SOME TYPICAL SOLID PROPELLANT ROCKET MOTORS

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Challenge the future

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Preface

This document is published within the framework of a Lecture Series on Chemical Rocket Propulsion at TU-Delft, Faculty of Aerospace Engineering and intends to provide the (future) propulsion engineer with a starting point for practical solid propellant rocket motor engineering.

This document is intended as a lively document. Hence, I would like to encourage any reader to provide the author with 'missing' information and/or suggestions for improvement of this document.

In addition, the author wishes to thank Ernst Hesper of TU-Delft, Faculty of Aerospace Engineering, for the careful proofreading of version 1.0 of this publication and for providing many useful comments.

Version 2.0 differs from version 1.0 in that two additions (Vega P80 and Ariane 4 separation motors) and some textual improvements have been made.

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Total number of pages: 40

List of acronyms

AISI	American Iron and Steel Institute
Al	Aluminium
AP	Ammonium Perchlorate
ASRM	Advanced Solid Rocket Motor
CTPB	Carboxy-Terminated PolyButadiene
EPDM	Ethylene Propylene Diene Polymer
ESA	European Space Agency
HTPB	Hydroxyl-Terminated PolyButadiene
MAGE	Moteur d'Apogée Géostationnaire Européen
MEOP	Maximum Expected Operating Pressure
MPS	Moteur Propergol Solide
NASA	National Aeronautics and Space Administration
PAP	Propulseur d'Appoint à Poudre
PBAN	PolyButadiene acrylic acid Acrylo-Nitrile
PSLV	Polar Satellite Launch Vehicle
S&A	Safe & Arm
SEP	Société Européenne de Propulsion (France)
SL	Sea Level
SRB	Space Shuttle Solid Rocket Booster
SRM	Solid propellant Rocket Motor
STAR	Spherical Thiokol Apogee Rocket
TVC	Thrust Vector Control
TBI	Through Bulkhead Initiator
USA	United States of America
US$	US dollar
UT CSD	United Technologies, Chemical Systems Division (USA)

Introduction

In this document, some typical Solid propellant Rocket Motors (shortly referred to as solid rocket motors; SRM's) will be described with the purpose to form a database, which allows for comparative analysis and applications in practical SRM engineering.

SRM's use a solid propellant[1] to provide the energy required for the generation of the required propulsive force (thrust). The solid propellant charge, in the form of one or several shaped blocks which are called grains, usually is stored and combusted in a so-called solid propellant combustor. The grain shape largely determines the thrust history of the motor. Large rocket motors, generally use a segmented grain to allow for ease of manufacturing. Smaller rocket motors usually are of a single-case design. Ignition is accomplished by a separate ignition system. The igniter, when lit, sends burning particles into the main propellant grain and fully ignites the rocket motor. SRM's typically operate with a single start and burn until the propellant is gone.

The solid propellants used mainly are either double-base or composite type of propellants. In a double-base propellant, both fuel and oxidizer belong to the same molecule. Typical examples are nitrocellulose and nitro-glycerine and/or a mixture of these. In a composite propellant, oxidant and fuel are separate compounds intimately mixed together. Typical composite propellants use powdered Aluminium (Al) as fuel and Ammonium Perchlorate (AP) as oxidizer. A synthetic rubber or plastic holds the fuel and oxidizer powder together. Nowadays, for space applications, mostly composite propellants are used. This is, because composite propellants allow for high vehicle performance (in terms of payload mass).

The first use of (high performance) SRM's dates back to as early as 11 October 1958, when the X-248 Altair I, which was used to provide part of the propulsive force to propel NASA's Pioneer I Lunar Probe, successfully completed its task. Since that date, over 2000 SRM's have been utilized in space programs for propulsion applications. Typical such applications are for boost[2] propulsion of launchers, upper stage or kick stage propulsion of space probes, and re-entry vehicle propulsion. Besides these, there are a number of special purpose applications, such as stage separation, liquid propellant pressurization, spin up and spin down, and liquid propellant settling.

Compared to current liquid propellant rocket motors, SRM's offer a specific impulse, which is somewhat lower, even when using composite propellants. In addition, SRM's are more difficult to throttle, and because, for a solid propellant, fuel and oxidiser are intimately mixed there is some potential for detonation of the propellant. The latter requires extensive safeguards during propellant manufacturing as well as launcher- and payload processing. Finally, and of a more recent nature, there is some concern about the effect that SRM's have on the environment [McDonald et al, 1991].

[1] With the term 'propellant', usually a combination of fuel and oxidiser is meant.

[2] Rocket motors, which provide a rapid acceleration of the launch vehicle during the launch phase, are, generally, referred to as booster motors.

The unique characteristic, however, that distinguishes an SRM from a liquid propellant rocket motor is the extremely simple way of controlling the energy release process. This, in general, leads to a highly reliable and low cost motor (with a reliability of 0,998 versus 0,985-0,989 for liquid propellant rocket motors and a cost of US$ 0,034 per Ns of total impulse vs. US$ 0,090 [Andrews and Haberman, 1991]). Also SRM's are capable of delivering high thrust more easily than liquid propellant rocket motors. Finally, solid propellants can be more easily stored than liquid propellants. It is basically these features, which make SRM's very attractive for space applications[3].

In this document, several SRM's are described, including large solid booster rocket motors, solid kick motors, as well as some special purpose motors. Each SRM is described in a separate chapter. For each SRM the following items will be discussed (depending on the available information):

- General items like costs[4], layout, type of case, igniter and nozzle, and grain configuration.
- Geometrical data: overall length, maximum diameter and nozzle expansion ratio.
- Propulsive performances:
  - Average thrust over the burn time. In general, it is found that the thrust increases with increasing altitude. The difference between Sea Level (SL) thrust and vacuum thrust typically is about 10% (depending on the geometric expansion ratio of the nozzle).
  - Specific impulse as a measure for propellant consumption; the higher the specific impulse, the lower the propellant consumption. Like the thrust, the specific impulse increases with increasing altitude.
  - Total impulse, defined as the integration of the thrust over the burn time, as a measure for the change in momentum, which can be accomplished.
  - Burn time or burn duration. The burn time is defined as the action time minus the ignition time and the time required for thrust tail-off[5].
  - (Motor) Propellant fraction, i.e. propellant-to-total motor mass ratio. This parameter allows for judging the quality of a SRM. Since this ratio is almost 1, this parameter can also be used as a starting point for mission calculations using e.g. Tsiolkovsky's relation.
- Propellant type, (mass) density and ingredients (in mass percentages).
- Materials.
- Mass characteristics.
- Ballistic performances, like the maximum and average chamber pressure and the burning rate of the propellant, i.e. regression rate.

