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Ariane 5ECA

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1 Introduction

Ariane 5 is a European heavy-lift space launch vehicle developed and operated by Arianespace for the European Space Agency (ESA). It is launched from the Centre Spatial Guyanais (CSG) in French Guiana. It has been used to deliver payloads into geostationary transfer orbit (GTO) or low Earth orbit (LEO). The launch vehicle had a streak of 82 consecutive successful launches between 9 April 2003 and 12 December 2017. A direct successor system, Ariane 6, is in development.Since its first launch, Ariane 5 has been refined in successive versions: ”G”, ”G+”, ”GS”, ”ECA”, and most recently, ”ES”. We are taking ECA varient specifications for our analysis.

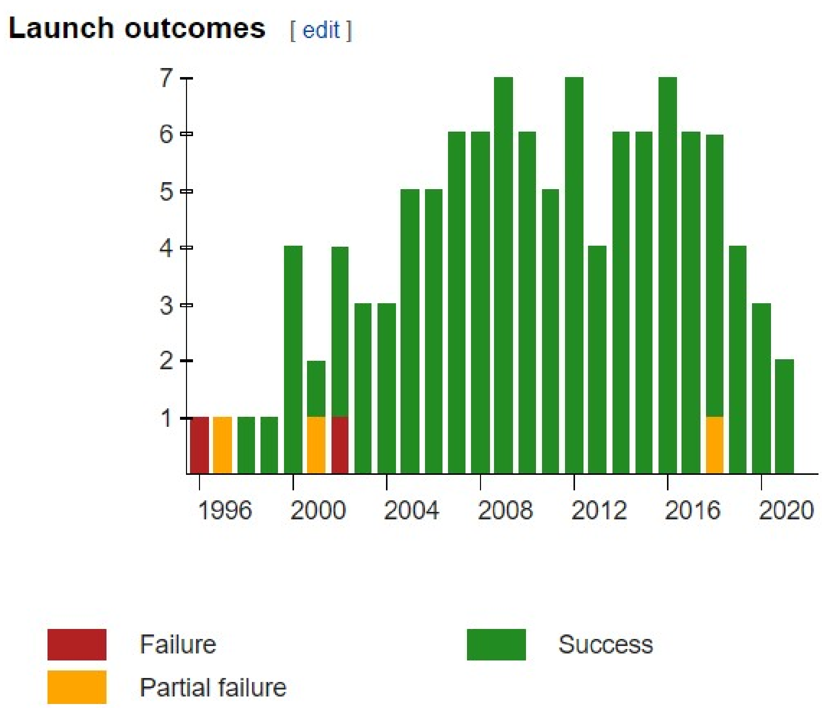
Ariane 5 Launch Vehicle

The **Ariane 5ECA** (Evolution Cryotechnique type A), first successfully flown in 2005, uses an improved Vulcain 2 first-stage engine with a longer, more efficient nozzle with a more efficient flow cycle and denser propellant ratio. The new ratio required length modifications to the first-stage tanks. The EPS second stage was replaced by the ESC-A (Etage Supérieur Cryogénique-A), which has a dry weight of 2,100 kg (4,600 lb) and is powered by an HM-7B engine burning 14,000 kg (31,000 lb) of cryogenic propellant. The ESC-A uses the liquid oxygen tank and lower structure from the Ariane 4’s H10 third stage, mated to a new liquid hydrogen tank. Additionally, the EAP booster casings were lightened with new welds and carry more propellant.

Ariane 5 uses 2.5 stages and four propulsion units. At liftoff, its twin segmented solid motors provide more than 500 tonnes thrust each to augment the single core stage Vulcain LOX/LH2 engine’s 110 tonnes of thrust. The original Ariane 5 G (”Generic”) version used a hypergolic EPS second stage with a 2.6 tonne thrust Aestus engine.

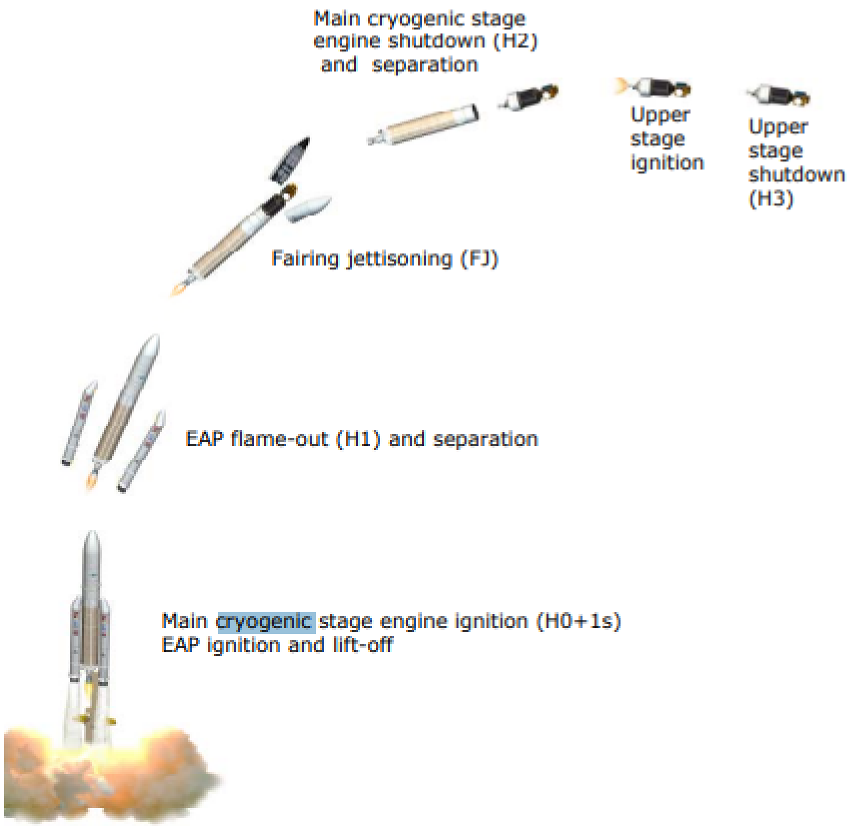
1.1 Main Features

First Launch	11-12-2002
Last Launch	24-10-2021
Total Launches	78
Successes	77
Failures	1 and 1(partial)
Thrust	15,360.00 kN
Gross mass	777,000 kg
Diameter	5.40 m (17.70 ft)
Height	59 m (193 ft)
Apogee	407 km (252 mi)
Stages	2
Payload to GTO	6,950 kg



Launch outcomes

1.2 Configuration and Launch Events



Configuration and Launch Events

Launch Sequence	Timing	Launch Events	Altitude
H0	0 : 00 : 00	Vulcain 2 ignition	**(km)**
H0 + 7 s	0 : 00 : 07	EAP ignition and liftoff	0
H0 + 143 s	0 : 02 : 23	EAP separation	69
H0 + 191 s	0 : 03 : 11	Fairing separation	107
H0 + 541 s	0 : 09 : 01	EPC separation	178
H0 + 1487 s	0 : 24 : 47	ESCA shut down	640
H0 + 1645 s	0 : 27 : 25	**Separation of 1st satellite**	1036
H0 + 1781 s	0 : 29 : 41	Separation of Sylda	1456
H0 + 2087 s	0 : 34 : 47	**Separation of 2nd satellite**	2585
H0 + 2976 s	0 : 49 : 36	End of launcher mission	6340

