd stage main engine shutdown	582
2nd stage engine start to medium thrust range	123.0	3rd stage steering engine shut-down	597
1st Stage engine shutdown	127.5	3rd/4nd separation	600
Nose cone jettison command	185.0	4th separation & orbital insertion	605
3rd stage start up	335.0	1st start of 4th stage engine	4,800
2nd stage engine shutdown	337.7	1st shutdown of 4th stage engine	5,250
2nd/3rd stage separation	338.4	2nd start of the 4th stage engine	25,530
3rd stage main engine start	340.8	2nd shutdown of 4th stage engine	25,760
Two half nose fairing separation	370.0	4th stage/spacecraft separation	25,776

3.4 ENERGIA

BACKGROUND ON THE VEHICLE

Exactly 30 years after the first blast-off of the "Seme-yerka" from korolev, the Soviet Union is launching its first heavy-lift rocket. Named "Energia", it will be spear-head of the major programs of the 1990s and even surpass the Soviet shuttle.

Energia is a two stage rocket weighing 2,400 tons at takeoff and can launch 100 tons into orbit. The first stage when it has completed its operation, is dropped and recovered in the Soviet Union. The second stage is continued its course until the satellite model is dropped [27]. Energia is the first Soviet rocket with cryogenic propulsion and the most powerful rocket produced by the USSR. The other Soviet rockets (Cyclone, Cosmos, Vosrok, Molniya, Soyuz, Zenith and Proton) launched between 1-20 tons of useful payload into a low orbit. In the present configuration, Energia can transport slightly more than 100 tons. It is the most power rocket in the world [28]!

The present version of the Energia has a height of 60 meters and a diameter of 20 meters and weights 2400 tons on takeoff and has 2000 tons of liquid ergols, which leaves an empty weight of about 300 tons with a payload of 100 tons [28].

PROPULSION SYSTEM

All of the engines of the first and second stages are ignited quasi-simultaneously before takeoff. The cryogenic engines are ignited first and then the RD 170 engines are ignited. The full thrust is achieved 3 seconds before takeoff. But 20 seconds after liftoff, it is possible to reduce the thrust of the engines of the second stage to reduce the aerodynamic loading, according to Goubanov, Chief designer of Energia [28].

The total thrust of the Energia thus exceeds 3550 tons of lift-off and 4000 tons in flight. The ratio of thrust to weight at liftoff of the rocket is therefore substantial (148/1). Energia is even superpowerful because it is dimensioned to ensure satellite placement, even if there is a failure of one of the first or second stage engines. Therefore, with a loss of thrust which could amount to 740 tons, which would make liftoff thrust somewhat less than 3000 tons, the ratio would still be sufficient (1.25/1). This over-dimensioning is designed to secure the orbiting of an inhabited version (of a spacecraft) [28].

FIRST STAGE

The Energia's first stage is made up of four cylinders with a maximum diameter of 5.9 m and a length of 40 m. Their upper ends are cut at a slant, and on the lower ends are installed individual propulsion engines with a thrust of 7848 KN each [29]. Each booster (4 in total) of the first stage has four RD 170 engines which burn a mixture of kerosene (RG1) and liquid oxygen and develop a unit thrust of 740 tons on the ground and 870 tons in vacuum. This amounts to respectively 2960 tons and 3227 tons of thrust for the first stage ensemble. These four chamber RD 170 engines are the most powerful in the world in their category [28]. The thrust produced is larger than that of the F1 engine located in the first stage of the American carrier rocket Saturn V. Together they produce a thrust of 31,392 KN (3,200,000 Kg. The cylidrical blocks of the first stage are joined in pairs on opposing sides of the rocket's second stage fuselage [29].

Every one of the rocket Energia's first stage blocks weighs 347,300 kg, of which 312,575 kg (90%) is constituted by fuel materials composed of kerosene and liquid oxygen. The four blocks together weigh 1,389,200 kg, of which 1,250,300 is constituted by fuel material. The fuel supply contained in the rocket's first stage block is sufficient for operating the engine for 118 seconds [29].

SECOND STAGE

The second stagee occupies the central part of the rocket's fuselage. It has the shape of a cylinder with a length of 60 m and a diameter of 8 m. Its upper end is rounded-off. The central body is propelled by four cryogenic monochamber engines with integrated cycle which develop 140 tons of thrust each on the ground and 200 tons in vacuum, which amounts to 592 tons and 800 tons of thrust for the second stage. These are the first cryogenic engines developed and used in the USSR [28]. The second stage weighs 700,800 kg, of which 651,800 kg (93%) is constituted by liquid hydrogen and liquid oxygen.This supply of fuel material allows the engines to operate for a period of 340 seconds. From a comparison of the stage's weight and the thrust of the engines, it becomes clear that the second stage of the Energia rocket could be used as a type of rocket on its own [29].

MISSION PROFILE

During rocket's take-off both the engines of the first and second stages are ignited. Thus producing a total thrust of 39,240 kg, which is larger than that produced by the American carried rocket Saturn-V. Since the engine thrust exceeds by more than two times the rocket's weight, this insures large acceleration of motion and easy lift-off from the launching pad. At the same time the fuel material is used more economically.

After 118 seconds of flight the fuel material contained in the rocket's first stage blocks is used up and at that point they fall off. This happens at an altitude of 40 km. The rocket's speed at that time is 3,500 m per seconds (calculated). The actual speed is lower by the difference due to all kinds of losses [29].

However, the second stage engines are still in operation (for 222 seconds), and due to their output the rocket can reach a speed of 9100 m per seconds (calculated). In reality the speed reached by the Energia rocket's second stage is slightly lower than the satellite speed, since many kinds of losses have to be subtracted from the calculated velocity (above all gravitational and aerodynamic losses). But this is intended; the second stage should not reach satellite speed and return to Earth. Therefore, the usable load on the Energia rocket must have a small additional rocket stage allowing it to reach satellite speed.

To meet high requirements on reliability and "vitality", the design and development of the Energia launch vehicle is based on the principle: one failure in any system does not affect the fulfilment of the program; the second failure in the system does not affect the flight safety [30].

IMPROVEMENT

To expand the Energia launch vehicle capabilities to inject heavy spacecraft into low - , medium- and high-altitude orbits, including geostationary one, plans are underway to:

- design and development of an oxygen/hydrogen booster stage 5.5 m in diameter, containing fuel of about 70 tons and a high specific thrust engine; this will make it possible to put payloads up to 18 tons into geostationary orbit and boost 32 tons payloads into translunar trajectory and 28 tons payloads into flight trajectories to Mars and Venus.

- design and development of a special small transport module for injecting laarge-scale payloads of orbital stations and platforms into orbits up to 1000 km.

- design and development of a special cargo module to be mounted on the launch vehicle instead of the orbiter (Buran), the module will accomodate cargos of 5.7 m in diameter and 37 meters in length [30].

That upper stage described above will be an all-cryogenic system, burning liquid hydrogen with liquid oxygen (total propellant mass 70 tons). Thrust is relatively low at 100 kN (10.2 tons), but claimed performance figures demand an usually high specific impulse of 490 seconds. The engine can be restarted up to ten times, and it would seems that such multiple burns will be needed to get the suggested payload (above) out of low orbit. Prime role of the stage will be injection of payloads into high orbits (including GEO) and trans-lunar and trans-planetary trajectories [31].

Also a Retro and Correction Stage (RCS) will be incorporated as part of the Energia Upper Stage. The RCS is a liquid oxygen/hydrocarbon (kerosene) enginee developing 85 kN (8.66 tons) thrust, with separate fuel and oxidizer tanks holding a total propellant mass of between 111 and 15 tons. The engine is able of being restarted up to seven times, and total operating lifetime is up to two years. It has a dimension of 5.5 m in length and 3.7 m in diameter [31]. Table 3.4.1 shows the characteristics of the Energia upper stage engines.

Table 3.4.1 ENERGIA UPPER STAGE ENGINES

	RETRO & CORRECTION STAGE	ENERGIA UPPER STAGE
Dry mass, (tons)	2.0	7.0
Maximum useful propellant mass, (tons)	15.0	70.0
Main engine maximum vacuumthrust, kN	85.0	100.0
Main engine specific impulse, sec	352.0	490.0
Maximum number of engine starts	7.0	10.0
Maximum engine operating life in space	2.0 years	4.0 days

PROJECTED PERFORMANCE FOR ENERGIA

PAYLOAD:

- Geostationary orbit : 18 tons
  - Mars : 28 tons
  - Venus : 28 tons
  - Moon : 32 tons
  - Low Earth orbit : 100 tons
  - Jupiter : 6 tons

3.5 BURAN

BACKGROUND ON THE VEHICLE

The Buran design is that of a "tailless" type with a low-placed triangular variablee sweptback wing served as the basis for ship construction. Buran makes an impression due to its inspiring size. Buran total length is 36.4 m, its wing span is 24 meters and its standing height is 16.5 meters. It has a 5.6 meters in diameter and a wing area of 250 square meters. Its load compartment could be a good railroad car which can freely hold a payload with a weight up to 30 tons to deliver to orbit and can return up 20 tons to Earth. The total launching weight of the ship is 105 tons and its landing weight is 82 tons [32].

Buran's body is not pressurized. It can be divided arbitrarily into three compartments: prow, middle (payload compartment), and tail. In the prow compartment there is a pressurized fully welded inserted cabin with a total volume of more than 70 cubic meters which in the future will hold the crew and the main part of the apparatus, ensuring the ship's flight in a rocket-space complex, autonomous flight in orbit, descent and landing.