In this document no attention is given to motor response data and thermal data. Both, however, are of importance for judging e.g. the structural integrity of the motor, but are considered to be too detailed for the purpose of this document.

[3] Although SRM's are used extensively for space applications, the main field of applications is still military.

[4] Most cost data have been taken from the work of [Koelle, 1995] and are given in man-year. This is because these data are considered to be valid internationally and independent from annual changes such as inflation rate.

5 Thrust tail-off

For further explanation of the above-mentioned parameters and items, the reader is referred to the general literature on (solid propellant) rocket propulsion.

As a general reference for most data the [interavia Space directory, 1990-1991] is consulted, except for most cost data, which have been taken from [Koelle, 1995]. Specific literature used is indicated at the end of each chapter. In addition, some data have been estimated using simple formula. These data are indicated in the text by "(e)".

Most data reported in this document, except when indicated otherwise, are valid for *standard* day conditions only. This is important, because hot or cold day conditions can lead to a significant change in propellant regression rate, i.e. the burn rate. This not only causes a change in combustion pressure, which can lead to a further change in regression rate, but moreover, causes a change in mass flow (and hence in thrust) and burn time. Note that the overall effect on total impulse remains small, because, if thrust increases through an increase in propellant mass flow, the burn time decreases and vice versa.

1. Ariane 4 segmented PAP

The European Space Agency's (ESA) Ariane 4 space rocket launcher can be equipped with two or more large segmented solid rocket strap-on[6] booster motors, to provide high thrust during the initial launch phase. These motors are often referred to as Propulseur d'Appoint à Poudre, or shortly as PAP. Ariane 4's PAP, basically, is an advanced version of Ariane 3's PAP. 2 to 4 of such PAP's are carried by the Ariane 42P, 44P and 44LP variants.

The total development effort for this motor is estimated at 700-800 man-year. Manufacturer of Ariane 4 PAP's is BPD of Italy. Some principal features of this motor are shown in Fig. 1.1. Characteristic motor data are given in Table 1.1.

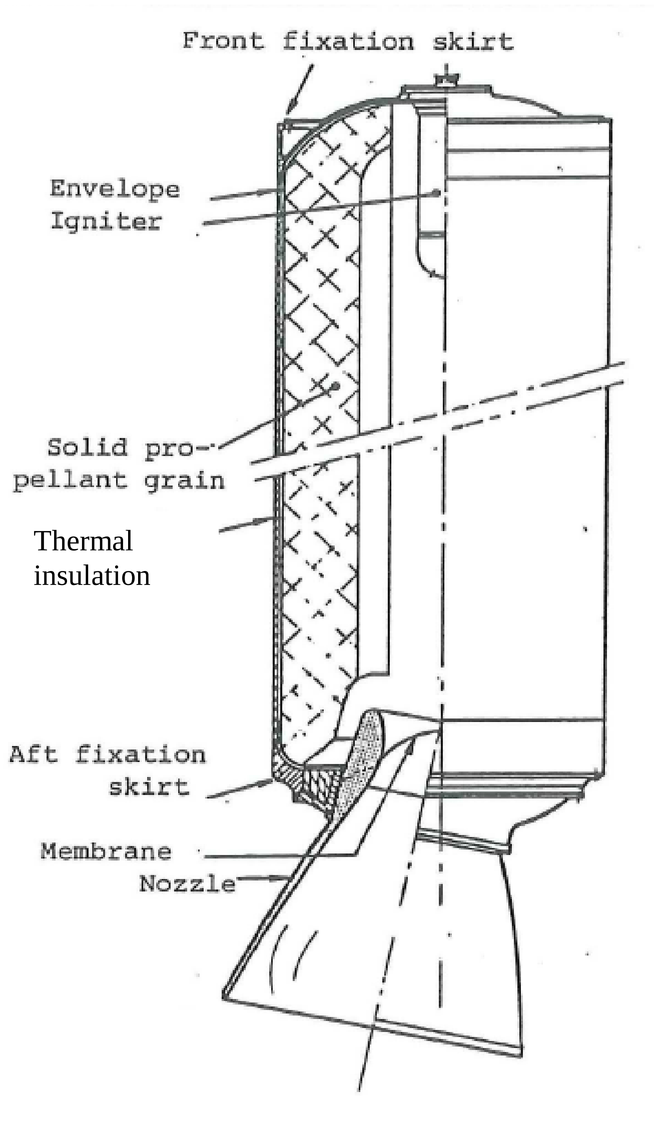

Figure 1.1: Propulseur d'Appoint a Poudre

Each Ariane 4 PAP motor has a mass of 11,57 tons[7] and delivers an average sea level thrust of 650 kN at an estimated average sea level specific impulse of 233 s. The average burn time of the motor is about 33 seconds. The sea level total impulse delivered during burning is estimated at about 21,5 MNs. Typical operation range is from +0 to +40 °C.

The motor basically consists of a cylindrical case, a nozzle, and a forward and aft dome. The latter two hold the front and aft fixation skirt. Its overall diameter is 1,07 m and its overall length is 11,32 m.

[6] Strap-on rocket motors are rocket motors, which are attached to the sides of the core vehicle (parallel staging concept).

[7] Fully equipped PAP stage (with nose cap, aft skirt and separation system) has a mass of 12.56 tons.

The motor case is made out of AISI 4130 steel[8]. The front and aft dome are forged and machined and parts are welded together. The complete case is subjected to a heat treatment to relax the internal tensions of the material.

The propellant is a composite propellant (trade name: Flexadine 1613), produced by BPD under license of Rocketdyne (USA) and consists of 13% Carboxi-Terminated Poly Butadiene (CTPB[9]) and 16% Aluminium (Al) as the fuel, and 71% Ammonium Perchlorate (AP) as the oxidiser. The propellant is cast into the case around a star-shaped bore in order to obtain a neutral or slightly progressive thrust programme[10]. The propellant curing process takes 16 days of which 15 at an elevated temperature of 80°C. Thermal insulation of the case is applied to protect the casing from the high temperatures during combustion.