— Typical Ariane 5 ECA launch sequence (GTO mission)

Ariane 5 ECA typical launch sequence for GTO missions

1.2.1 Stage 0 (Booster stage):

No. of boosters	2
Gross Mass	278,330 kg (613,610 lb)
Empty Mass	38,200 kg (84,200 lb)
Thrust (vac)	7,000 kN
Isp	274.5 sec
Burn time	130 sec
Isp(sl)	250 sec
Diameter	3.05 m (10.00 ft)
Span	3.05 m (10.00 ft)
Length	31.60 m (103.60 ft)

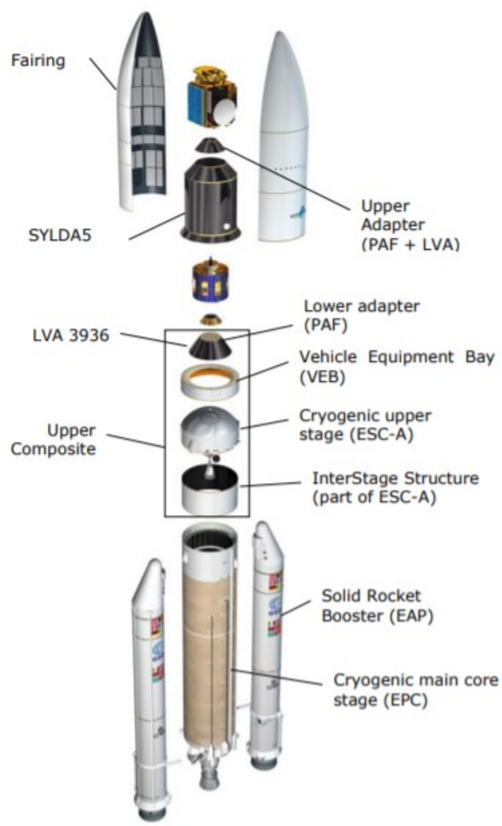
1.2.2 1st stage:

Gross Mass	186,000 kg (410,000 lb)
Empty Mass	14,700 kg
Thrust (vac)	1,390 kN
Isp(vac)	432 sec
Burn time	540 sec
Isp(sl)	310 sec
Diameter	5.4 m
Span	5.4 m
Length	23.8 m

1.2.3 2nd stage:

Gross Mass	16,500 kg (36,300 lb)
Empty Mass	4,540 kg
Thrust (vac)	64.7 kN (14,545 lbf)
Isp	446 sec
Burn time	945 sec
Diameter	5.4 m
Span	5.4 m
Length	9 m

2 Propulsion System Study



Segment

2.1 Stage 0

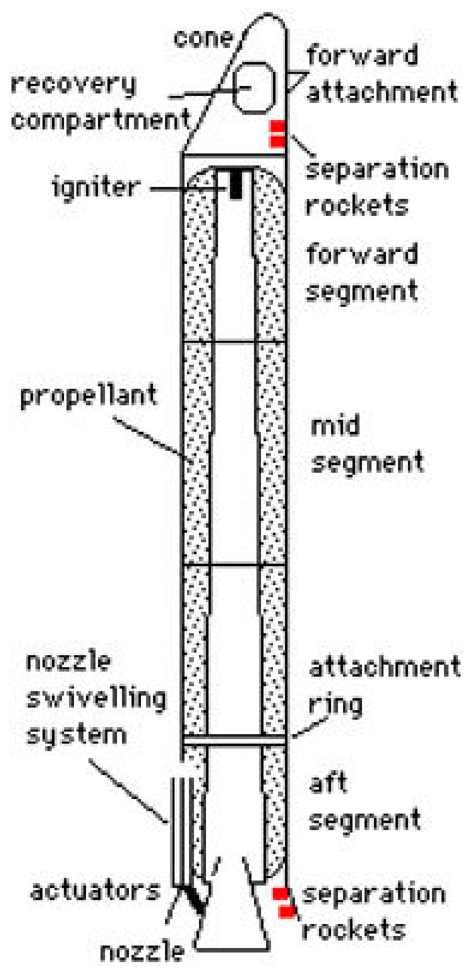
Stage 0 is the booster stage. The Ariane-5 solid propellant boosters are the largest solid rocket boosters ever produced in Europe. They weigh about 37 tonnes each when empty. They are referred to as the Etage d'Acceleration à Poudre, from French and can contain about 238 tonnes of propellant. Ariane-5 boosters provide 1100 tonnes of thrust, roughly 92% of the total thrust at liftoff. The construction of the system happens all over Europe, namely

- Responsible contractor: EADS/LV (France)
- motor: EUROPROPULSION (France-Italy)
- nozzle actuators and structures: SABCA (Belgium)
- sea-recovery systems: Dutch Space the Netherlands)

2.1.1 Principal Construction Features

This booster stage uses EAP Launch Vehicle, aka the stage integrator, and Europropulsion systems as the motors. The size is 31.2m long, 3.05m in diameter and it has 40 tonnes of empty mass. Each EAP consists of a steel casing enclosing three segments filled with propellant and joined together.

The top segment is loaded with 23.5 tonnes of propellant, while the middle and bottom segments are filled in the solid propellant production plant (UPG). The middle segment contains 107.5 tonnes of propellant and the bottom segment contains 107 tonnes of propellant. Although the casings are only 8 mm thick, they can resist pressures of up to 64 bar.This stage is powered by 23.5 tonnes of solid propellant which generates a thrust of 5250 kN each at launch with a burn time of 132 seconds.



Launcher Booster

2.1.2 Design Features

The motor is assembled using 3 sections, each made of 8mm-thick steel walls. Three 3.35m long cylinders sit in the two lower section of the boosters. Two hydraulic actuators are used for nozzle steering using flex-bearing for 6° deflection. The recovery of this two booster stage for inspection is done by using parachutes that are carried in nosecones.

The boosters' separation is triggered by an acceleration threshold detection and the fairing is released approximately one minute later when the aerothermal flux becomes lower than the required flux (The fairing can be jettisoned as soon as aerothermal flux less than 1600 W/m2 at 99%).

2.1.3 Thrust Vectoring

At the base of each booster is the 3.8 m long solid rocket engine nozzle. This can be swivelled up to 7.3° degrees around its axis by a nozzle-actuating unit to vary the direction of thrust. The outer diameter of the nozzle is 3.1 m.

2.1.4 Recovery Mechanism

Approximately 132 seconds after liftoff, at an altitude of 60 km, pyrotechnic devices free the boosters and separation rockets distance the spent boosters from the main stage. The boosters then continue their trajectory for about 100 km before falling into the Atlantic Ocean, approximately 450 km from the launch site.

From time to time, when performance of the launcher allows it, the boosters are equipped with a parachute recovery system. After sea recovery, boosters are examined to check that the stage performed as expected.