On the outer side of the body there is a specially developed heat insulating covering. Two types of covering are used, depending on the place to which it is applied on the body in form of "tiles" fabricated from a superfine quartz fiber and flexible elements of high-temperature organic fibers. A construction material based on carbon is used for the most heat-stressed sectors of the body surface, such as the wing edges, the prow cap and leading edge of the keel [32]. The adopted design of the thermal shielding provides for the installation of a total of about 39,000 ceramic tiles fabricated on lathes under specially formulated programs. In the course of passage through the dense layers of the atmosphere the temperature exceeds 1600 on individual "windward" surface due to drag, whereas the temperature of the body covering must not exceed 150 degrees [32]. It is only because of this tiles that the spaceship, which is made of aluminum alloys, steel and titanium, does not burn up while returning to Earth. Buran behaves like a conventional aircraft in the atmosphere. It is directed by ailerons, a rudder and air brakes. If necessary, Buran can accomplish a lateral offset of 2000 km. The ship can take a crew of two to four plus another six passengers on board. Its cargo bay is 18 meters long and 4.7 meters in diameter. The volume of the crew's cockpit is about 70 cubic meters. The cockpit has six windows with an overall glass area of 2.25 square meters [32].

Buran is carried into space by the Energia general purpose booster rocket; once in orbit it can perform all the traditional operations characteristics of the conventional spacecraft, but in descending through the Earth's atmosphere it functions like a regular aircraft and lands at a speed of roughly 340 km/h, just like one of our modern fighters [33].

PROPULSION

The Energia superlauncher alone provides the lift-off thrust for Buran. The four first stage boosters, each powered by a single RD-170 four-chamber engine, developing a vacuum thrust of 806 tons and 740 tons at sea level burning oxygen and kerosene, burns for about 2.2 minutes. Specific impulse is 308 seconds at sea level, 336 seconds in vacuum. The exhaust nozzles can be gimballed through 3.5 degrees for thrust vectoring. The four strap-ons separate initially in pairs and then each pair splits spart prior to ground impact [34].

The four single-chamber cryogenic core engines (which could be considered as the second stage) each develops a thrust of 200 tons in vacuum and 148 tons at sea level. The ratio of sea level to vacuum thrust is greater than normal because the engines have been optimized for operation at altitude. Specific impulse for this stage is 458 seconds and nozzle vectoring movement is 8.6 degrees. The core engines are ignited on the ground to give higher launch acceleration and reduce the chances of a launch abort due to non-ignition of one or more engines. These engines burn for about eight minutes and the core splashes-down in the Pacific Ocean about 40 minutes after the launch [34].

Energia booster rocket does not put Buran into orbit but completes the active segment by creation of conditions for the realization of entry of the Buran into orbit by Buran's own means and at the same time, conditions for sinking of the central body module of the booster rocket into the water of the Pacific Ocean. The Reaction Control System (RCS) thrusters on the Buran fired to move the orbiter away from the depleted Energia core as both vehicles flew eastward over central USSR. Separation occurs at an altitude of 150 km.

In the segment of final insertion into orbit, Buran's combined engine is fired twice. The total time of operation of Buran's service propulsion engines is about 100 seconds in the course of 45 minutes of flight, after which Buran enters a so-called circular reference orbit [32]. Buran carries a combined engine operating on high-energy fuel components (oxygen-hydrocarbon fuel) for putting the vehicle into working orbit, for interobital transfers, for precise maneuvers near serviced orbital complexes, for orientation and stabilization. The engine is constructed in the form of a single assembly (base module) situated in the tail compartment of the body and two "belts" of engines near the front part of the body, in front of the cabin and in the rear part of the tail compartment. All engines are fed from unified fuel tanks. The total fuel supply is about 14 tons [32].

MISSION PROFILE

Lift-off thrust is provided by Energia alone for takeoff. Two and a half minutes later, the Energia-Buran system has already reached an altitude of 60 km. The strap-on boosters separated by sliding down from the booster, rather than laterally, to avoid damaging the wings. The main engine cut off at T + 486 seconds . Eight minutes later, the shuttle separates from the launcher at an altitude of 110 km. Buran separates using its thrusters and glides in a suborbital trajectory until T+36 min 19 seconds, when it performs a 67 seconds OMS 1 burn at an altitude of 160 km. The 42 seconds OMS 2 burn follows at T+47 minutes. Buran finishes up in a 252-256 km orbit, inclined at 51.5 degrees [35]. Table 3.5.1 shows the characteristics of the Buran/Energia configuration [34].

3.5.1 CHARACTERISTICS OF THE BURAN/ENERGIA

	BURAN	ENERGIA	
		CORE	STRAP-ONS (each of four)
Length (m)	36.4	59.0	38.0
Diameter (m)	5.6	8.0	4.0
Wingspan (m)	24.0	---	---
Wing area (m^2)	250.0	---	---
Cabin volume (m^3)	70.0	---	---
Crew (maximum)	10.0	---	---
Dry empty mass (t)	62.0	---	---
Fueled empty mass (t)	75.0	---	---
Payload bay Length (m)	18.0	---	---
Payload bay diam. (m)	4.7	---	---
Payload to orbit (t)	30.0	---	---
Return payload (t)	20.0	---	---
Dry mass (t)	----	85.5	35.0
Propellant mass (t)	14.0	820.0	350.0
Number of engines	----	4.0	1.0
Unit thrust, $_{vacuum}$ (t)	----	200.0	806.0
Specific Impulse (s)	----	470.0	336.0
Burn time (sec)	100.0	480.0	145.
Liftoff thrust* (t)	3,500.0		

* @ sea level, provided by Energia alone

CHAPTER IV

ASIAN AND INDIA LAUNCH VEHICLES

4.1 H-I

BACKGROUND ON THE VEHICLE

The development of H-I launch vehicle began in 1981. In recent years, the need to launch large capacity satellite has been increasing. The H-I launch vehicle is a three-stage rocket developed to meet the needs for launching large geostationary satellite weighing about 550 kg from the latter half of 1980's through earlier half of 1990's [36].

The H-I rocket is a two or three stage rocket approximately 40.3 meters in total length and 2.5 meters in outside diameter; it weighs approximately 140 tons when fully equipped. This rocket can launch a geostationary satellite weighing approximetely 550 kg. The first and second stages use liquid propellant. The trajectory and attitude are controlled by the inertial guidance system from lift-off to second/third stage separation, and the attitude of the third stage is stabilized by spinning [37].

Geostationary satellite can be launched by the three stage rocket. In this case, the rocket's mission ends by injection of the satellite into transfer orbit; thus the satellites should be equipped with apogee kick motor (AKM) which will boost the satellite into drift orbit.

Satellite of medium or low earth mission can be injected into the required orbit by two stage rocket without using the third stage solid rocker motor. The two stage rocket can achieve high accuracy of injection into target orbit. Furthermore, the satellite does not necessarily have to be spinned.

If the satellite weight is light, the number of strap on boosters can be reduced to 6 or several satellites can be launched simultaneously, meeting a variety of mission requirements [37].

The H-I rocket can inject a satellite into a geostationary transfer, medium altitude, elliptical, or earth gravity escape orbit [37].

PROPULSION SYSTEM

First Stage

The first stage of the H-I rocket comprises of one main engine (MB-3), two Vernier engines and up to nine solid strap-on boosters (CASTOR-II) [38].

Second Stage

The second stage propulsion system uses liquid oxygen and liquid hydrogen as propellants, with an average vacuum thrust of 10.5 tons an a vacuum specific impulse of 445 sec. This engine (LE-5) is a high-performance engine, it features a re-ignition capability [38].

Also the second stage has a Reaction Control System (RCS). This system provides thrust needed for the second stage three-axis attitude control and acceleration necessary for settling/retention of the main tank propellant (LOX/LH_2). It consists of two modules installed in the second stage engine section. There are two types, one is for restart mission and the other for non-restart mission [37].

Third Stage

The third stage has a solid-fuel rocket engine which employs polybutadiene composite propellant and high strength titanium alloy motorcase and lighter nozzle, resulting in a light motor with excellent performance [39]. Tables 4.1.1 - 4.1.6 summarize the features of the H-I launch vehicle.