Sea Level Total Impulse, (MNs)	21,5 (e)
Av. Sea Level Thrust, (kN)	650
Sea Level Specific Impulse, (s)	233 (e)
Burn Time, (s)	33
Motor Diameter, (m)	1,07
Motor Length, (m)	11,32
Approx. Total Motor Mass, (ton)	11,57
Approx. Usable Propellant Mass, (ton)	9,64
Thrust Vector Control	No

Table 1.1: Ariane 4 SRM Characteristics

The propellant is ignited by a pyrogen igniter[11] (Fig. 1.2), which is mounted in the forward dome[12]. It basically consists of an initiator, a booster charge and a main charge. The initiator ignites the booster charge, which in turn ignites the main charge. Finally, through holes in the igniter, the hot igniter gases are led over the solid propellant surface.

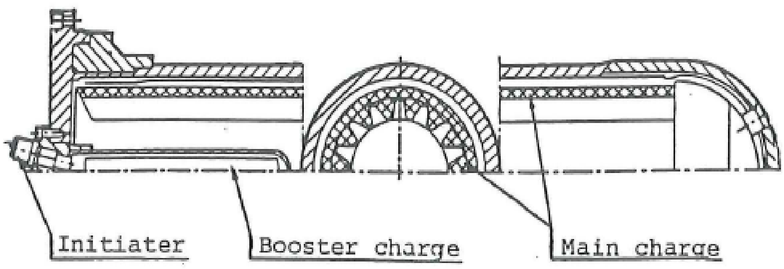

Figure 1.2: Pyrogen igniter used in PAP

[8] It has been estimated that the use of composite materials reduces the mass of the case by about 15%.

[9] CTPB basically is a hydrocarbon binder material.

[10] By choosing a proper shape of the propellant grain, it is possible to vary the propellant mass flow in time and hence also the thrust. This generally is referred to as thrust programming. The resulting thrust-time history is referred to as the thrust program. A thrust program with increasing thrust in time is referred to as a *progressive* thrust program; this opposed to a *regressive* thrust program. When the thrust remains constant, one generally speaks of a *neutral* thrust program.

[11] A pyrogen igniter basically is a small rocket motor that is used to ignite a larger rocket motor.

[12] An igniter, mounted in the forward dome of the case usually is referred to as a head-end igniter.

The nozzle is mounted in the aft dome and is canted away from the launcher at an angle of 14^{o}, causing the thrust vector of the motor to point towards the center of mass of the complete launch vehicle. The nozzle is made of carbon-phenolic material with a steel (identical steel as for the case) reinforcement near the throat. The expansion ratio is 8,3:1 and the exit diameter 0,958 m. Fig. 1.3 shows a cross-section of the nozzle. To limit the length of the motor, the nozzle is partly submerged into the combustion chamber. A membrane is mounted in to the nozzle to ensure a proper pressure build-up during ignition of the PAP and to prevent contamination of the motor prior to the start.

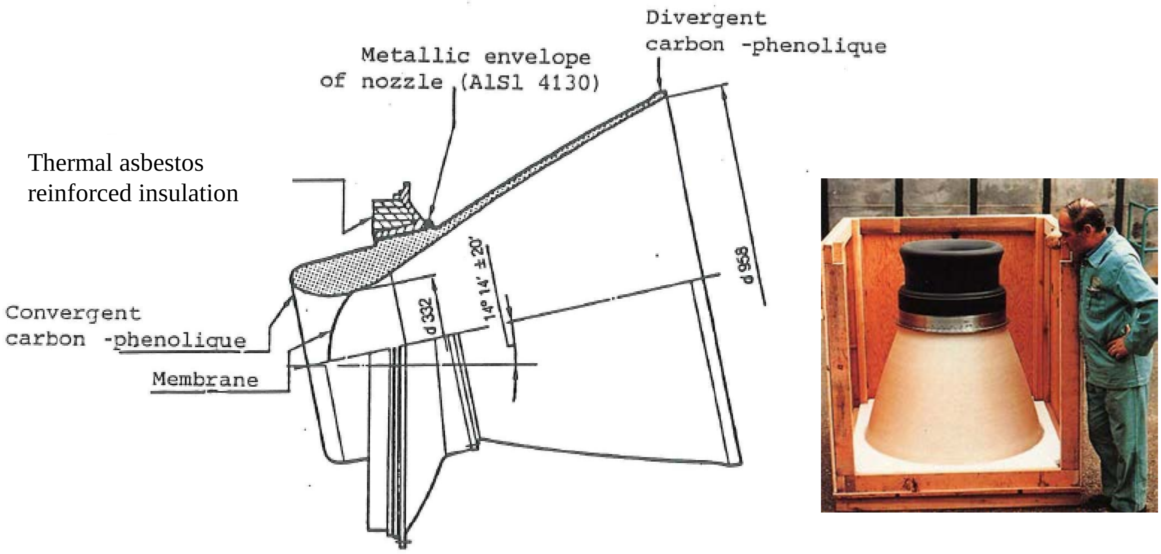

Figure 1.3: Nozzle of PAP

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   Chemical Rocket Propulsion, Lecture Series D-35, Delft University of Technology, Faculty of Aerospace Engineering.

2. Ariane 5 P-230 segmented SRM

The P-230 SRM, also referred to as Moteur Propergol Solide (MPS), is the strap-on booster motor for the European Ariane 5 launcher. It has been developed by GIE Europropulsion, a joint venture between SNIA BPD (Italy) and SEP (France) at an estimated cost of about 6000 man-year [Koelle, 1995]. Like so many large SRM booster motors, it too is of a segmented design. Some typical characteristics of the MPS are listed in Table 2.1, while the MPS itself is shown in Figure 2.1.