2.1.5 Propellant

This stage uses HTPB solid propellant which comprises of 68% ammonium perchlorate, 18% aluminium and 14% liner which is produced and cast into casings. It also comprises of 3.4 m-long forward sections which are loaded with BPD. Stated below are the roles played by each element in the propellant mixture:

- ammonium perchlorate: the oxidiser
- aluminium powder: acts as the reducer
- polybutadiene: binder and catalyser

It takes 350 milliseconds for the igniter in the upper segment of the booster to ignite the propellant. The radial rate of combustion, from the centre to the exterior of the booster segment, is approximately 7.5 mm/sec. Specific impulse is 262 sec., maximum pressure 64 bar and thrust 5400 kN. Gases from the combustion are expelled at an average flow rate of 2 tonnes/sec.

2.1.6 Operational Data

Size	3.05 m x 31.6 m
Number of Boosters	2
Height	31.6 m
Diameter	3.05 m
Empty mass	33 tonnes
Gross mass	273 tonnes
Powered by	P241
Maximum thrust	7080 kN
Total thrust	14160 kN
Mean thrust	7000 kN (vaccuum)
Sea level thrust	5400 kN
Mass flow rate	2 tonnes/sec
Isp	274.5 s
Burn time	130 s
Propellant	Ammonium Perchlorate, Aluminium, HTPB (total 240 tonnes)
Nozzle expansion ratio	11.0
Nozzle Exit Area	7.55 m^2
Avg. mass flow	2000 kg/s
Chamber Pressure	64 bar
Attitude control	Steerable nozzle
Avionics	Flight control, flight termination and telemetry subsystems, connected to VEB via data bus + autonomous telemetry

2.2 Stage I

Two solid rocket boosters provide 90 percent of Ariane 5's thrust at lift-off. A cryogenic core stage, ignited and checked on ground, provides the remaining thrust for the first part of the flight up to the upper stage separation.

2.2.1 Construction features

The EPC is composed of five subsystems: the Vulcain engine and its actuation system, the main tank, the thrust frame, and the forward skirt.

The EPC stage is 5.4 m in diameter and 31 m long. The structure consists of two Aluminium alloy tanks separated by a a common bulkhead, creating a 120 m3 LO2 forward tank and a 390 m3 LH2 aft tank. The tank's external surface carries a 2 cm-thick insulation layer to help maintain the cryogenic temperatures. The metal case is made up of three bulkheads and seven cylinders. Each bulkhead is made up of eight sectors and a Y-ring which provides an interface between the bulkhead and the other EPC components.

The seven cylinders, each one comprising three pre-formed and welded panel are manufactured at the Cryospace plant in Les Mureaux (France). The cylinders and the bulkheads are then welded together, pressure-tested, insulated, equipped, and finally delivered to Aerospace's plant on the same site, for assembling into the EPC stage.

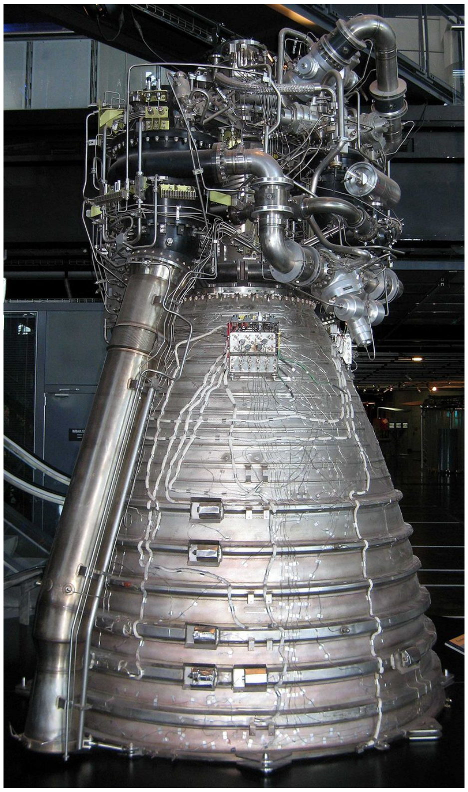
The forward skirt is made of aluminium and fibre composites. It transmits the thrust generated by the boosters through two fittings to the launcher's central body. These fittings also contain the booster forward jettison mechanisms.

2.2.2 Propellants

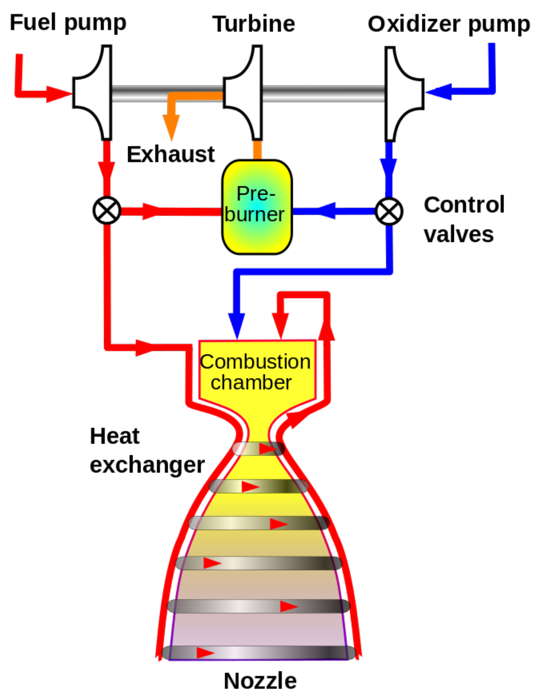
The EPC cryogenic core stage burns cryogenic propellants combining liquid hydrogen and oxygen. It carries 174 tonnes of propellant: 149 tonnes of liquid oxygen on the top tank and 25 tonnes of liquid hydrogen in the tank below. It is powered by one Vulcain 2 engine that burns liquid hydrogen (LH2) and liquid oxygen (LO2). The Vulcain engine is made up of four main components: the hydrogen turbopump, the oxygen turbopump, the combustion chamber and the nozzle. The Vulcain 2 engine develops 1,390 kN maximum thrust in vacuum. It operates for 9 minutes, before separating from the upper stage at an altitude of around 145 km.

2.2.3 Feed System and Gas Generator cycle

The Vulcain is a gas-generator cycle rocket engine fed with cryogenic liquid oxygen and liquid hydrogen. It features regenerative cooling through a tube wall design, and the Vulcain 2 introduced a particular film cooling for the lower part of the nozzle, where exhaust gas from the turbine is re-injected in the engine. The engine is turbo pumped and regeneratively cooled. The thrust chamber is fed by two independent turbo pumps using a single gas generator.



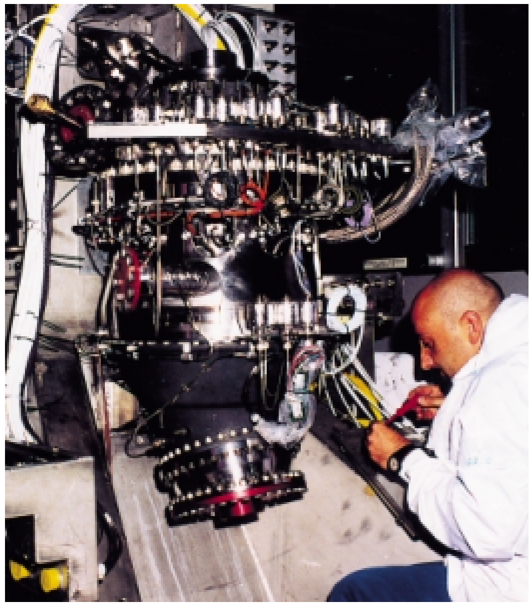
Vulcain 2



Gas generator cycle

2.2.4 Turbine Assembly

The hydrogen turbopump is manufactured by SEP in Vernon (France), the oxygen turbopump by Fiat Avio in Turin (Italy), the combustion chamber by Daimler Benz Aerospace (DASA) in Ottobrunn (Germany), and the nozzle by Volvo Aero Corporation in Trollhättan (Sweden). The engine is assembled in the SNECMA/SEP cryogenic motor assembly building in Vernon. Firing tests to support the development of the Vulcain engine started in 1990 on two identical, specially-built test stands, the PF50 stand at SEP in Vernon and the P5 stand at DLR in Lampoldshausen (Germany).