TABLE 4.1.1 CHARACTERISTICS OF H-I MAIN ENGINE

		CHARACTERISTICS OF THE MAIN ENGINE (MB-3)
PROPELLANT		LOX/RJ-1
THRUST	(tons)	77.1 @ sea level
MIXTURE RATIO		2.15
ENGINE CYCLE		Gas generator cycle
COMBUSTION PRESSURE	(kg/cm^2a)	40
SPECIFIC IMPULSE	(s)	253
BURNING TIME	(s)	270
NOZZLE AREA RATIO		8.0
START SYSTEM		Start tank, hypergolic and pyrotechnic igniters

TABLE 4.1.2 CHARACTERISTICS OF THE LE-5 ENGINE

CHARACTERISTICS OF THE LE-5 ENGINE	
THRUST) vacuum	10. 5 tons
Isp) vacuum	450 sec
MIXTURE RATIO	
NON-RESTART MISSION	5.50
RE-START MISSION	5.65
-First burn	332 sec
-Second burn	21 sec
-Coasting	44 minutes
NOZZLE EXPANSION RATIO	140
ENGINE CYCLE	Gas generator cycle
COMBUSTION PRESSURE	37 kg/cm^2a
PROPELLANT WEIGHT	
LOX/LH_2	8.7 tons
DRY WEIGHT	1.8 ton
LOX TANK PRESSURE	3.2 kg/cm^2 a
LH_2 TANK PRESSURE	2.5 kg/cm^2 a

TABLE 4.1.3 CHARACTERISTICS OF THE REACTION CONTROL SYSTEM

	RESTART TYPE	**NON-RESTART TYPE**
Propellant	**Anhydrous hydrazine**	
Total Impulse	10,661 kg.s	2,176 kg.sec
Propellant Weight	54 kg	12 kg
Thrust per one Thruster	1.8 kg	1.8 kg
Total Dry Weight	40 kg	30 kg
Thruster Number	12 (unit)	12 (unit)
Pressure Gas and Type of Propellant Tank	Regulator Type	Blow down type

TABLE 4.1.4 THIRD STAGE SOLID ROCKET MOTOR CHARACTERISTICS

Mass Fraction	0.92
Propellant Isp	291 sec (nominal, propellant delivered)
Motor case	Hight-strenght titanium alloy
Insulation	Ethylene propylene dienee monomer (EPDM)
Propellant	Polybutadiene, aluminum and ammonium perchlorate composite propellant. Maximum 1,900 kg loaded (1,840 kg for the H-I rocket)
Nozzle	Expansion ratio is approximately : 54 Carbon Fiber Reinforced Plastic (exit cone)

TABLE 4.1.5 MAIN CHARACTERISTICS OF H-I

ALL STAGES					
Total length	40.30 (m)				
Diameter	2.49 (m) second stage				
Total Weight	139.3 (m) payload weight not included				
Guidance syst	Inertial guidance system				
	First stage	**Strap on booster**	**Second stage**	**Third stage**	**Payload Fairing**
Length	22.44 (m)	7.25 (m)	10.32 (m)	2.34 (m)	7.91 (m)
Maximum diam.	2.44 (m)	0.79 (m)	2.49 (m)	1.34 (m)	2.44 (m)
Weight	85.8 (tons)	40.3 (tons) 9 boosters	10.6 (tons)	2.2 (tons)	0.6 (tons)
Propellant weight	81.4 (tons)	33.6 (tons) 9 boosters	8.8 (tons)	1.8 (tons)	
Av. thrust	77.1* (ton) 0.9**	135.6 (tons) (for 6 boosters)	10.5 (tons) in vacuum	7.9 in vacuum	
Burning duration	270* (s) 276** (s)	39 (sec)	370 (sec)	68 (sec)	
Propellants	LOX/RJ-1	Polybutadiene composite solid propellant	LOX/LH_2	Polybuta-diene composite solid propellant	
Propellant feed system	Turbo pump		Turbo pump		
Isp.	253* (s) 209** (s)	235 (sec) @ sea level	450 (sec) in vacuum	291 (sec)	
Pitch & Yaw	Gimbal		Gimbal*1 Reaction control*7		
Roll	Vernier engine		Reaction control		

*: Main engine; **: Vernier engine (for 2 engines); *[1]: During powered flight phase; *[2]: During coast flight phase.

TABLE 4.1.6 H-I TYPICAL FLIGHT SEQUENCE

EVENTS	TIME (sec)	ALTITUDE (km)	INERTIAL SPEED (Km/s)
Lift-off (First stage main engine ignition & 6 SRB's)	0.0	0.0	0.4
3 Solid Rocket Boosters Ignition	40.0		
9 Solid Rocket Boosters Separation	85.0	19.0	0.9
First Stage Main Cut-off (MECO)	270.0	100.0	4.0
First/Second Stage Separation	278.0		4.0
Second Stage Engine Ignition	283.0		
Payload Fairing Separation	305.0	122.0	
Second Stage Engine Cut-off (SECO)	640.0 1500.0	176.0	7.8
Third Stage/Satellite spin up	1505.0		
Second/Third Stage Separation	1528.0		
Third Stage Engine Ignition	1596.0	199.0	7.8
Third Stage Engine Burnout (TEBO)	1650.0	201.0	10.3
Spacecraft Separation (injection)		219.0	10.3

4.2 H-II

BACKGROUND ON THE VEHICLE

To accommodate the numerous demands for launching increasingly larger satellites, NASDA initiated in 1986 the development of the H-II rocket, a larger rocket which incorporates Japan's independent technologies throughout [40].

The H-II is designed to serve as NASDA's main space transportation system in the 1990's to meet the demand for a high degree of reliability. It will be capable of sending a single two tons class payload or multiple payloads totaling two tons into geostationary orbit. The H-II is a two-stage rocket equipped with two large solid rocket boosters (SRBs) on the first stage for thrust augmentation [41]. The H-II is also capable of sending a payload into low, medium, and high orbits as well as geostationary orbit and launching space probes to the moon, and planets. It will also be used to launch the H-II Orbiting Plane (HOPE). Hope will be mated to the upper stage of the H-II rocket. It can also be used to recover material from a space station [36].

For example, H-II is capable of sending a payload weighing approximately 10 tons into an orbit of 30 degrees inclination and 300 km altitude, and 2 to 3 tons probe to Venus or Mars. As two one-ton class satellites can be launched simultaneously into geostationary orbit [36].

PROPULSION SYSTEM

First Stage

A new large-scale liquid hydrogen and liquid oxygen engine calles LE-7 was developed for the first stage. The LE-7 is a high performance engine with approximately 120 tons thrust. In order to supplement the thrust of its first stage, two solid rocket boosters (SRBs) with about 180 tons of thrust each. Gimbal control of the LE-7 alone is not enough for the attitude control of the vehicle, so the solid rocket boosters are equipped with a movable nozzle for thrust vector control [36]. The first stage has a total length of 28 meters and a diameter of 4 meters [40].

Second Stage

The second stage engine is an improved version of the restartable engine, LE-5, developed for the HI second stage. The second stage has a dimension of 11 meter long and 4 meters in diameter for LH2 tank and 2.4 meters in diameter for the LOX tank.

The first experimental launch of the H II rocket is scheduled to take place in 1992. Tables 4.2.1 - 4.2.6 summarize the features of the H-II launch vehicle.

TABLE 4.2.1 H-II MAIN ENGINE LE-7 CHARACTERISTICS

CHARACTERISTICS OF THE MAIN ENGINE (LE-7)	
SEA LEVEL THRUST	93.0 (tons)
VACUUM THRUST	120.0 (tons)
Isp $_{,vacuum}$	449.0 (sec)
MIXTURE RATO	6.1
DURATION	320.0 (sec)
ENGINE CYCLE	Staged-Combustion Cycle
COOLING	Regenerative
EXPANSION RATIO	54:1
AUXILIARY ENGINE	
VACUUM THRUST	2 kN (.2 ton)x 2
PROPELLANT BEFORE MECO	GH_2 Bleed
PROPELLANT AFTER MECO	GN_2 Blowdown
USABLE PROPELLANT	86.3 (tons)
LOX	73.7 (tons)
LH_2	12.6 (tons)
COMBUSTION PRESSURE	134.0 kg/cm^2a

TABLE 4.2.2 H-II SOLID ROCKET BOOSTER CHARACTERISTICS

SOLID ROCKET BOOSTER CHARACTERISTICS	
PROPELLANT MASS	59 (tons each)
PROPELLANT COMPOSITION	14% HTPB/18% Aluminum/ 68% AP
THRUST	160 (tons @ sea level)
BURNING DURATION	94 (sec)
CHAMBER PRESSURE	60 Kg/cm^2a MEOP
DIMENSION	1.8 meter in diameter 23 meters in length
SPECIFIC IMPULSE	273 (sec) in vacuum
MOVABLE NOZZLE	Flexible Joint (5 deg)
CASE	4 segments NT-150 low alloy steel

TABLE 4.2.3 H-II LE-5A ENGINE CHARACTERISTICS

CHARACTERISTICS OF THE LE-5A	
PROPELLANT	LOX/LH_2
THRUST	12.4 (tons) in vacuum
MIXTURE RATIO	5.0
ENGINE CYCLE	Hydrogen Bleed Cycle
COMBUSTION PRESSURE	40.0 kg/cm^2a
USABLE PROPELLANT	16.7 (tons)
LOX	13.9 (tons)
LH_2	2.8 (tons)
SPECIFIC IMPULSE	452.0 (sec) in vacuum
BURNING TIME	609.0 (sec)
EXPANSION RATIO	130.0
REACTION CONTROL SYSTEM	Hydrazine Thrusters
RE-STARTABILITY	1^{st} Burn: 310 sec GEO Missions
	2^{nd} Burn: 200 sec GEO Missions
	3^{rd} Burn: 13 min GEO Missions

PAYLOAD FAIRING CAPABILTY	
STANDARD FAIRING (size)	4.1 m diameter x 12 m long
PAYLOAD ENVELOPE	3.7 m diameter x 10 m long
FAIRING FOR PLURAL SATELLITES (size)	4.1 m diameter x 15 m long
FAIRING FOR WIDE-BODY SATELLITE (size)	5.0 m diameter x 12 m long