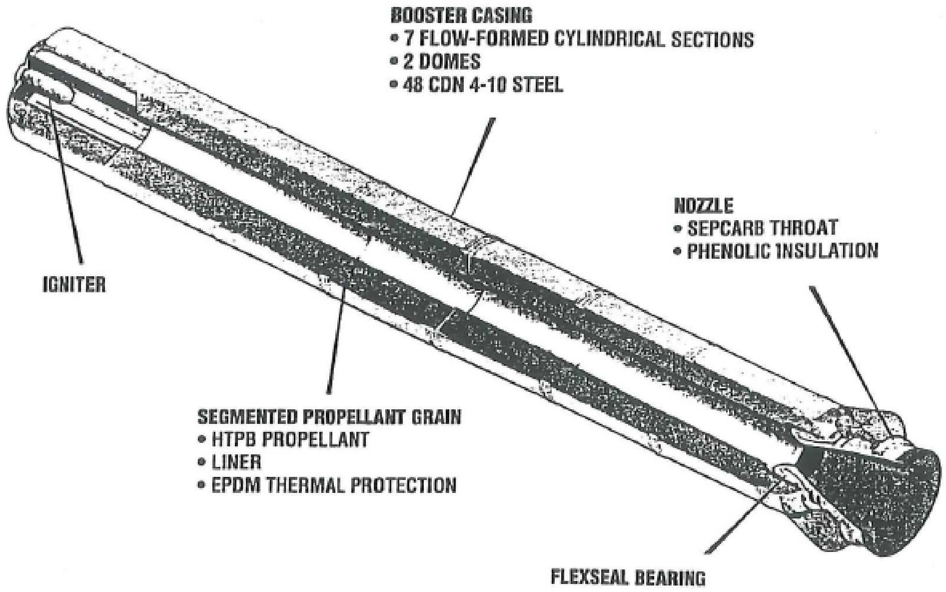

Figure 2.1: P-230 Solid Rocket Motor (MPS)

The 26,77 m long MPS motor is capable of delivering a total impulse in vacuum better than 610 MNs. It delivers an instantaneous 6,0 MN of vacuum thrust at ignition (5,4 MN sea level thrust), building to around 6,7 MN after 20 s. By between 30-40 s the top segment burns out and thrust is down to approximately 4,0 MN, building gradually back up to around 6,0 MN at 110 s. Over the next 20 seconds the thrust tails of to zero as the motor burns out. The motor has a vacuum specific impulse of 275,4 s. The nominal burn time of the motor is 129 s. Its mass is 269 ton[13] of which approximately 88,4% is propellant (237,7 ton). Operation range is from +0 to +40 °C.

Each MPS consists of 6 main cylindrical segments plus a forward segment with a domed closure and an aft dome with the (submerged) rocket nozzle. The aft cylindrical segment, also referred to as the attachment segment, holds an attachment ring for the connection of the MPS to the launch vehicle. The aft dome has a large attachment flange (2,05 m diameter), which holds the nozzle. The length of the case is 24,77 m and its diameter is 3,05 m. Its total mass is 19700 kg.

[13] Fully equipped EAP (including nose cap, front skirt and separation system) is 276 tons.

Average Vacuum Total Impulse, (MNs)	643 (e)
Average Vacuum Thrust, (kN)	4984
Sea Level Specific Impulse, (s)	275,4
Burn Time, (s)	129
Max pressure (MPa)	6,1
Motor Diameter, (m)	3,05
Motor Length, (m)	27,34
Approx. Total Motor Mass, (ton)	269,0
Approx. Usable Propellant Mass, (ton)	237,7
Thrust Vector Control	6 deg. Pitch/yaw

Table 2.1: Ariane 5 MPS Characteristics

All cylindrical segments have a case roll formed from (D6AC) steel forgings. The case is weld free with a nominal wall thickness of 8 mm; its maximum pressure limit is 6,1 MPa (safety factor is 1,25). Tang-and-clevis joints (see next entry) with two elastomer O-ring seals connect the segments. A layer of synthetic rubber insulation (silica-filled or aramid-filled) protects the inner surfaces of the steel segment cases. The total mass of this thermal protection is 4900 kg.

The MPS composite propellant is a mixture of synthetic rubber (Hydroxyl Terminated Poly-butadiene or shortly HTPB)[14] and aluminium (Al) powder as fuel and ammonium perchlorate (AP) as oxidiser, in a 14:18:68 HTPB:Al:AP mix. The propellant has a density of 1770 kg/m^3 and burns at a rate of 7,4 mm/s (at 27^{o}C and 6,1 MPa). The latter is ensured by adding a small amount of burning rate catalyst. The case-bonded propellant grain is tubular shaped with an almost cylindrical central bore with a diameter of 1,265 m, which forms the main burning surface. Only the most forward segment is configured to a 15-star shape and has a slightly faster burning propellant to allow providing the required thrust level.

The pyrogen igniter is mounted internally at the forward end of the forward dome. It consists of an electrically ignited pyrotechnic igniter, containing a pyrotechnic mixture of Boron as fuel and potassium nitrate ($B\text{-}KNO_3$) in the form of small pellets, which ignites the main igniter initiator. The latter in turn ignites the main igniter. The main igniter- and main igniter initiator propellant is the same. It is, basically, a composite propellant with 14% of aluminium and 14% of HTPB-binder. Total igniter mass is 315 kg of which about 65 kg is propellant. Igniter envelope has a diameter of 0,47 m and a length of 1,18 m.

[14] HTPB is known to offer a larger solids loading than other binder materials, like carboxy-terminated poly-butadiene (CTPB) and polybutadiene acrylonitrile (PBAN). In addition, HTPB offers both superior mechanical properties and good process ability.

The rocket nozzle is manufactured by SEP and has a mass of 6400 kg. It has a carbon throat and a carbon phenolic exit cone inside a steel and steel-faced aluminium honeycomb outer structure. The submerged nozzle is 3,78 m long with a 0,900 m throat diameter and a 2,99 m exit diameter, which gives a 11,0:1 expansion ratio.

For Thrust Vector Control (TVC), the nozzle can be moved through 6^{o} on a ring-segmented rubber and steel flex bearing.

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   Solid-propellant stage development for Ariane-5, ESA-bulletin 69 (Also published as AIAA paper 92-156).

2. Donaldson, P.
   Boosting Ariane 5, Space, May 1993.

3. Fabrizzi, R.
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4. Coutrot, A.
   Ariane 5 solid propellant boosters, SEP.

5. Gonzalez-Blasquez, A, and Constanzo, A.
   First Test Firing of an Ariane-5 Production Booster, ESA Bulletin 104, November 2000.

3. Commercial Titan 3 segmented SRM

The Commercial Titan 3 launcher uses two large segmented (five and a half-segment) solid rocket motors to provide the bulk of thrust during the initial launch phase of the vehicle. These segmented strap-on motors are manufactured by Chemical Systems Division (CSD) of United Technologies Corporation (UTC), using CSD's Titan 3 five-segment motor as a starting point. The latter has been used for more than 65 times to successfully assist the launch of the Titan 3C/D/E Space Launch Vehicles (not to be confused with the commercial Titan 3[15]) in to orbit.