Turbopump Assembly

2.2.5 Thrust vectoring system

The nozzle is gimballed for pitch and yaw control. A cluster of 4 GH2 thrusters is used for roll control. The engine thrust actuation unit is the hydraulic system that activates the thrust vector control of the Vulcain engine.

2.2.6 Starting system

Ignition of the Vulcain-2 engine is obtained by pyrotechnic igniters. It is ignited at H0+1s. Until H0+7.05 seconds, the on-board computer checks the good behavior of the engine and authorizes the lift-off by the ignition of the two solid rocket boosters.

The boosters' separation is triggered by an acceleration threshold detection and the fairing is released approximately one minute later when the aerothermal flux becomes lower than the required flux (The fairing can be jettisoned as soon as aerothermal flux 1600 W/m2 at 99%).

2.2.7 Propellant Loading and Pressurization

The 120 m3 forward tank contains LO2 at -180°C which is pressurised to 3.5 bar by helium. The 390 m3 aft tank contains LH2 at -250°C which is pressurised to 2.5 bar by gaseous H2.

2.2.8 Engine shutdown

The engine shut down command is sent by the On Board Computer (OBC) when the launcher has reached a pre-defined orbit or when a critical level of depletion of one of the propellant tanks has been reached. The separation happens 6 seconds after. After its separation, the main stage is put in a flat spin mode by opening a lateral venting hole in the hydrogen tank. This control procedure provides a re-entry and a splashdown in the Atlantic Ocean for standard A5ECA GTO missions.

2.2.9 Operational Data

Size	5.4 m x 23.8 m (without engine)
Dry mass	14700 kg
Engine mass	2100kg
Nozzle extension mass	461kg
Structure	Aluminium alloy tanks
Propulsion	Vulcain 2 - 1 chamber
Propellants	174 t of LOX + LH2
Mixture ratio	6.7
Total engine mass flow	323kg/s
Thrust	960 kN (SL) - 1390 kN (Vacuum)
Isp	310 s (SL) 432 s (Vacuum)
Nozzle expansion ratio	58.2
Nozzle exit area	3.43 m^2
Chamber pressure	117.3 bar
Combustion chamber temperature	1500 C
Feed system	2 turbo-pumps driven by a gas generator
Pressurization	GHe for LOX tank and GH2 for LH2 tank
Combustion time	540 s
Attitude control	Pitch and yaw: gimballed nozzle Roll: 4 GH2 thrusters
Avionics	Flight control, flight termination, power distribution and telemetry subsystems, connected to VEB via data bus

2.3 Stage II

The second stage is on top of the main stage and below the payload. It provides a final thrust once the EPC runs out of fuel. The Ariane 5ECA uses the ESC (Cryogenic Upper Stage), which is fuelled by liquid hydrogen and liquid oxygen. The upper section consists of the upper composite (UC), including the upper stage (ESC-A) and the Vehicle Equipment Bay (VEB), and on top the payload composite.

2.3.1 Upper Composite (UC) - Lower Part : Cryogenic Upper Stage (ESC-A)

Ariane 5 ESC-A is an upper stage craft, which enabled Ariane 5 to place loads of 10,000kg-10,500kg into geostationary orbits. The ESC-A stage is 5.4 m in diameter and 4.8 m long between the I/F rings.

SNECMA HM7B rocket engine

2.3.2 Engine

The HM7B is a European cryogenic upper stage rocket engine used on the vehicles in the Ariane rocket family. The HM7B is a regeneratively cooled gas generator rocket engine fed with liquid oxygen and liquid hydrogen. It has no restart capability: the engine is continuously fired for 950 seconds in Ariane 5. It provides 67 kN of thrust with a specific impulse of 446 s. The engine's chamber pressure is 3.5 MPa.

2.3.3 Propellants

The HM7B engine burns liquid hydrogen (LH2) and liquid oxygen (LO2) stored in two fully separated tanks. The LO2 tank is pressurized by gaseous helium and the LH2 one by a part of gaseous hydrogen coming from the regenerative circuit. The HM7B engine develops 67 kN maximum thrust in vacuum. The engine is turbopump-fed and regeneratively cooled. The thrust chamber is fed by two pumps (LH2 and LO2) driven by a gas generator, a common turbine and a gear box.

2.3.4 Design

The orbit injection stage ensures payload orientation and separation. Required to nestle inside the VEB under the payload fairing, it is designed for compactness: the engine is embedded within the four propellant spheres (each 1.41 m-diameter, pressurised to 18.8 bar by helium). Main structural element is frustum continuing VEB's frustum at 3936 mm-diameter lower face and supporting payload adapters on 1920 mm-diameter forward face

2.3.5 Thrust vectoring system

During the powered flight, the attitude control in pitch and yaw is ensured by the gimballing of the nozzle, and 4 GH2 thrusters are used for roll control. During the ballistic phase, roll, pitch and yaw control uses 2 clusters of 3 GH2 thrusters. 2 GO2 thrusters are also implemented for longitudinal boosts.

ESC-A

2.3.6 Engine shutdown

The engine shut down command is sent by the On Board Computer (OBC) when the launcher has reached a predefined orbit or when the OBC detects a thrust tail-off on depletion.

2.3.7 Upper Composite (UC) - Upper Part : Vehicle Equipment Bay (VEB)

All guidance, stage sequencing, telemetry, tracking and safety systems are supported by the VEB. In addition to separation commands, the spacecraft could be provided with additional commands (electrical or pyrotechnic), power and data transmission to the ground. Two redundant ring laser gyroscopes ensure inertial reference and guidance.