TABLE 4.2.4 MAIN CHARACTERISTICS OF H-II

ALL STAGES				
Total length	50.0 m			
Diameter	4.0 m			
Total Weight	260.0 tons (payload not included)			
Guidance System	Strap-down inertial guidance system			
	First Stage	**Solid Rocket Boosters**	**Second Stage**	**Payload Fairing**
Overall Length	28.0 m	23.0 m	11.0 m	12 m (Standard)
Diameter	4.0 m	1.8 m	4.0 m	4.1 m (Standard)
Stage Weight	98.0 (tons)	141 (tons) (2 units)	20.0 (tons)	1.4 m (Standard)
Propellant Weight	86.0 (tons)	118 (tons) (2 units)	17.0 (tons)	
Thrust	86.0[*1] (tons)	320 (tons) (2 units)	12.0[*2] (tons)	
Burning time	348.0 (sec)	94.0 (sec)	608 (sec)	
Propellant	LOX/LH_2	Polybutadiene composite solid propellant	LOX/LH_2	
Propellant feed system	Turbo pump		Turbo pump	
Specific Impulse	445[*2] (sec)	273 (sec)	452[*2] (sec)	
Pitch & Yaw	Gimbal	Gimbal	Gimbal[*3] RCS[*4]	
Roll	Auxilliary engine		Reaction Control System	

[*1]: @ Sea Level; [*2]: In vacuum; [*3]: During powered flight phase; [*4]: Reaction Control System: (during coast flight phase)

TABLE 4.2.5 PAYLOAD CAPABILITY OF THE H-II LAUNCH VEHICLE

PAYLOAD WEIGHT	Geostationary orbit, Single launching approximately 2 tons (approximately 4 tons into geostationary transfer orbit)
	Low Earth orbit (h = 300 km, i = 30 deg.) approximately 9 tons
	Sun synchronous orbit (h = 700 km) approximately 5 tons
	Moon or planetaryy mission approximately 2 to 3 tons
ALLOWABLE VOLUME OF SATELLITE	3.7 m diameter x 10 m long Diameter will be enlarged to 4.6 m for the Space Shuttle compatibility.

TABLE 4.2.6 H-II TYPICAL FLIGHT SEQUENCE

EVENTS	TIME (sec)	ALTITUDE (km)	INERTIAL SPEED (m/s)
First Stage Engine (LE-7) Ignition	- 6.0	0.0	0.0
Solid Rocket Boosters (SRBs) Ignition	0.0	0.0	0.0
Lift-off	0.0	0.0	0.0
SRBs Burn out	97.0	36.0	1,763.0
SRBs Separation	102.0	40.0	1,792.0
Fairing Jettison	280.0	140.0	4,610.0
First Stage Engine Cutoff (MECO)	320.0	161.0	5,672.0
First Stage Separation	328.0	167.0	5,664.0
Second Stage Engine Ignition	338.0	173.0	5,654.0
Second Stage Engine Cutoff	646.0	252.0	7,756.0
Second Stage Engine Restart	1425.0	246.0	7,761.0
Payload Separation	1625.0	273.0	10,177.0
Injection into geostationary transfer orbit	1645.0	286.0	10,166.0

4.3 LONG MARCH

BACKGROUND ON THE VEHICLE

Rockets are anything but new to the Chinese. Tang Fuxian invented them in 1000 AD, and the infamous Kublai Khan fired rockets against the Japanese in 1275. China even developed a primitive two stage missile in the 15th century [42].

The modern Chinese rocket programme began during the late 1950s with the development of ballistic missiles. In 1964 China made its first launch vehicle. The Chinese began their satellite programme in 1970 with the successful launch of Tungfanghung (China I). Since that time, they have successful placed 23 satellites in orbit with a success rate of about 85%.

The Chinese entered the commercial space industry in 1987 with their Long March. The Long March is a serious contender in international markets [42]. The Long March vehicle program was initiated during the 1960s. The Long March vehicle was developed on the basis of medium- and long-range rockets. The operational Long March vehicles now include Long March 1D (CZ-1D), Long March 2E (CZ-2E), Long March 3A (CZ-3A), Long March 3B (CZ-3B) and Long March 4A (CZ-4A) [43].

LONG MARCH 1D

Long March 1D is a three-stage rocket. The lift off mass of the CZ-1D is 80 tons. Total length is 28.22 m. The first two stages have a diameter of 2.25 m and are the modified version of an intermediate range rocket [43]. The payload and third stage fairing is 2.05 m in diameter. Stages 1 and 2 are liquid rockets that use nitrogen acid (NA 27S) and unsymmetrical dimethydrazine (UDMH) as propellants. Stage 3 is a solid rocket. CZ-1D is used mainly to launch small space payloads [42]. This launch vehicle has a strap-down inertial guidance system. Jet vans are used in 1st and 2^{nd} stages to produce the control forces. After the main engine cut-off of the 2^{nd} stage, the vehicle flight in a ballistic coast phase. A nitrogen gas nozzle system is used to maintain attitude control and to minimize the ignition attitude error of the 3^{nd} stage. The 3^{nd} stage is spin stabilized during its powered phase [44]. Tables 4.3.1 - 4.3.2 show the features of the CZ-1D configuration.

TABLE 4.3.1 LONG MARCH 1D PAYLOAD CAPACITY

INCLINATION (degree)	STABILISATION	PAYLOAD MASS (kg)
28.5	3-axis	740 kg
28.5	Spin	720 kg
57.0	3-axis	600 kg
57.0	Spin	790 kg
70.0	3-axis	550 kg
70.0	Spin	740 kg

TABLE 4.3.2 LONG MARCH 1D CHARACTERISTICS

Total Liftoff mass (t)			80.00
Total Length (m)			28.20
Diameter (m)			2.25
	1st Stage	2nd Stage	3rd Stage
Propellants	NA27S/UDMH	N_2O_4/UDMH	SOLID
Propellant mass (t)	61.4	12.2	1.574
Thrust (t)	112.3	10.0	----
Specific Impulse (sec)	241.0	300.0	291.0
Total mass (t)	65.5	15.0	1.83
Engine	YF-2A	YF-3	IRIS

LONG MARCH 2E (CZ-2E)

Long March 2E is a 21/2 stage rocket. Its total lift off mass is 460.8 tons. Overall length is 51 m. The nominal diameter of the core and second stage is 3.35 m. There are four liquid-fueled, strapped-on boosters, each with a diameter of 2.25 m and a length of 15 m attached to the first stage. The propellants used for all stages are nitrogen tetraoxide (N_2O_4) and UDMH [43]. The first stage has four YF-20 engines, which provide a combined thrurst of 300 tons. Each engine can be gimbaled in one direction. The engines for the second stage are combined by one YF-22 four YF-23 vernier engines. The guidance system of the vehicle is an inertial platform/ computer system [44].

CZ-2E is designed for lifting heavy payloads into elliptical and circular low Earth orbits- a payload of 8,800 kg can be placed into a 200 km circular orbit at an inclination of 28.5 degrees.

The CZ-2E can also carry spacecraft with their own perigee stage. The launch vehicle is compatible with Mc Donnell Douglas PAM upper stage system and with the Thikol Star 63F perigee kick motor. Such compatibily gives the launch vehicle the capability to place spacecraft of over 3,000 kg into GTO [45]. Table 4.3.3 shows the characteristics of this configuration.

TABLE 4.3.3 LONG MARCH 2E CHARACTERISTICS

Total Liftoff mass (t) Total Length (m) Diameter (m) Payload capacity (kg)			460.8 51.0 8.0 8800 to LEO
	Strap-on Boosters	**1st Stage**	**2nd Stage**
Propellants	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Propellant mass (t)	4 x 38	187.0	86.0
Thrust (t)	4 x 75	300.0	79.8
Specific Impulse (sec)		260.0	296.0

LONG MARCH 3A (CZ-3A)

Long March 3A is a three stage rocket. The first and second stage are modified versions of the Long March 2 vehicle. The load mass of the first stage is increased by 30 to 40 tons. The third stage is a newly developed cryogenic rocket with a diameter of 3.0 m. The lift off mass is 240 tons. Sea level thrust of the first stage is 300 tons. Total length is 52.6 m [43]. The third stage is powered by YF-73 engine, which has four thrust chambers. The YF-73 produces 4.5 tons of thrust [44].

The Long March 3A will be commercially available in 1992. It is designed to place one or two spacecrafts into GTO. It is also capable of placing payloads into elliptical and circular low Earth orbits, and into SSO.

The standard CZ-3A) mission is to place a spacecraft with a mass of 2,500 kg into a geosynchronous transfer orbit with the following characteristics [45]:

Perigee: 200 km

Apogee: 35,786 km

Orbit inclination: 28.5 degrees

Argument of perigee: 179.6 degrees

A new guidance and control system is used for CZ-3A, which consists of a four-frame inertial platform and a computer [43].

LONG MARCH 3B (CZ-3B)

Long March 3B is a 31/2 stage rocket. The core stages of CZ-3B are the same with CZ-3A. Two or four liquid boosters are used to augment the first stage performance. The lift off mass of CZ-3B is 403 tons [43]. Tables 4.3.4 and 4.3.5 show the characteristics of both CZ-3A and CZ-3B.