The first flight of the commercial Titan 3 five and a half-segment motor, hereafter shortly referred to as SRM, occurred in 1982, approximately 20 years after the start of the development of the initial Titan 3 strap-on motor (five-segment version). Total development of this SRM as given by Koelle is ~5000 man-year. Some principal features of this MPS are shown in Fig. 3.1. Characteristic motor data are given in Table 3.1.

Each Commercial Titan 3 (5½-segment) motor has a mass of ~238 ton[16] and produces an estimated average vacuum thrust of 4,8 MN and a total impulse of 545 MNs (in vacuum). The vacuum specific impulse is 265,2 s. The burn time is 113,7 s. About 88,5% of the 238 ton mass of each SRM is propellant.

Each motor is segmented for ease of rail transport from the manufacturer to the launch site, and consists of five cylindrical segments with a length of 3,20 m each, one cylindrical segment 60% of the normal length, plus a forward segment with a domed closure and an aft segment with the rocket nozzle and an aft skirt that incorporates launch pad mounting structures. Motor diameter is 3,11 m and overall length is 27,57 m.

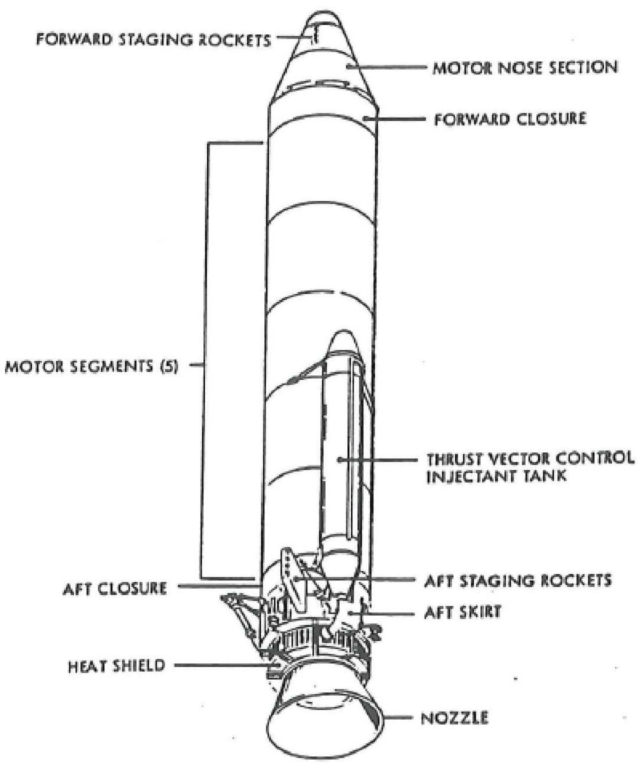

Figure 3.1: Original Titan 3 SRM (5-segments)

[15] The commercial Titan 3 space launcher its the latest version of the Titan 3, which has been derived from the Titan 34D space launcher, which in itself was a successor to the Titan 3C/D/E series.

[16] The total mass of the strap-on stage is 250 ton.

Vacuum Total Impulse, (MNs)	545
Av. Vacuum Thrust, (MN)	4,8(e)
Vacuum Specific Impulse, (s)	265,2
Burn Time, (s)	113,7
Motor Diameter, (m)	3,11
Motor Length, (m)	27,57
Approx. Total Motor Mass, (ton)	238
Approx. Usable Propellant Mass, (ton)	210,6
Thrust Vector Control	7 deg. Pitch/Yaw

Table 3.1: Commercial Titan-3 SRM Characteristics

All cylindrical segments have a case shear-spun from (D6aC)[17] steel ring forgings, thus eliminating the welded joints used in the original Titan 3 strap-on design, with a length of 3,2 m and a wall thickness of less than 12,7 mm. The segments are connected by clevis joints of the type made famous by the Challenger disaster; basically tongue-and groove joints, see Fig. 3.2, held together by 240 steel pins with an O-ring seal. A layer of synthetic rubber insulation (silica-filled buna-N rubber) protects the inner surfaces of the segment cases.

The propellant is a mixture of synthetic rubber (PolyButadiene acrylic acid Acrylo-Nitrile or PBAN) and Al powder as fuel and AP as oxidiser, in a 30:70 fuel:oxidiser mix. Maximum combustion pressure is 5,8 MPa. For propellant casting, the segments are fitted with a mandrel that is removed, once the propellant is set, to leave a cylindrical central bore 1,2 m in diameter that forms the main burning surface. The upper surface of the propellant grain in each segment is covered in inert rubber to inhibit burning, but the lower surface of the grain is an extra burning surface. This arrangement gives a regressive thrust variation as the area of burning surface reduces as propellant is consumed.

The igniter system is mounted in the forward dome (head-end pyrogen igniter) and produces a flame that washes the entire length of the bore. It consists of three rockets, which burn after each other. The igniter uses a staged ignition sequence in which the first and smallest rocket is ignited electrically. This rocket in turn ignites the second somewhat larger rocket and so on until finally the third and largest rocket ignites the rocket motor.

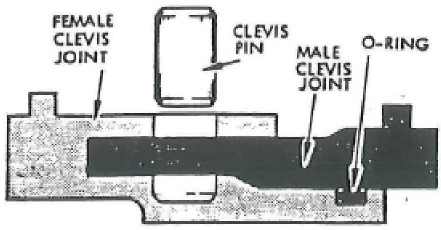

Figure 3.2: Clevis joint

The rocket nozzle is designed to withstand two minutes burn time. It has a carbon throat and a silica phenolic exit cone inside a steel and steel-faced aluminium honeycomb outer structure. The expansion ratio is 10:1 and the exit diameter is 3,05 m. The nozzle is canted 6^o from the axis of the SRM so that the un-deflected thrust is in the direction of the centre of gravity of the Titan 3 stack.

[17] Some typical properties of D6 AC steel are: Ultimate Tensile Strength of 1344 MN/m^2, Yield Strength of 1241 MN/m^2 and a toughness of 100 MN/m$^{(-3/2)}$. In addition, the cost of D6AC steel is lower than 15 CDV and maraging (M)250 steel [Nagappa, 1989].

Thrust Vector Control (TVC) is achieved by injecting a fluid (nitrogen tetroxide) into the nozzle gas flow downstream of the throat and thus creating a shock wave that deflects the flow in the required direction. Twenty-four valves in four banks of six are mounted round the nozzle to control the fluid injection (Fig. 3.3). The TVC fluid is carried in a 1,07 m diameter cylindrical tank (capacity of 3630 kg) mounted alongside the three aft segments of the SRM. A nitrogen gas tank mounted above the fluid tank serves to pressurise the TVC system. With all six valves in one bank fully open a thrust deflection of 7^{o} can be attained. At zero deflection all the valves are partially opened to consume the fluid mass, and in this state the nitrogen tetroxide adds about one percent to the specific impulse of the motor.