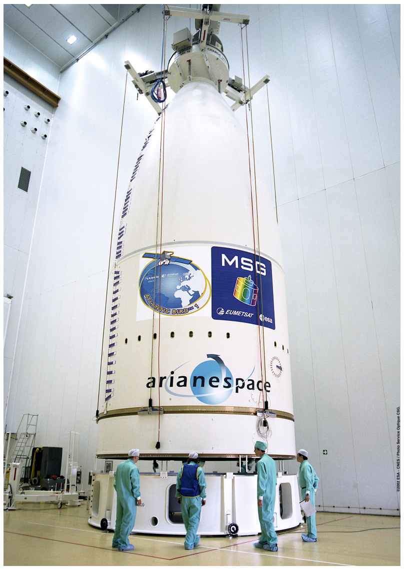
Vehicle Equipment Bay (VEB)

2.3.8 Operational Data

Size	ϕ 5.4 m x 4.711 m between I/F rings
Dry mass	4540 kg
Structure	Aluminium alloy tanks
Propulsion	HM7B engine - 1 chamber
Propellants loaded	14.9 t of LOX + LH_2
Thrust	67 kN
Isp	446 s
Vacuum Thrust	64.8 kN
Nozzle Expansion Ratio	83.1
Nozzle Exit Area	0.773 m^2
Chamber Pressure	37 bar
Propellant mix ratio	4.9
Engine mass flow	14.8 Kg/s
Feed system	1 turbo-pump driven by a gas generator
Pressurization	GHe for LOX tank and GH2 for LH2 tank
Altitude of firing	Between 160 km and 210 km depending on the mission trajectory
Combustion time	945 s
Attitude control powered phase	Pitch and yaw: gimballed nozzle Roll: 4 GH2 thrusters
Attitude control ballistic phase	Roll, pitch and yaw : 4 clusters of 3 GH2 thrusters Longitudinal boost : 2 GO2 thrusters
Avionics	Guidance from VEB

2.4 Payload Composite

2.4.1 Fairing



Fairing

The fairing is positioned on the very top of the Ariane-5 launcher. Its function is to protect the payload against aerodynamic, thermal and acoustic phenomena as the launcher rises from the launch pad through the atmosphere to an altitude of approximately 100 km. Once the launcher leaves the Earth's atmosphere, approximately three minutes after liftoff, the fairing is jettisoned. This lightens the remaining launcher's load as it loses approximately two tonnes of this no-longer required structure. The payload fairing consists of two large composite half shells whose inside surfaces are covered with acoustic attenuation panels. This acoustic protection is used to absorb noise generated by the engines mainly during the lift-off event. The payload fairing has an external diameter of 5.4 m and a total height of 17 m. In order to increase the volume available for the S/C, a fairing raising cylinder can be used.

2.4.2 Dual Launch System

In the dual launch configuration, the SYLDA 5 carrying structure is used. The Sylda 5 is housed inside the fairing. At the bottom it has a 50 cm cone which interfaces with the vehicle equipment bay. When empty the Sylda 5 weighs 440 kg.
During a standard dual-launch mission, the upper satellite is released first. Then, the Sylda-5 is jettisoned in order to release the second satellite. Both systems have pyrotechnic separation systems at their base and push-off springs.
Separation is triggered by two detonators which sever the steel attachment between the vehicle equipment bay and the Sylda. The dual payload structure is then pushed away by means of eight special steel springs. The standard SYLDA 5 composite structure consists of a rear conical part of 0.6 m, a cylinder height of 3.2 m and another conical part (height 1.1 m) reaching a total height of 4.9 m, with a usable internal diameter volume of 4 m. The cylinder can be extended by up to 1.5 m in steps of 0.3 m. An additional version with a cylinder extension of 2.1 m is also contemplated. The total height can then reach 7 m.

2.4.3 Cone 3936 or LVA 3936

The cone 3936 or LVA 3936 is an adaptation structure between the VEB upper frame (ϕ3936) and the lower frame of the Ariane 5 standard adaptors (respectively ϕ2624 or ϕ1780). Cone 3936 is 783mm high and LVA3936 is 1187 mm high. They are composed of a carbon structure and 2 aluminium rings. They both comprise a membrane, which separate the satellite compartment from the upper stage. It is designed to prevent helium transfer upper part compartment to satellite compartment.

2.4.4 Adapters

Payload adapters, generally of conical shape, ensure interfaces between the launcher and the spacecraft.
They consist of
• a conical or a cylindrical structure with
• an upper interface (937, 1194, 1663, 1666 and 2624mm) compatible with the spacecraft
• a bottom bolted interface (ϕ 1780mm or ϕ 2624mm)
• a separation system (generally a clamp-band) with springs to meet spacecraft separation requirements; a four-bolt separation system is also available for the 1663 interface
• an electrical system (connectors, microswitches. . .) including satellite umbilical lines and vibration sensors

3 Propulsion System Performance Analysis

3.1 Mass calculations

3.1.1 Initial Gross Mass and Burnout Mass for each Stage

1. Initial Mass

$$M_{01} = \sum_{1=0}^{2}(n * M_{Ei})$$

$$M_{\infty} = 2 * M_{E0} + M_{E1} + M_{E2} + M_L$$

where M_{00} is the Initial Gross Mass of stage 0
M_{E0}, M_{E1}, M_{E2} are the Individual Engine Masses of stage 0, I, and II.

$$M_{S0} = 33000\,Kg\,;\ M_{P0} = 240000\,Kg\,;\ M_{E0} = M_{S0} + M_{P0} = 273000\,Kg$$

$$M_{S1} = 14700\,Kg\,;\ M_{P1} = 174000\,Kg\,;\ M_{E1} = M_{S1} + M_{P1} = 188700\,Kg$$

$$M_{S2} = 4540\,Kg\,;\ M_{P2} = 14900\,Kg\,;\ M_{E2} = M_{S2} + M_{P2} = 19440\,Kg$$

$$M_{00} = 2 * 273000 + 188700 + 19440 + 6950 = 761090\,Kg$$

Similarly we can do the calculations for other stages.

$$M_{01} = M_{E1} + M_{E2} + M_L = 188700 + 19440 + 6950 = 215090Kg$$

$$M_{02} = M_{E2} + M_L = 19440 + 6950 = 26390Kg$$

2. Burnout Mass

$$M_{B0} = M_{b0} + M_{E1} + M_{E2} + M_L$$

where M_{B0} is the Burnout Mass of stage 0
and M_{b0} is Burnout mass of Engines of Stage 0

$$M_{b0} = M_{S0} * 2 = 66000Kg$$

$$M_{B0} = 66000 + 188700 + 19440 + 6950 = 281090\,Kg$$

Similarly we can do the calculation for the other two stages:

$$Mb1 = M_{S1} = 14700\,Kg$$

$$M_{B1} = M_{b1} + M_{E2} + M_L = 14700 + 19440 + 6950 = 41090\,Kg$$

$$M_{b2} = M_{S2} = 4540\,Kg$$

$$M_{B2} = M_{b2} + M_L = 4540 + 6950 = 11490\,Kg$$

3.1.2 Mass Ratio(R)

$$R_i = \frac{M_{0i}}{M_{Bi}} = \frac{Initial\ Mass\ of\ Stage}{Burnout\ Mass\ of\ Stage}$$

Stage 0:

$$R_0 = \frac{M_{00}}{M_{B0}} = \frac{761090}{281090} = 2.708$$

Stage 1:

$$R_1 = \frac{M_{01}}{M_{B1}} = \frac{215090}{41090} = 5.235$$

Stage 2:

$$R_2 = \frac{M_{02}}{M_{B2}} = \frac{26390}{11490} = 2.297$$

3.1.3 Propellant Mass Fraction(η)

$$\eta_i = \frac{Mass\ of\ Propellant}{Initial\ Mass\ of\ Stage} = \frac{M_{pi}}{M_{0i}} = \frac{M_{0i} - M_{Bi}}{M_{0i}}$$

For Stage 0:

$$\eta_0 = \frac{M_{00} - M_{B0}}{M_{00}} = \frac{2 \times 240000}{761090} = 0.631$$