TABLE 4.3.4 LONG MARCH 3A CHARACTERISTICS

Total liftoff mass (t) Total Length (m) Diameter (m) Payload capacity (kg)			239.80 52.60 3.35 2500 to GTO
	1st Stage	**2nd Stage**	**3rd Stage**
Propellants	N_2O_4/UDMH	N_2O_4/UDMH	LOX/LH_2
Propellant mass (t)	170.0	31.0	18.2
Thrust (t)	300.0	79.8	16.0
Specific Impulse (sec)	260.0	296.0	440.0
Engine	YF-20	YF-22	----

TABLE 4.3.5 LONG MARCH 3B CHARACTERISTICS

Total Liftoff mass (t) Total Length (m) Diameter (m) Payload Capacity (kg)				402.8 52.6 8.0 4500 to GTO
	Booster Phase	**1st Stage**	**2nd Stage**	**3rd Stage**
Propellants	N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH	LOX/LH_2
Propellant mass (t)	4 x 38	170.0	31.0	18.2
Thrust (t)	4 x 75	300.0	79.8	16.0
Specific Impulse (sec)	-----	260.0	296.0	440.0

LONG MARCH 4A (CZ-4A)

Long March 4A is a three stage rocket. All stages use N_2O_4/UDMH storable propellants. The lift off mass of CZ-4A is 240 tons. The total length of the rocket is 41.9 m. The diameter of the first and second stage is 3.35 m, and 2.9 m for the third stage [43].

The first stage is the equivalent to the first stage of the Long March 3, but with the propellant tanks lengthened. It has a thrust of 2,971 kN on the ground. The second stage is the same as that used for the Long March 3 [45]. The propellant loading mass is 180 tons. The third stage is a newly developed upper stage with two gimbaled engines and a common bulk-head tank [43]. The third stage delivers a thrust of 100 KN in vacuum [45].

The CZ-4 is designed primarily for SSO missions, and can place a 1,500 kg payload into a 900 km SSO. The launch vehicle is also capable of placing payloads into elliptical and circular orbit. (For example, a payload of 2,700 kg can be placed into a 600 km circular orbit, or a payload of mass up to 3,800 kg into an elliptical orbit of 200 km perigee [45].)

The CZ-4 can also be used to loft piggy-back payloads contained in a transition bay between the primary payload and the vehicle. Table 4.3.6 shows the characteristics of the CZ-4A.

TABLE 4.3.6 LONG MARCH-4 CHARACTERISTIC (CZ-4)

Total Liftoff mass	(t)			240.00
Total Length	(m)			41.90
Diameter	(m)			3.35
Payload Capacity	(kg)			1500 to SSO
		1st Stage	**2nd Stage**	**3rd Stage**
Propellants		N_2O_4/UDMH	N_2O_4/UDMH	N_2O_4/UDMH
Propellant mass	(t)	177.0	35.5	8.5
Thrust	(t)	300.0	78.1	10.0

4.4 Polar Satellite Launch Vehicle (PSLV)

BACKGROUND ON THE VEHICLE

The Indian Satellite Launch Vehicle development programmes started in modest way about a decade ago, and the first successful launch was achieved in 1980, when a 40 kg Rohini satellite was orbited using a developed SLV-3 vehicle. The objective of the PSLV mission is to inject one ton class "remote sensing spacecraft" into a 900 km Sun-synchronous polar orbit [46].

The PSLV has a basic stage diameter of 2.8 m, and is 45 meters long with the lift-off weight of 275 tons [46].

FIRST STAGE

The PSLV first stage is a solid motor of 2.8 m diameter and 20 meters long with a propellant loading of 125 tons, produces a maximum thrust of 4500 kN. Six solid strap-ons boosters of one meter diameter out of which four are ground lit and the remaining 2 are air-lit. The high energy HTPB propellant used for the booster motor are cast in 5 segments individually, and assembled together with a dry joint. The propellant grain and insulation system is designed to give the M-type thrust time curve. The igniter is of pyrogen type and is located at the head end of motor. The pitch and yaw control is provided by secondary injection thrust vector control system and roll control is by two number of swivellable bipropellant reaction control thrusters. The nozzle is a convergent divergent type with carbon-phenolic and silica phenolic ablative liners [46].

SECOND STAGE

The second stage is a liquid stage with 2.8 m in diameter and 10 meters long, carries 37.5 tons of hypergolic propellants N_2O_4 & UDMH. It has a high thrust gimballed engine. It has a convergent divergent contour nozzle with an area ratio of 31 to produce a maximum thrust of 700 kN. The thrust chamber and nozzle are made of high temperature material. On-off hot gas reaction control thrusters mounted on the interstage skirt, provides the roll control for the stage [46].

THIRD STAGE

The third stage has a high performance solid motor incorporating flex nozzle for control. This motor has a nominal propellant loading of 7 tons and burns approximately 80 seconds producing maximum thrust of 300 kN. The motor case is light weight kevlar epoxy. The nozzle is submerged contoured type and is provided with conical flex seal bearing for the thrust vector control of the stage. The nozzle throat material is of graphite and other portions are made of carbon phenolic and silica phenolic liners, supported by fibre reinforced composite material [46].

FOURTH STAGE

The fourth stage is a liquid propulsion system using 2 tons of propellant, N_2O_4 and MMH, and has two regeneratively cooled engines with multiple start capability. The two thrusters with a maximum area ratio of 60, produce a thrust of 7 kN each [46].

The stage control during the powered phase is achieved with gimballing the engine in two planes by electromechanical actuators. During the coast phase the attitude control is established by 6 on-off bipropellant Reaction Control (RC) thrusters [46].

The vehicle equipment bay consisting of inertial guidance and control system are housed around the fourth stage to have guidance and tracking capability with the associated ground segments till the spacecraft injection into orbit.*

FLIGHT SEQUENCE AND TRAJECTORY

4 strap-on motors are ignited on the ground together with the solid engine for lift-off. The other two are ignited at T+25 seconds. The ground lit are jettisoned at T+61 sec. after burn out at an altitude of 21 km, and the first stage separation occurs at T+96 at an altitude of 56 km. Before the separation of the first stage, the ullage rockets on the second stage are ignited to ensure positive acceleration at liquid stage ignition. The retro rockets are ignited immediately after separation to decelerate the separated first stage. The second stage burns for 150 sec. and takes the vehicle to an altitude of 243 km at burn out. After separation of the second stage, the third stage is ignited at T+250 sec. at an altitude of about 245 km. The third stage motor burns for 90 sec. at the end of which an altitude of 400 km is reached. There is a long

coast phase after third stage burn out and separation. At the end of coast phase, at 800 km altitude, fourth stage ignition takes place. The fourth stage engines burn for about 400 sec. taking the vehicle to an altitude of 904 km, 18 minutes after lift off [47]. Table 4.4.1 shows the propulsion characteristics of the PSLV [47].

TABLE 4.4.1 PSLV PROPULSION SYSTEM

	1st Stage	**2nd Stage**	**3rd Stage**	**4th Stage**
Dimension (m)	2.8 x 20.0 long	2.8 x 11.0 long	2 dia. x 3.5 long	1.34 x 2.6 long
Engine	1 Solid motor	1 liquid engine	1 Solid motor	2 liquid engines
Propellant	HTPB	UDMH & N_2O_4	HTPB	MMH + N_2O_4
Propellant mass (t)	125 + 6 solid booster (8.6t Propellant) for thrust augmentation	37.5	7.0	2.0

CHAPTER V

ORBITAL MECHANICS

5.1 Ground trace

The objective of this chapter is to present the main equations used in a ground trace (track) process, also, it will carry out a typical ground trace calculation for a chosen launching site. The ability to ground track a space vehicle while in orbit is a very important tool. It enables someone to accurately pinpoint at any moment in time what part of the world (country) the vehicle is crossing over. In the case of a satellite, one can know precisely when there is a viewing window over a certain country. While knowing the orbital elements of a satellite enables you to visualize the orbit and its orientation in the IJK inertial reference frame, it's often important to know what the ground track of a satellite is. One of the most valuable characteristic of an artificial earth satellite is its ability to pass over large portions ot the earth's surface in a relatively short time. As a result it has tremendous potential as an instrument for scientific or military surveillance.

Ground Track on a Nonrotating Earth

The orbit of an earth satellite always lies in a plane passing through the center of the earth. The track of this plane on the surface of a nonrotating spherical earth is a great circle. If the earth did not rotate the satellite would retrace the same ground track over and over [48].

Effect of Earth Rotation on the Ground Track

The orbital plane of a satellite remains fixed in space while the earth turns under the orbit. The net effect of earth rotation is to displace the ground track Westward on each successive revolution of the satellite by the number of degrees the earth turns during one orbital period.

Therefore, instead of retracing the same ground track over and over a satellite eventually covers a swath around the earth between latitudes north and south of the equator equal to the inclination. A global surveillance satellite would have to be in a polar orbit to overfly all of the earth's surface.

If the time required for one complete rotation of the earth on its axis (23 hrs 56 min) is an exact multiple of the satellite's period then eventually the satellite will retrace exactly the same path over the earth as it did on its initial revolution. This is a desirable property for a reconnaissance satellite where you wish to have it overfly a specific target once each day. It's also desirable in manned spaceflights to overfly the primary astronaut recovery area at least once each day [48].

GROUND TRACE PROCEDURE

Suppose we assume a launch from Kennedy Space Center on July 31, 1989 @ 10:AM CDT. We want to ground trace the satellite for one revolution. We want to know the latitude, longitude, nodes, perigee, apogee, most northern and southern point as well as $\vec{r}$ and $\vec{V}$.

The launch parameters are:

1. δ= Latitude = 28.317510
2. λ= Longitude = 279.457745
3. Vo = 26,500 ft/s
4. ϕ = Flight path = 4°
5. A = Heading = 80°

A few definitions are in order:

Flight path angle: It's the angle between the velocity vector and the horizontal plane. It's also called flight path elevation angle.