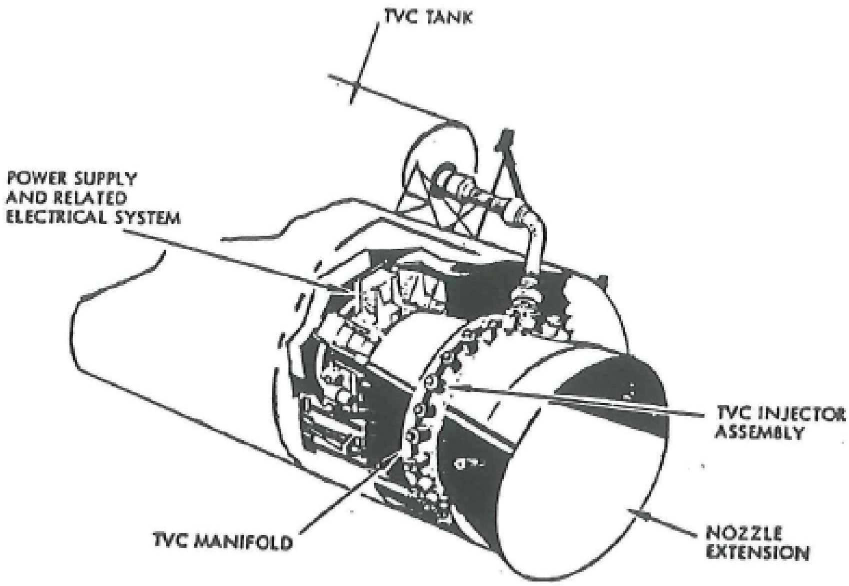

Figure 3.3: TVC system

The forward dome includes two explosively jettisonable 0,9 m diameter plates that could be blown out to cancel the SRM thrust, if an emergency develops and the manned craft has to fire its escape rockets. This prevents a still-burning strap-on from catching and colliding with the escaping craft. The dome is enclosed in a conical aerodynamic fairing that also carries four of the eight 20 kN thrust, 1 s burn time, solid rocket motors used to separate the SRMs from the central core at burn-out. The remaining four rockets are mounted on the aft fairing surrounding the nozzle and the TVC valves. These rockets are modified Titan missile stage-separation motors, also manufactured by CSD of UTC. The SRM thrust is transmitted to the core by two steel cantilever structures mounted on the aft segment and connecting to the core first stage thrust structure. These structures also support the core stages while the vehicle is on the launch pad. SRM lateral position is maintained by two links on the forward segment connected to the core inter-stage structure.

Specific literature

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   Chemical Rocket Propulsion, Lecture Series D-35, Delft University of Technology, Faculty of Aerospace Engineering.

2. Richards, G.R., and Powell J.W.
   Titan 3 and Titan 4 space launch vehicles, Journal of the British Interplanetary Soc., Vol.46, pp.123-144, 1993.

4. Titan 4 segmented SRM

Doubts about the ability of United Technologies Chemical Systems Division (USA) to meet the required strap-on motor production rate for Titan 4, resulted in the United States Air Force seeking a second source at the same time specifying rather higher performance to allow for payload growth. In October 1987 Hercules Aerospace (USA) was selected to supply fifteen sets of up-rated strap-ons, known as SRMU for SRM Upgrade at a total cost of $960 million. Some principal features of the SRMU are shown in Fig. 4.1.

The new design has three 3,2 m diameter segments using filament-wound graphite-epoxy cases and HTPB-type of propellant. Propellant mass is about 330 ton and the empty mass at burn out about 45 ton. This gives an approximate total mass (neglecting the mass of inerts expended) of about 375 ton. TVC will be by hydraulically steerable nozzles rather than the fluid injection method of the CSD motors.

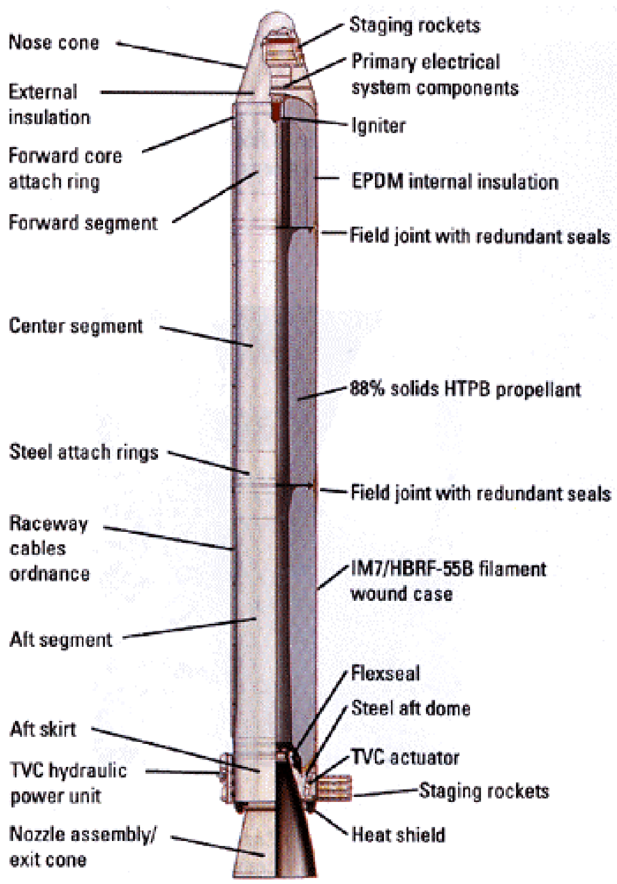


Figure 4.1: SRMU

The rocket nozzle is of a carbon/phenolic design with a 3-D carbon/carbon throat insert.

The first test firing of this motor was due to take place in the spring of 1990, but was delayed, partly by accidents including a fatal incident at Edwards when a segment ignited after being dropped when the crane lifting it collapsed. The test eventually took place on 1 April 1991, but the motor generated more internal pressure than expected due to unpredicted radial gas-flow effects and the casing ruptured after 2 seconds, causing considerable damage to the Edwards test stand. The first SRMU flight was scheduled for 1992, but was further delayed to 1994. The delay has led to studies of alternative types of thrust augmentation. Use of the SRMU's should give a geosynchronous payload capability of 5,6 ton.