Stage 1:

$$\eta_0 = \frac{M_{01} - M_{B1}}{M_{01}} = \frac{174000}{215090} = 0.809$$

Stage 2:

$$\eta_0 = \frac{M_{02} - M_{B2}}{M_{02}} = \frac{14900}{26390} = 0.565$$

3.1.4 Pay Load Fraction (λ)

$$\lambda_i = \frac{M_{0(i+1)}}{M_{0i} - M_{0(i+1)}}$$

For Stage 0:

$$\lambda_0 = \frac{M_{01}}{M_{00} - M_{01}} = \frac{761090}{546000} = 1.394$$

Stage 1:

$$\lambda_1 = \frac{M_{02}}{M_{01} - M_{02}} = \frac{215090}{188700} = 1.140$$

Stage 2:

$$\lambda_2 = \frac{M_L}{M_{02} - M_L} = \frac{6950}{19440} = 0.358$$

3.1.5 Structural Coefficient (ϵ)

$$\epsilon_i = \frac{M_{Ei}}{M_{0i} - M_{0(i+1)}}$$

For Stage 0:

$$\epsilon_0 = \frac{M_{S0}}{M_{00} - M_{01}} = \frac{66000}{546000} = 0.121$$

Stage 1:

$$\epsilon_1 = \frac{M_{S1}}{M_{01} - M_{02}} = \frac{14700}{188700} = 0.078$$

Stage 2:

$$\epsilon_2 = \frac{M_{S2}}{M_{02} - M_L} = \frac{4540}{19440} = 0.234$$

3.2 Performance Calculations

3.2.1 Effective Exhaust Velocity, (c)

$$c_i = I_{spi} * g_0$$

For Stage 0:

$$c_0 = I_{sp0} * g_0 = 274.5 * 9.81 = 2692.845 m/s$$

For Stage I:

$$c_1 = I_{sp1} * g_0 = 310 * 9.81 = 3041.1 m/s$$

For Stage II:

$$c_2 = I_{sp2} * g_0 = 446 * 9.81 = 4375.26 m/s$$

3.2.2 ΔU for Each Stage

$$\Delta U_i = c_i * ln(R_i)$$

For Stage 0:

$$\Delta U_0 = c_0 * ln(R_0) = 2692.845 * ln(2.708) = 2682.64 m/s$$

Stage 1:

$$\Delta U_1 = c_1 * ln(R_1) = 3041.1 * ln(5.235) = 5034.14 m/s$$

Stage 2:

$$\Delta U_2 = c_2 * ln(R_2) = 4375.26 * ln(2.297) = 3638.5 m/s$$

3.2.3 Total ΔU

$$\Delta U = \sum_{1=0}^{2} (\Delta U_i) = \Delta U_0 + \Delta U_1 + \Delta U_2 = 2682.64 + 5034.14 + 3638.5 = 11355.38 m/s$$

3.2.4 Total Impulse, (I_T)

$$I_{Ti} = I_{si} * M_{pi} * g_0$$

Stage 0:

$$I_{T0} = I_{s0} * M_{p0} * g_0 = 274.5 * 240 * 1000 * 9.81 = 646.283MN - sec$$

Stage 1:

$$I_{T1} = I_{s1} * M_{p1} * g_0 = 310 * 174 * 1000 * 9.81 = 529.151MN - sec$$

Stage 2:

$$I_{T2} = I_{s2} * M_{p2} * g_0 = 446 * 14.9 * 1000 * 9.81 = 65.191MN - sec$$

3.2.5 Thrust(F) for Each Stage

$$F_i = \frac{M_p * c_i}{t_b}$$

Stage 0:

$$F_0 = \frac{M_{p0} * c_0}{t_b} = \frac{240000 * 2692.845}{130} = 4971.41kN$$

Stage 1:

$$F_1 = \frac{M_{p1} * c_1}{t_b} = \frac{174000 * 3041.1}{540} = 979.91kN$$

Stage 2:

$$F_2 = \frac{M_{p2} * c_2}{t_b} = \frac{14900 * 4375.26}{945} = 68.99kN$$

3.2.6 Thrust to Weight Ratio

Stage 0:

$$\frac{F_0}{W_0} = \frac{F_0}{M_{00} \times g_0} = \frac{2 \times 4971.41}{760.09 \times 9.81} = 1.33$$

Stage 1:

$$\frac{F_1}{W_1} = \frac{F_1}{M_{01} \times g_0} = \frac{979.91}{215.09 \times 9.81} = 0.46$$

Stage II:

$$\frac{F_2}{W_2} = \frac{F_2}{M_{02} \times g_0} = \frac{68.99}{26.39 \times 9.81} = 0.27$$

3.2.7 Thrust to Burnout Weight Ratio

Stage 0:

$$\frac{F_0}{W_{B0}} = \frac{F_0}{M_{B0} \times g_0} = \frac{2 \times 4971.41}{281.09.09 \times 9.81} = 3.61$$

Stage 1:

$$\frac{F_1}{W_{B1}} = \frac{F_1}{M_{B1} \times g_0} = \frac{979.91}{41.09 \times 9.81} = 2.43$$

Stage II:

$$\frac{F_2}{W_{B2}} = \frac{F_2}{M_{B2} \times g_0} = \frac{68.99}{11.49 \times 9.81} = 0.61$$

3.2.8 Maximum Acceleration for each stage

$$a_{max,i} = \frac{F_i}{M_{Bi}} - g$$

Stage 0:

$$a_{max,0} = \frac{F_0}{M_{B0}} - g_0 = \frac{2 \times 4971.41}{281.09} - 9.81 = 25.56 m/s^2$$

Stage I :

$$a_{max,1} = \frac{F_1}{M_{B1}} - g = \frac{979.91}{41.09} - 9.81 = 14.04 m/s^2$$

Stage II: (considering horizontal flight when Stage-II fires)

$$a_{max,2} = \frac{F_2}{M_{B2}} = \frac{68.99}{11.49} = 6.00 m/s^2$$

3.3 Nozzle Performance Calculations

3.3.1 Stage 0

Stage 0 consists of 2 EAP boosters and uses a combination of HTBP and Ammonium perchlorate as propellants

Calculations for 1 EAP Booster of Stage-0

- Sea Level Thrust for single booster (given in data)

$$F_{sea} = 5400 kN$$

- Vacuum Thrust for single booster(given in data)

$$F_{vaccum} = 7000 kN$$

- Formula for sea level Thrust,

$$F_{sea} = \dot{m} \times V_2 + (P_2 - P_3) \times A_2$$

$$F_{vacuum} = \dot{m} \times V_2 + (P_2 - 0) \times A_2$$

- Chamber Pressure is given to be

$$P_1 = 64 * 10^5 Pa$$

- Nozzle Area Expansion ratio is given

$$\epsilon = \frac{A_2}{A_t} = 11$$

- Nozzle Exhaust Pressure can be calculated from the following equation,

$$\frac{A_t}{A_2} = \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}} \left(\frac{P_2}{P_1}\right)^{\frac{1}{k}} \sqrt{\frac{k+1}{k-1}\left(1 - \left(\frac{P_2}{P_1}\right)^{\frac{k-1}{k}}\right)} = \frac{1}{\epsilon} = 0.091$$

$$\Rightarrow \frac{P_2}{P_1} = 0.0075$$
$$\Rightarrow P_2 = 48000 Pa$$

- Nozzle Exit Area, is given to be

$$A_2 = 7.55 m^2$$

- We can calculate mass flow rate as follows:

$$\dot{m} = \frac{M_p}{t_b} = \frac{240000}{130} = 1846.15 kg/s$$

- From the values of $\dot{m}$, P_2 and A_2, we can calculate relative exhaust velocity, of the stage

$$V_2 = \frac{F_{vacuum} - P_2 A_2}{\dot{m}} = \frac{7000000 - 48000 \times 7.55}{1846.15}$$
$$V_2 = 3595.37 m/s$$