Julian Date: It's a concept of fundamental importance in the reckoning of time, denoted by J.D. It's nothing more than another arbitrary benchmark that is a continuing count of each day elapsed since some particular epoch. This epoch was arbitrarily selected as January 1, 4713 B.C. Each Julian Date is measured from noon to noon, and hence, is an exact integer 12 hours after midnight [49].

Greenwich Sidereal Time = θg= It's the angle between the unit vector I (vernal equinox direction) and the Greenwich meridian.

Local Sidereal Time = θ = It's the sum of θg and λ.

Canonical Units

It's nothing more than a normalized system of units used by astronomers primarily for the purpose of simplifying the arithmetic for orbit calculations. In that unit system, the distance unit (DU) is defined to be the radius of the reference orbit. The time unit (TU) is defined such that the speed of a satellite in the hypothetical reference orbit is DU/TU.

Standard Classical Orbital Elements:

$$\text{Shape}\begin{cases} a = & \text{Semimajor axis of the orbit} \\ e = & \text{The eccentricity of the conic section} \end{cases}$$

$$\text{Orientation} = \begin{cases} i = & \text{(Orbital inclination): It's the angle between the orbit and the equatorial planes.} \\ \omega = & \text{(the longitude of the ascending node): It's the angle measured in the equatorial plane between the principal axis (Vernal equinox) and the line defining the intersection of the equatorial and orbit planes.} \\ W = & \text{(argument of perigee): The angle measured in the orbit plane from the line defined by the longitude of the ascending node to another line in the orbit plane which contains the focus and passes through perifocus.} \\ Tp = & \text{Time of perigee passage.} \end{cases}$$

Now let's find the Julian date [50]:

10:00 A.M CDT = 9:00 A.M CST + 6 = 3:00 P.M = 15:00 GMT

$$J.D = 367 \cdot y - \left(7 \cdot \frac{\left(y + \left(\frac{M+9}{12}\right)\right)}{4}\right) + \frac{275 \cdot M}{9} + D + 1721013.5$$

Where: y = year ; M = month ; D = day

Let's calculate the local Sidereal time [50]:

$\theta = \theta_g + \lambda$

In order to find θ, let's find the Greenwich Sidereal time at 12 midnight [50] or 0^{hr} U.T.

$\theta_{go} = 99°.6909833 + 36000°.7689 \cdot T^2 u + 0°.00038708 \cdot T^2 u.$

Where $Tu = \frac{J.D - 2415020.0}{36525} = 0.89578$; time measured in centuries

Substitute Tu in $\theta_{g0} \longrightarrow \theta_{g0} = 308°.46$

Now since 15^{hr}:00 = 900 minutes, thus, the Greenwich sidereal time is:

$$\theta = \theta_{g0} + (t - t_o) \cdot \frac{d\theta}{dt}$$

Where $\frac{d\theta}{dt} = 1 + \frac{1}{365.2482}$ revolution/year

$= 0.25068447 \cdot 10^{-1}$ degree/minute (since we know that there is one extra sidereal day for every tropical year). One tropical year is equal to 365.2422 mean solar days. Thus, $\theta = 174°076$

Now the local sidereal time can be found:

$\theta = \theta_g + \lambda_E = 174.076 + 279.457745 = 93°5260$ (LST). Note that θ has to be $\leq 360°$. Whenever it's $> 360°$, the answer has to be minus by 360°.

From now on θ will be replaced by a new symbol in order to avoid confusion with our regular θ : α = LST.

Let's find the longitude of the ascending node, ω

$$\omega = \theta_o - \Delta\lambda_o$$

$$\tan \Delta\lambda = \tan A_{Zo} \sin \delta_o$$

$$\Delta\lambda = 69°609$$

$$\omega = 23°917$$

<u>Orbital Element Calculation</u>

Let's change the given data in canonical units:

$$V_o = \frac{26,500}{Du} = \frac{26,500}{25963.247} = 1.020674\ Du/Tu$$

$$r_o = 1 + \frac{\text{altitude}}{Du} = 1 + \frac{275n.m}{3443.922786} = 1.0798Du$$

$a)$

Let's find h [48] (specific momentum)

$$h = rv\cos\phi = (1.0798)\cdot(1.020674)\cdot\cos 4° = 1.099\ Du^2/Tu$$

b) Let's find p (parameter or ''semi-latus rectum'')

$$p = \frac{h^2}{\mu} = \frac{(1.099)^2}{1} = 1.2078$$

c) Let's find ε (specific mechanical energy)

$$\varepsilon = \frac{V^2}{2} - \frac{\mu}{r}$$

$$= \frac{(1.020674)^2}{2} - \frac{1}{1.0798} = -0.40521Du^2/Tu^2$$

d) Let's find a (semimajor axis)

$$\varepsilon = -\frac{\mu}{2a} = -0.40521$$

$$a = -\frac{\mu}{2\varepsilon} = -\frac{1}{2\cdot(-0.40521)}$$

$$a = 1.234Du$$

e) Let's find e (eccentricity)

$$e = \sqrt{1 - \frac{p}{a}} = \sqrt{1 - \frac{1.2078}{1.234}}$$

$$e = 0.1457$$

f) Let's find ν (true anomalie)

$$r_o = \frac{p}{1 + \text{ecos}\nu} \longrightarrow 1.0798 = \frac{1.2078}{1 + .1457 \cos \nu}$$

$\nu_o = 35°55$

Now let's check the quadrant:

$\cos \nu_o = + \longrightarrow$ I or IV

$\phi_o = + \longrightarrow$ I or IV

$\longrightarrow \nu$ is first quadrant

g) Let's find i (orbital inclination)

$\text{cosi} = \text{sinA}_{\text{Zo}} \cdot \cos \delta_o$

$= \sin 80° \cos 28°317510$

i = 29°89

h) Let's find u (argument of latitude)

$\text{cotu} = \text{cosA}_{\text{Zo}}\text{cot}\delta \quad = \cos 80° \cot 28°317510$

u = 72°138

i) Let's find w (argument of perigee)

w = u - ν = 72°138 - 35°55

= 36°588

j) Let's find E_o (Eccentric anomaly)

$$E_o = \frac{(e + \cos \nu_o)}{(1 + e \cdot \cos \nu_o)} \cos^- 1 \quad = 30°947$$

k) Let's find T_p (time of perifocal passage)

$M_o = n(t - T_p) = E_o - e\sin E_o$ = mean anomaly

$n(0 - T_{op}) = .540 - .1457 \cdot \; sin30.947$

$-T_p = \frac{0.46507}{n}$

where n = $\frac{\sqrt{\nu}}{a^3}$ = .72950

T_p = - 0.6375 Tu = - 8.5747 minutes.

Calculation of Longitude and Latitude @ Perigee

u = w + ν

@ perigee ν = 0   and w = 36°588

thus, u = w = 36°588

$\sin\delta = \sin u \sin i \quad = \sin 72.138 \sin 29.89$ = 0.4743

$\delta_p = 0.4743 \cdot \sin^{-} 1$ = 28°315

E_p = 0 ; ν_p = 0

t = $\frac{1}{n \cdot (E - esinE)} + T_p$ = 107 minutes + 21 secondes

$A_Z = \frac{\sin_{-1} \cdot \cos i}{\cos\delta}$ = 79°70

$\Delta\lambda = \mathrm{Tan}^{-1}\mathrm{Tan}A_Z\mathrm{Sin}\delta$ = 69° ; Quadrant check okay!

$\alpha = \Delta\lambda + \Omega$ = 148°869

$\lambda_{Ep} = \alpha - GST - W_\theta t$ = Longitude at perigee

where W_θ = 15°.041/hr = 0.250685

$\lambda_{Ep} = 307°9$

3. Calculation of Longitude and Latitude @ Apogee

@ apogee $\nu_{ap} = 180°$

u = w + ν $= 36°588 + 180° \quad = 216°$

$\mathrm{Sin}\delta_{\mathrm{ap}} = \mathrm{sinisinu}$

$\delta_{ap} = 342°72$

$E_{ap} = \nu_{ap} = 180°$

$n \cdot (t_{ap} - T_p) = E - \text{esinE}$

$t_{ap} = \frac{E\text{-esinE}}{n} + T_p$

t_{ap} = 3.66898*13.45 = 49.3478 minutes

$A_Z = \frac{\sin^{-1}\text{cosi}}{\cos\delta}$ = 65°22

$\Delta\lambda = \text{Tan}^{-1}\text{tanA}_Z\sin\delta \ = 327°25$

$\alpha_{ap} = \Delta\lambda + \Omega \ = 47°1$

$\lambda_{ap} = \alpha - \theta_g - W_\theta t \ = 220°65$

The same procedure will be followed to generate the attached table. ν will be chosen from a range of 0° to 360° @ an increment of 30°.