Specific literature

1. Richards, G.R. and Powell J.W.
   Titan 3 and Titan 4 space launch vehicles, Journal of the British Interplanetary Soc., Vol.46, pp.123-144, 1993.

5. Space Shuttle segmented SRM (SRB)

The Space Shuttle SRB's are manufactured by Thiokol (Utah, USA) and are used as strap-on booster motor for the USA Space Shuttle System. According to [Koelle, 1995], its development has taken ~7000-8000 man-year[18]. Some principal features of this MPS are shown in Fig. 5.1. Characteristic motor data are given in Table 5.1.

Vacuum Total Impulse, (MNs)	1415
Av. Vacuum Thrust, (MN)	11,5
Vacuum Specific Impulse, (s)	286,4 (e)
Burn Time, (s)	123,7
Motor Diameter, (m)	3,71
Motor Length (excl. nose cone), (m)	38,5
Total Motor Mass (excl. nose cone), (ton)	569,9
Approx. Usable Propellant Mass, (ton)	501,7
Thrust Vector Control	8 deg. Pitch/Yaw

Table 5.1: Space Shuttle SRB Characteristics

Each SRB has a total mass of 569,9 ton (excluding the nose cone) and produces an average vacuum thrust of 11,5 MN during about 123 s. Each SRB contains approximately 501,7 ton of propellant, which amounts to about 88,0% of the total mass. It can easily be verified, that the average propellant fuel mass flow is equal to 4093 kg/s, leading to an average vacuum specific impulse of about 286,4 s. By multiplication of the propellant mass, the specific impulse, and the gravitational acceleration, we then find for the total impulse in vacuum a value of about 1415 MNs.

Each motor is segmented for ease of rail transport from the manufacturer to the launch site and consists of four cylindrical segments with a length of 3,20 m, one cylindrical segment 60% of the normal length, plus a forward segment with a domed closure and an aft segment with the rocket nozzle and an aft skirt that incorporates launch pad mounting structures. The motor diameter is 3,71 m and the overall length is 38,5 m (excluding the forward skirt and nose cone). The casing material used is most likely D6AC steel.

The SRB is cast in four segments: one forward segment, two center segments and one aft segment. When insulated, lined and loaded with propellant, and with igniter and nozzle installed, the casting segments are assembled at the launch site by means of tang-and-clevis joints to form a complete SRB. Each joint is fastened with 180 headless steel pins inserted in matching holes.

[18] [Andrews and Haberman, 1991] mention a development cost of about US$ 336 million. Unfortunately, however, it is not clear from their work, whether this value is a historic value or has been corrected for inflation, etc. Hence, this makes their value hard to compare with the value of Koelle.

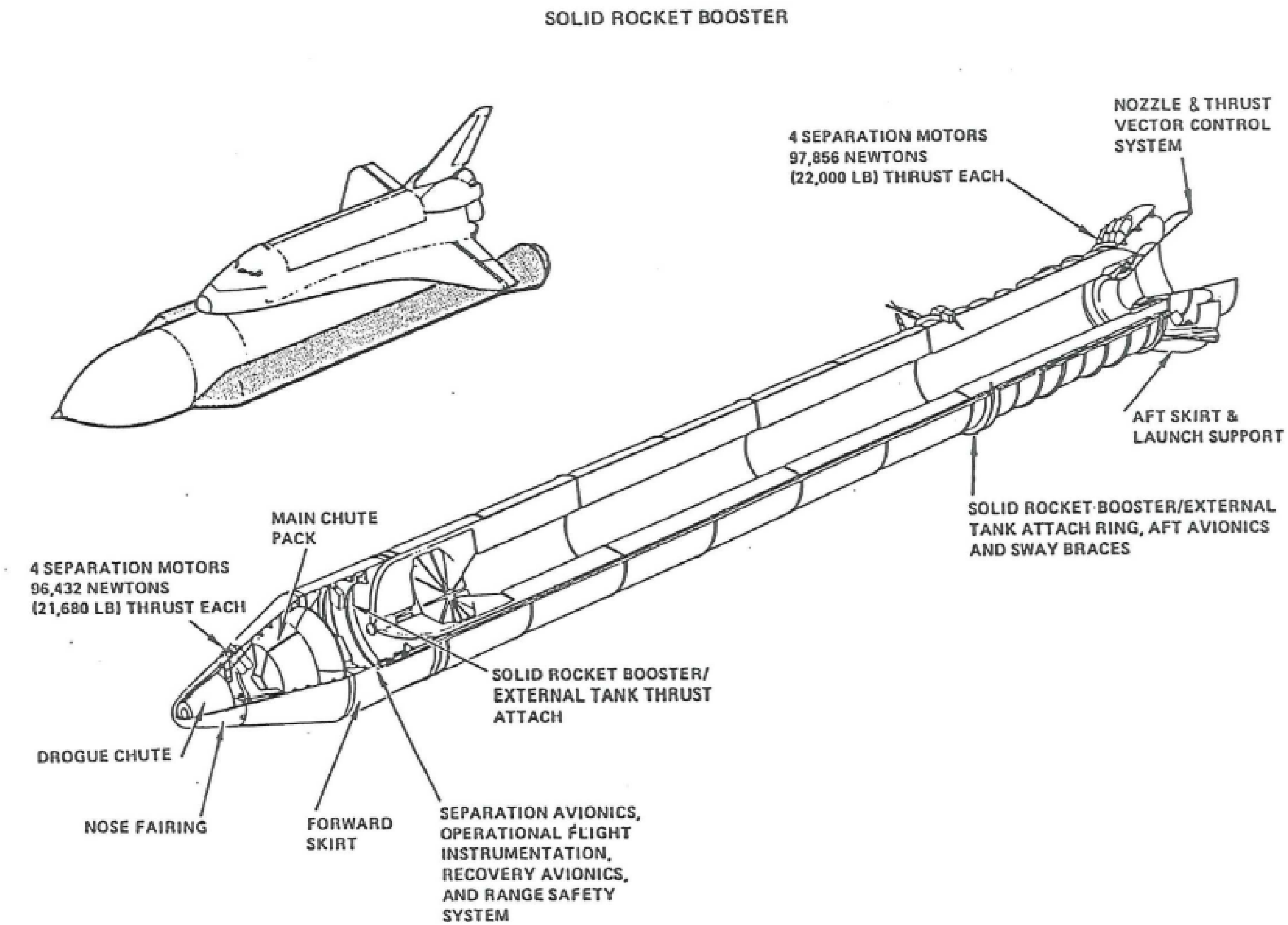


Figure 5.1: Space Shuttle SRB

The propellant mixture in the SRB consists of ammonium perchlorate (oxidizer), aluminium (fuel), iron oxide (a catalyst) and a PBAN binder that holds the mixture together and also acts as a fuel. The total solid load of the propellant is 86%. The propellant grain in the forward section of the forward segment is configured to an 11-point star, which transitions into a tapered, center-perforated cross section at the aft end. The two center segment grains are identical, tapered, center-perforated grains with cast-in-place inhibit on the forward surfaces to control the burning pattern. The aft segment contains a tapered center-perforated grain that widens rather abruptly at the aft end to accommodate the forward part of the nozzle.