- Nozzle area expansion ratio is given and can be used to calculate throat area,

$$\epsilon = \frac{A_2}{A_t} \implies A_t = \frac{A_2}{\epsilon} = 7.55/11 = 0.686 m^2$$

- Characteristic Velocity can now be calculates using the values of $\dot{m}$, P_1 and A_t, by using the following equation:

$$\dot{c^*} = \frac{P_1 \times A_t}{\dot{m}} = \frac{64 \times 10^5 \times 0.686}{1846.15} = 2378.13 m/s$$

- Molecular Weight of HTPB-AP, (from literature)

$$MW = 23 g/mol$$

- Specific Heat Constant,

$$k = 1.2\,(for\,high\,temperatures)$$

- Gas Constant of fluid mixture,

$$R_{xxx} = \frac{R_u}{MW} = \frac{8314}{23} = 361.47 J/Kg - K$$

- Using the following formula for characteristic velocity, combustion chamber pressure, T_1 can be calculated as follows by substituting the values of R_{xx}, k, and c^*:

$$c^* = \frac{\sqrt{k \times R_{xx} \times T_1}}{k\sqrt{\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}} \implies \frac{\sqrt{1.2 \times 361.47 \times T_1}}{1.2\sqrt{\left(\frac{2}{1.2+1}\right)^{\frac{1.2+1}{1.2-1}}}} = 2378.13 \implies T_1 = 6580.53K$$

- Finally, Sea level thrust coefficient is given by:

$$C_F = \frac{F_{sea}}{P_1 A_t} = \frac{5400000}{64 \times 10^5 \times 0.686} = 1.23$$

3.3.2 Stage I

Stage I uses a Vulcain 2 Engine LOX-LH2 combination as fuel
Calculations for Vulcain 2 Engine of Stage-I

- Sea Level Thrust (given in data)

$$F_{sea} = 960 kN$$

- Vacuum Thrust (given)

$$F_{vaccum} = 1390 kN$$

- Formula for sea level Thrust,

$$F_{sea} = \dot{m} \times V_2 + (P_2 - P_3) \times A_2$$

$$F_{vacuum} = \dot{m} \times V_2 + (P_2 - 0) \times A_2$$

- Chamber Pressure is given to be

$$P_1 = 117.3 \times 10^5 Pa$$

- Nozzle Area Expansion ratio is given

$$\epsilon = \frac{A_2}{A_t} = 58.2$$

- Nozzle Exhaust Pressure can be calculated from the following equation,

$$\frac{A_t}{A_2} = \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}} \left(\frac{P_2}{P_1}\right)^{\frac{1}{k}} \sqrt{\frac{k+1}{k-1}\left(1 - \left(\frac{P_2}{P_1}\right)^{\frac{k-1}{k}}\right)} = \frac{1}{\epsilon} = 0.018$$

$$\Rightarrow \frac{P_2}{P_1} = 0.0011$$

$$\Rightarrow P_2 = 12903 Pa$$

- Nozzle Exit Area, is given to be

$$A_2 = 3.43 m^2$$

- We can calculate mass flow rate as follows:

$$\dot{m} = \frac{M_p}{t_b} = \frac{174000}{540} = 322.23 kg/s$$

- From the values of $\dot{m}$, P_2 and A_2, we can calculate relative exhaust velocity, of the stage

$$V_2 = \frac{F_{vacuum} - P_2 A_2}{\dot{m}} = \frac{1390000 - 12903 \times 3.43}{322.23}$$

$$V_2 = 4176.34 m/s$$

- Nozzle area expansion ratio is given and can be used to calculate throat area,

$$\epsilon = \frac{A_2}{A_t} \implies A_t = \frac{A_2}{\epsilon} = 3.43/58.2 = 0.059 m^2$$

- Chamber Pressure is given to be

$$P_1 = 117.3 \times 10^5 Pa$$

- Characteristic Velocity can now be calculated using the values of $\dot{m}$, P_1 and A_t, by using the following equation:

$$\dot{c^*} = \frac{P_1 \times A_t}{\dot{m}} = \frac{117.3 \times 10^5 \times 0.059}{322.23} = 2147.75 m/s$$

- Molecular Weight of 6.7 MR LOX-LH2

$$5O_2 + 12H_2 \Longrightarrow 10H_2O + 2H_2$$

$$Avg.\,Molecular\,weight = \frac{5 \times 32 + 12 \times 2 + 10 \times 18 + 2 \times 2}{5 + 12 + 10 + 2} = \frac{368}{29}$$

$$MW = 12.69\,gm/mol$$

- Specific Heat Constant,

$$k = 1.2(for\ high\ temperatures)$$

- Gas Constant of xxx,

$$R_{xxx} = \frac{R_u}{MW} = \frac{8314}{12.69} = 655.16J/Kg - K$$

- Using the following formula for characteristic velocity, combustion chamber pressure, T_1 can be calculated as follows by substituting the values of R_{xx}, k, and c^*:

$$c^* = \frac{\sqrt{k \times R_{xx} \times T_1}}{k\sqrt{\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}} \Longrightarrow \frac{\sqrt{1.2 \times 655.16_1}}{1.2\sqrt{\left(\frac{2}{1.2+1}\right)^{\frac{1.2+1}{1.2-1}}}} = 2147.75 \Longrightarrow T_1 = 2961.296K$$

- Finally, Sea level thrust coefficient is given by:

$$C_F = \frac{F_{sea}}{P_1 A_t} = \frac{960000}{117.3 \times 10^5 \times 0.059} = 1.387$$

3.3.3 Stage II

Stage II consists of the HM7B engine which uses the LOX-LH2 propellant combination
Calculations for HM7B engine Stage-II