<u>Calculation for most Southern Point:</u>

$A_{Zs} = 90°$

δ = -i = 330°1

$u_s = 270°$

$\Delta\lambda_s = 270°$

$\nu = 233°41$

<u>Calculation for most Northern Point:</u>

$A_{Zn} = 90°$

$u_n = 90°$

$\delta = 29°89$

$\Delta\lambda_n = 90°$

$\nu_n = 53°41$

Calculation for the Descending Node:

@ the descending node:

$\delta = 0°$; u = 180° ; $\Delta\lambda = 180°$

$\nu_\vartheta = 180$ - w $= 143°41$

$\theta_\vartheta = 180 + \Omega \ = 259°83$

$\lambda_E\vartheta = \theta_\vartheta - \theta_{GM1} \ = 259.83 - 174.076 \ = 85°75$

Calculation for $\vec{r}_o; \vec{V}_o$

$x = r_o \cos\delta\cos\theta_o =$ 1.0798cos28.317510cos93.5260 = - .0585

$y = r_o \cos\delta\sin\theta_o =$ 1.0798cos28.317510sin93.5260 = - .9488

$z = r_o \sin\delta =$ 1.0798sin28.317510 = 0.5122

$\vec{r}_o = -.0585\vec{i} + .9488\vec{j} + .5122\vec{k}$

$v_s = -v\cos\phi_o\cos A_{Zo}$ = - 1.0207cos4cos80 = -0.1768

$v_e = v\cos\phi\sin A_{Zo}$ = 1.0207cos4sin80 = 1.003

$v_r = v\sin\phi_o$ = 1.0207sin4 = 0.0712

$\dot{X} = v_s \sin\delta\cos\theta - v_e \sin\theta + v_r \cos\delta\cos\theta$ = -.1768sin28.3cos93.5 - 1.00sin93.5

+ .071cos28.3cos93.5260

= -.99969

$\dot{Y} = v_s \sin\delta\sin\theta + v_e \cos\theta + v_r \cos\phi\sin\theta$

= -.1768sin28.3sin93.5 + 1.003cos93.5260 + .0712cos4sin93.5260

$= -.07448$

$\dot{Z} = -v_s\cos\delta + v_r\sin\delta$ $= .1768\cos 28.3175 + .0712\sin 28.31750$ $= 0.1894$

$\underline{\vec{V}_o = -.99969\vec{i} - .07448\vec{j} + .1894\vec{k}}$

Table 5.1.1 shows the data resulting by varying E_o from 0 degree to 360 degrees and used the same equations shown before to calculate the parameters shown in table 5.1.1. Figure 2.0 is based on the data obtained from table 5.1.1. Figure 3.0 is the known ground trace case provided by NASDA [1] for verification purpose and figure 4.0 is the verification of the fortran code retracing the same trajectory path as the one from NASDA [1]. It needs to point out that only the first pass was mapped out since it was decided to tie up the ground trace with the mission profile of the launch vehicle selected in order to make the entire process complete (from launch to the first pass of the spacecraft in orbit.)

TABLE 5.1.1 ORBITAL GROUND TRACE TABLE

ν	u	δ	E	t	Λ_z	$\Delta\lambda$	α	λ_E
0	36°588	28°315	0	7.97109	79°70	69°	148.°869	307°91
30	66°59	27°21	26°06	8.5068	77°13	63°46	87°38	244°62
60	96°59	29°67	52°99	.4707	86°20	82°35	106°27	290°61
90	126°59	23°58	81°62	1.1176	71°08	49°41	73°32	255°47
120	156°59	11°42	112°47	1.86866	62°19	20°57	44°487	224°11
150	186°59	356°72	145°52	2.7309	60°27	354°28	18°197	194°91
180	216°59	342°72	180°	3.6689	65°22	327°25	47°1	220°65
210	246°59	332°78	214°48	4.6083	77°15	296°51	320°42	130°81
240	276°59	330°32	249°95	5.5301	86°29	277°5	301°4	108°68
270	306°59	336°41	278°38	6.2203	71°09	310°56	334°56	139°43
300	336°59	348°58	307°	6.8669	62°18	339°42	3°337	166°11
330	6°59	3°278	333°94	7.4397	60°27	5°71	29°63	190°47
360	36°59	17°28	0°	5.6689	65°22	32°75	56°67	230°22

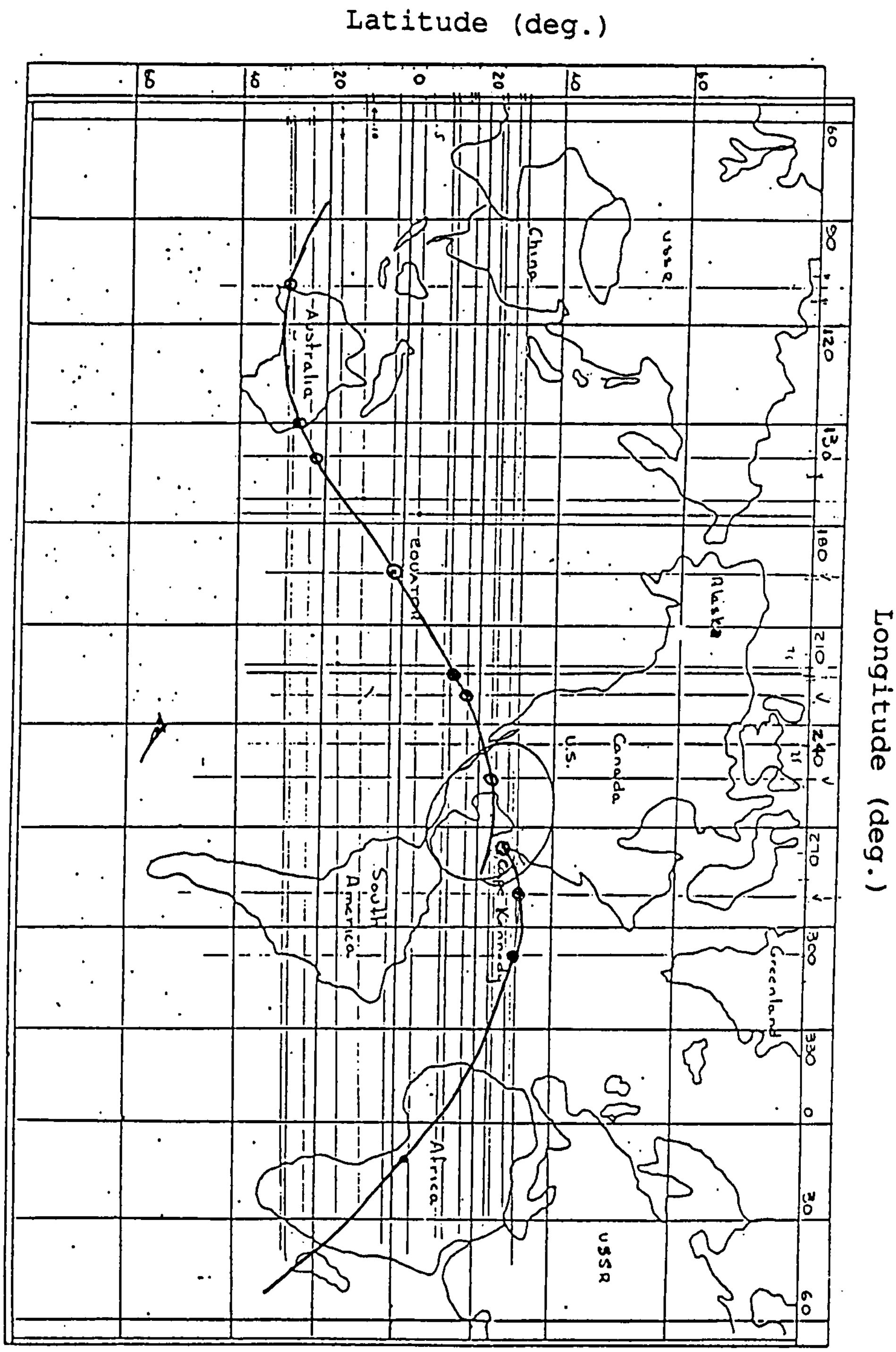

Fig. 2.0 Latitude Vs. Longitude

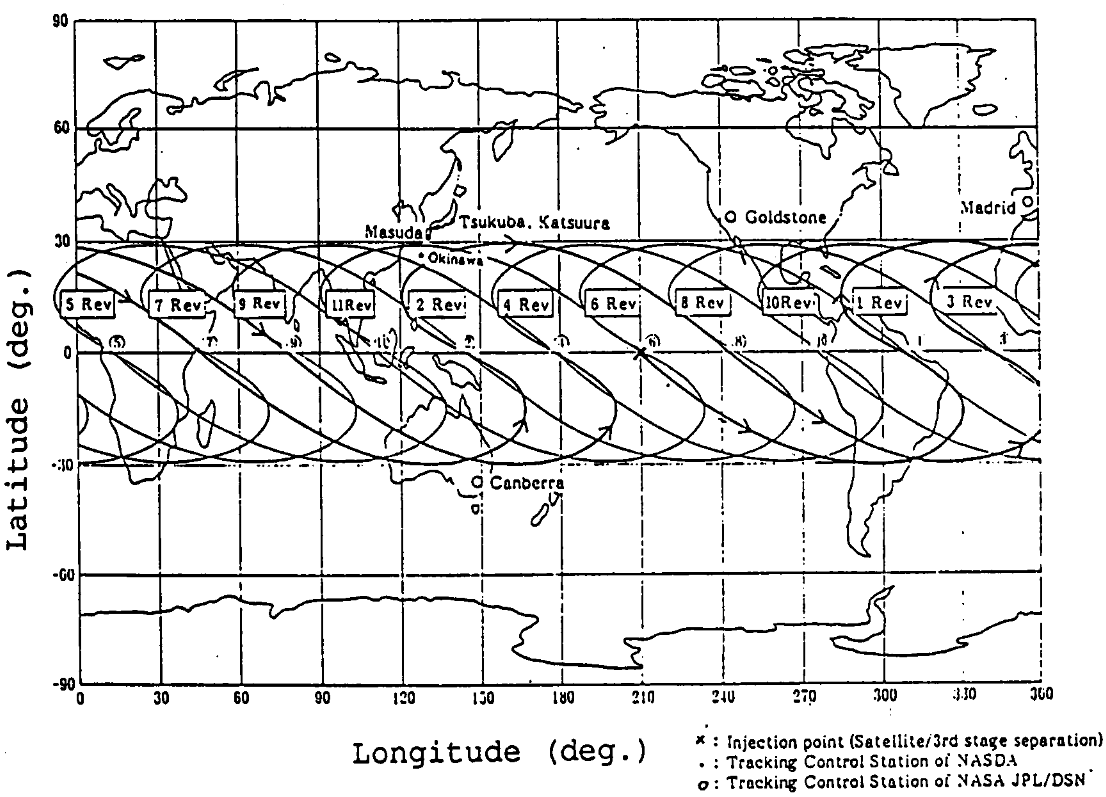

Fig. 3.0 Ground Trace of Satellite (NASDA)