- Sea Level Thrust, (given)

$$F_{sea} = \frac{M \times Isp \times g}{t_b} = 67KN$$

- Vacuum Thrust, given,

$$F_{vaccum} = 64.8kN$$

- Formula for sea level Thrust,

$$F_{sea} = \dot{m} \times V_2 + (P_2 - P_3) \times A_2$$

$$F_{vacuum} = \dot{m} \times V_2 + (P_2 - 0) \times A_2$$

- Chamber Pressure is given to be

$$P_1 = 37 \times 10^5 Pa$$

- Nozzle Area Expansion ratio is given

$$\epsilon = \frac{A_2}{A_t} = 83.1$$

- Nozzle Exhaust Pressure can be calculated from the following equation,

$$\frac{A_t}{A_2} = \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}} \left(\frac{P_2}{P_1}\right)^{\frac{1}{k}} \sqrt{\frac{k+1}{k-1}\left(1 - \left(\frac{P_2}{P_1}\right)^{\frac{k-1}{k}}\right)} = \frac{1}{\epsilon} = 0.012$$

$$\Rightarrow \frac{P_2}{P_1} = 0.0007$$
$$\Rightarrow P_2 = 2590 Pa$$

- Nozzle Exit Area, is given to be

$$A_2 = 0.773 m^2$$

- We can calculate mass flow rate as follows:

$$\dot{m} = \frac{M_p}{t_b} = \frac{14900}{945} = 15.77 kg/s$$

- From the values of $\dot{m}$, P_2 and A_2, we can calculate relative exhaust velocity, of the stage

$$V_2 = \frac{F_{vacuum} - P_2 A_2}{\dot{m}} = \frac{64800 - 2590 \times 0.773}{15.77}$$
$$V_2 = 3982.11 m/s$$

- Nozzle area expansion ratio is given and can be used to calculate throat area,

$$\epsilon = \frac{A_2}{A_t} \implies A_t = \frac{A_2}{\epsilon} = 0.773/83.1 = 0.0093 m^2$$

- Characteristic Velocity can now be calculates using the values of $\dot{m}$, P_1 and A_t, by using the following equation:

$$\dot{c^*} = \frac{P_1 \times A_t}{\dot{m}} = \frac{37 \times 10^5 \times 0.0093}{15.77} = 2181.99 m/s$$

- Molecular Weight of 4.9 MR LOX-LH2

$$5O_2 + 16H_2 \Longrightarrow 10H_2O + 6H_2$$

$$Avg.\,Molecular\,weight = \frac{5 \times 32 + 16 \times 2 + 10 \times 18 + 6 \times 2}{5 + 16 + 10 + 6} = \frac{384}{37}$$

$$MW = 10.38\,gm/mol$$

- Specific Heat Constant,

$$k = 1.2\,(for\ high\ temperatures)$$

- Gas Constant of xxx,

$$R_{xxx} = \frac{R_u}{MW} = \frac{8314}{10.38} = 800.96 J/Kg - K$$

- Using the following formula for characteristic velocity, combustion chamber pressure, T_1 can be calculated as follows by substituting the values of R_{xx}, k, and c^*:

$$c^* = \frac{\sqrt{k \times R_{xx} \times T_1}}{k\sqrt{\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}} \Longrightarrow \frac{\sqrt{1.2 \times 800.96 \times T_1}}{1.2\sqrt{\left(\frac{2}{1.2+1}\right)^{\frac{1.2+1}{1.2-1}}}} = 2181.99 \Longrightarrow T_1 = 2500.09K$$

- Finally, Sea level thrust coefficient is given by:

$$C_F = \frac{F_{sea}}{P_1 A_t} = \frac{67000}{37 \times 10^5 \times 0.0093} = 1.947$$

4 Summary and Observations

Following are the major observations and conclusions derived from the study and analysis of the propulsion system of ARIANE 5 ECA launch vehicle

Propulsion System performance calculations of ARIANE 5ECA			
Stage No.	0	I	II
Gross Mass, M_0 (Tonnes)	761.09	215.09	26.39
Burnout Mass, M_b (Tonnes)	281.09	41.09	11.49
Propellant Mass M_p (Tons)	480	174	14.9
Structural Mass M_s (Tonnes)	33	14.7	4.54
I_{sp}(sl), (sec)	274.5	310	446
I_{sp}(vacuum)	NA	432	NA
Burn Time, t_b, (sec)	130	540	945
Mass Ratio, R	2.708	5.235	2.297
Propellant Mass Fraction	0.631	0.809	0.565
Pay Load Mass Fraction	1.394	1.140	0.358
Structural Coefficient	0.121	0.078	0.234
Altitude, (km)	0	0	164
Local g value, (m/s^2)	9.8	9.8	9.3
Equivalent Velocity, c, (m/s)	2692.845	3041.1	4375.26
ΔU, (m/s)	2682.64	5034.14	3638.5
Total Impulse, I_t (MNsec)	646.28	529.15	65.19
Thrust (kN)	4971.41	979.91	68.99
T/W_0	1.33	0.46	0.27
T/W_b	3.61	2.43	0.61
Max acceleration (g)	25.56	14.04	6.00

Thrust to Weight Ratio

We observe that the thrust to weight ratio for stages I and II are less than one These values should ideally be greater than 1, but the discrepancies arise due to the following reasons:

- The first two stages of Ariane 5 ECA are parallely staged. Stage 0 and Stage-I fire simultaneously at lift-off, and the boosters (Stage-0) separate at 130 s after EAP flame-out. The main cryogenic stage continues to fire for the next 410s until burn-out ans subsequent separation. For simplicity of analysis, we have considered all the stages to be staged serially i.e. The Boosters fire at lift-off and Stage-I fires after the boosters are separated. Due to this assumption, the propellant mass of Stage-1 considered in our calculations is much larger than the actual mass. Hence the thrust to weight ratio is less than 1.

- Actual trajectories of launch vehicles are not completely vertical. Most launch vehicles employ gravity turn manoeuvres to gradually change the angle of inclination of the launch vehicle. This is done because the launch vehicle needs to be oriented almost horizontally at the time of orbit injection. For simplicity of analysis, we have assumed strictly vertical motion of the launch vehicle during the booster stage and Stage I operation.

Sea Level Thrust

Sea level Thrust Comparison			
Stage No.	0	I	II
Thrust, given (kN)	5400*2	960	67
Thrust, calculated (kN)	4971.41*2	979.91	68.99

- We observe a very close comparison of our calculations and the data provided.
- There are slight discrepancies in the values, but these are expected since the manufacturers don't always provide complete data to the public which can be used to calculate exact thrust.

Mass Flow Rate

Mass Flow Rate Comparison			
Stage No.	0	I	II
$\dot{m}$, given (kg/s)	2000	323	14.8
$\dot{m}$, calculated (kg/s)	1846.15	323.23	15.77

- We observe a very close comparison of our calculations of the mass flow rate and the data provided by the manufacturers
- Our calculations of mass flow rate made use of the assumption that the burn rate of the propellants in all stages is constant. The fact that this yields results close to actual values provided, leads us to conclude that the actual engines also work on the principle of constant burn rate.

5 References

1. $https://en.wikipedia.org/wiki/Ariane_5Solid_propellant_stage$
2. $https://www.arianespace.com/vehicle/ariane-5/$
3. $https://www.researchgate.net/figure/Parameters-of-main-stage-EPC-H-173-40-41_tbl1_303828811$
4. $https://www.esa.int/esapub/bulletin/bullet93/b93carr.htm$
5. $https://www.esa.int/esapub/bulletin/bullet102/Coulon102.pdf$
6. $https://www.arianespace.com/wp-content/uploads/2016/10/Ariane5-users-manual-Jun2020.pdf$
7. $https://www.esa.int/Enabling_Support/Space_Transportation/Launch_vehicles/Ariane_{5ECA}$
8. $https://www.spacelaunchreport.com/ariane5.html$
9. $https://www.esa.int/esapub/achievements/Sc72s6.pdf$
10. European Space Agency