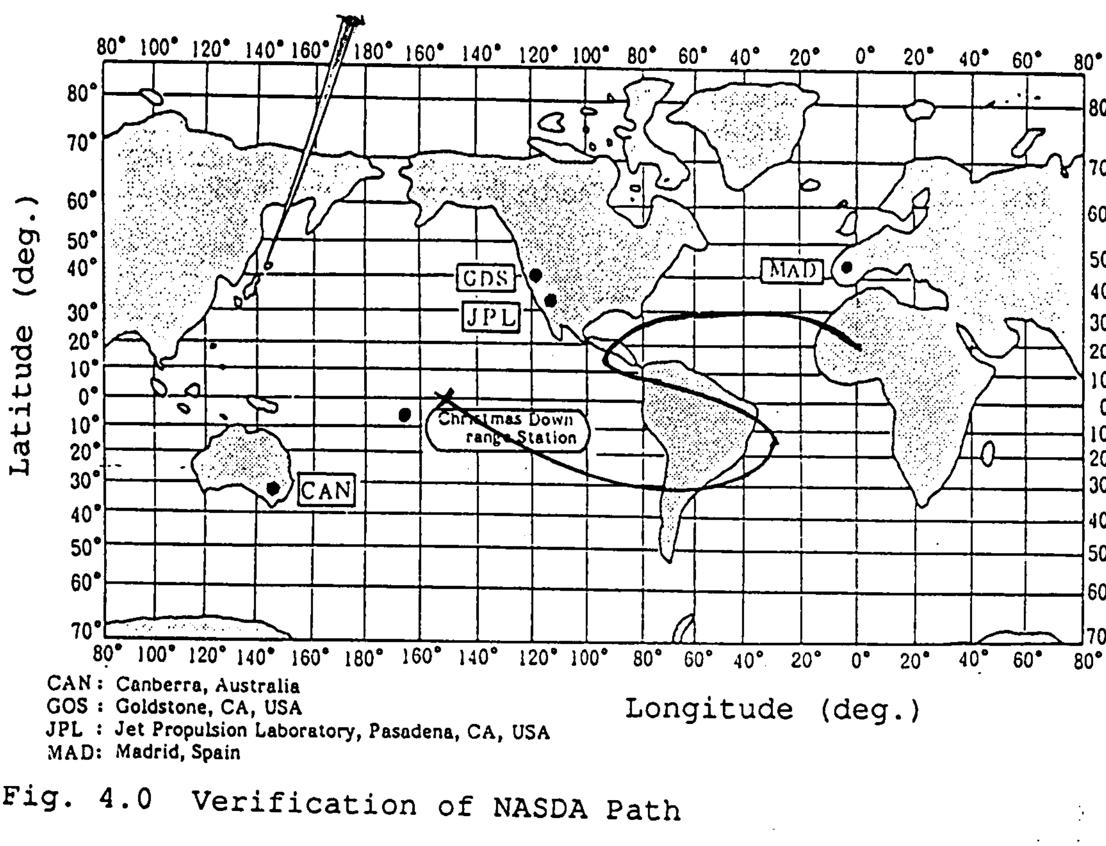

Fig. 4.0 Verification of NASDA Path

CHAPTER VI

ANALYSIS AND RECOMMENDATIONS

Social demand for the use of communications and broadcasting satellites is increasing year by year with the progress of the information-oriented society through the advancement of new media and computerization. Clear images and realistic sounds are being enjoyed in many homes through TV broadcasting using satellites. The utilization of satellites will become more and more popular, as exemplified by the birth of private coorporations specialized in providing communication services by communications satellites.

An estimate of the satellite launcher business in the 1990's indicates that between 200 and 270 satellites will require commercial launches up to the year 2000 [51]. About 134 satellites will require launchers between 1992-1995 and 131 between 1996-200 [51]. Based on this projection, the work done in this present thesis has become a matter of considerable interest.

This present work allows a user to find the launch vehicle available on the world market that best suits the requirements for the payload being launched just by accessing the database. It, also, allows a user to groundtrace the payload, at any moment in time, once the payload is in orbit.

The technical information provided by the database is very extensive. It provides information such as dimension of the launch vehicle, type and quantity of propellant used for each stage, thrust and specific impulse per stage, mission profile, payload capacity for each orbit, launch vehicle characteristics, a thorough description of the propulsion system, launch site parameters, rate of successful launch and so on. Whenever enough parameters were available, others that were not were calculated using known rocket propulsion equations in order to increase the amount of technical information available on the launcher.

The following launch vehicles were reviewed for this thesis:

Delta II family, Atlas family, Titan family, Taurus, Scout, Ariane family, U.S Shuttle, Proton, Buran, Energia, Long March family, H family, PSLV (India).

All of the information mentioned above is stored in the database for each launcher and be accessed by anyone to determined which launcher is best suited for a specific payload given a specific orbit. The database is able to match the user payload with a specific launch vehicle based on the following input from a user:

1. Payload Total Weight (including system payload)
2. Payload Final Orbit
3. Payload Volume

Once the match is found, an output is given containing all the technical information on the chosen vehicle.

The second aspect of this work is the groundtrace feature. To access this mode, the user has to input the following parameters:

1. Longitude of the Launch Site.
2. Latitude of the Launch Site.
3. Inclination Angle of the Launch Site.
4. Vo, Initial Velocity of the Spacecraft.
5. Final Altitude of the Spacecraft.
6. Date and Time of the Launch.

Based on this input, the six Orbital elements will be calculated and the initial position and velocity of the spacecraft in a vectorial form will also be performed. In addition, a graph mapping the orbit trajectory of the vehicle as a function of time will be generated.

The program is menu-driven. All the parts of this work are interconnected and any part can be accessed independently.

This research work can be regarded as a launch vehicles handbook of the 90's for commercial launch vehicles when taken into account the wealth of technical information presented on each vehicle, but also it can be used to assess today's and near future level of technology in the rocket industry. Some of the vehicles reviewed in this thesis will not be available not until mid-1990's, thus, it does project into near future as well. By analysing the level of thrust, specific impulse, the type of propellant and additive used, the type of control system used for the nozzle for thrust optimization (nozzle vectoring, gimballing, ect), the kind of composite material used for the nozzle and the rocket motor casing, the attainable chamber temperature and so on, a very good assessment, indeed, can be made of the today's rockets technology.

Recommendation for future work or an extension of this present one would be to update the database after a five years period in order to incorporate the avalanche of new launchers predicted soon to come so as to cope with the growing demand for satellites. Also, develop a graphic system capable of animating the mission profile phase of the database from liftoff to orbit insertion. Futhermore, incorporate a world map in the Orbital Mechanics part so that the graph which maps the orbit trajectory of the spacecraft shows a world map in the background instead of refering to the spot overflown by the spacecraft as a function of latitude and longitude as it is done in the present work, and look it up on a world map so that the exact location overflown can be known. In addition, consideration should be given to the followings:

1. The Orbital Mechanics part was carried out under the assumption of a two-body motion which is defined as the motion of body A with respect to body B, with only the mutual attraction of A and B taken into consideration. Thus, all the perburtative influence upon the motion of a body in space, such as drag or electromagnetic forces, which tend to deflect motion from two-body motion are ignored.

2. Also, for a two-body motion it is assumed that a satellite under the influence of an inverse square gravitational law has truly constant orbital elements. However, perturbative effects cause deviations from basic two-body motion made the orbital elements no longer constant.

Thus, the followings need to be taken into account for future work:

a. The force field of the primary body about which motion occurs is not truly of an inverse square variety, the effect of atmospheric drag enters into the equations of motions.

b. Radiative pressure of the sun produces accelerations.

c. The closeness of a neighboring celestial body attracts the object in question.

d. Also, consideration should be given to non-elliptical orbit such as parabolic and hyperbolic orbit in order to make the Orbital Mechanics part truly complete.

Finally, the spacecraft position and velocity should be calculated in vectorial form, not only for the initial position, but also able to be updated at any time.

vision should be made to update the Julian date, drift causes by drag pertubation should be taking into account for the argument of the ascending node. Finally, the spacecraft position and velocity should be calculated in vectorial form, not only for the initial position, but also able to be updated at any time.

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