The grain is shaped to reduce the thrust of the motor approximately one-third, 55 seconds after lift-off to prevent overstressing of the vehicle during the period of maximum dynamic pressure. When the two strap-ons are jettisoned at an altitude of 50 km, after 123 s of flight, the Shuttle will have achieved a velocity of 5150 km/h. After separation, the strap-ons will begin a re-entry recovery phase. As the strap-on descends, a series of parachutes - first a pilot parachute, then the drogue and finally the main parachute cluster - are deployed to decelerate the motor to a vertical water impact velocity of about 96 km/h. Following splashdown, the motor and parachute are retrieved from the ocean and towed back to the launch site. The rocket motor segments and other components will be treated to limit salt-water corrosion, and returned to the loading point for refurbishment and reuse. Motor case segments and other metal components are designed for 20 flights.

The nozzle is a 20,4 % submerged[19], omni directionally movable, flexible bearing nozzle with a throat diameter of 138,4 cm, an exit plane diameter of 375,9 cm and a total mass of 10,9 ton. The nozzle consists of insulated aluminium and steel components. Ablative cooling is applied; the carbon cloth nozzle liner erodes and chars during the SRB firing.

The motor nozzle is gimballed for thrust vector control. Each motor incorporates its own auxiliary power unit and hydraulic system with hydraulic actuators for gimballing the nozzle.

The kg pyrogen igniter consists of a D6aC steel case, insulated inside and out, containing approximately 80 kg of fast-burning propellant in the shape of a 40-point star. A moulded silica phenolic throat insert directs the flame pattern to the propellant grain. Total igniter mass is 200 kg.

Specific literature

1. Laan, F.H. van der, and Timnat, Y.M.
   Chemical Rocket Propulsion, Lecture Series D-35, Delft University of Technology, Faculty of Aerospace Engineering.

[19] A submerged nozzle is characterized by that the forward part (convergent part and throat) of the nozzle is hidden inside the combustion chamber to reduce the overall length of the vehicle.

6. Advanced segmented SRM (ASRM)

Development of the ASRM as a successor to the Thiokol developed Space Shuttle segmented solid rocket strap-on motor, discussed in the previous section, started as early as 1979. A major redesign effort followed the tragic incident with the Space Shuttle Challenger, which has been attributed to a faulty design of the seals around the motor segment. Since 1989, the further development has been put in the hands of Lockheed Space Co. and Aerojet team at an estimated cost of US$3 billion (1991 figure, which resembles about 16000 man-year). The major purpose of the ASRM-development is to enhance the Space Shuttle's payload delivery capacity with about 6000 kg and to improve the safety of the Space Shuttle system. Some principal features of the ASRM are shown in Fig. 6.1.

Each ASRM has a total mass of 650,9 ton (excluding the nose-cone and front skirt) and produces an average vacuum thrust of 11,7 MN. The motor contains approximately 546,6 ton of propellant, which amounts to about 84,0% of the total mass.

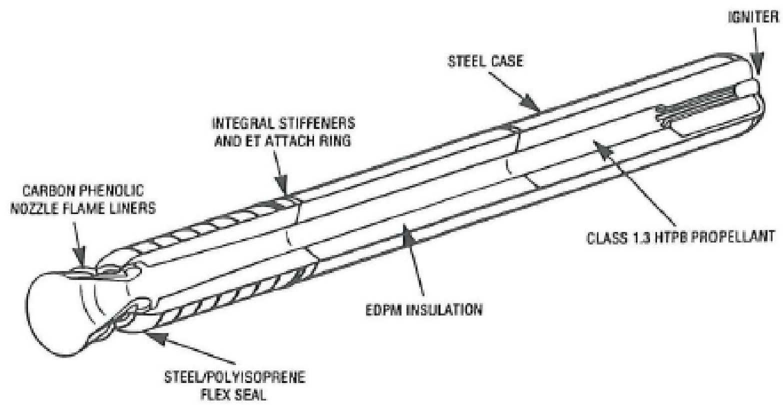


Figure 6.1: ASRM design

The motor is segmented for ease of rail transport from the manufacturer to the launch site and consists of three cylindrical segments with a domed closure at the head end. The most aft segment holds the nozzle and is provided with integral stiffeners and an attachment ring for attachment to the Space Shuttle (External Tank; ET). All segments are made out of 9Ni-4Co-0,3C[20] steel, which is a weldable alloy. The motor diameter is 3,81 m (10 cm wider than the original SRB's, see previous section) and the overall length is 38,4 m (excluding the forward skirt and nose cone).

The ASRM is cast in three segments: one forward segment, one center segment and one aft segment. This is one segment less than for its predecessor, which reduces the mass of the motor with about 4000 kg in inert mass. When insulated, lined and loaded with propellant, and with igniter and nozzle installed, the casting segments are assembled at the launch site using bolts rather than pins like for the tang-and-clevis joint. For the ASRM, a polyamide filled liner, which is free of asbestos, is used for case insulation, thereby reducing the mass of the liner with approximately 900 kg compared to its predecessor.

[20] 9 % Nickel, 4% Colombium, and 0,3% Carbon (mass percentages)

The propellant is a 88% solids (19% aluminium) HTPB-type propellant. The propellant grain in the forward section of the forward segment is configured to a star, which transitions into a more cylindrical perforation at the aft section. The center segment grain is tapered, center-perforated grain with cast-in-place inhibit on the forward surfaces to control the burning pattern. The aft segment contains a tapered center-perforated grain, the core of which widens rather abruptly at the aft end to accommodate the forward part of the partially submerged nozzle.

The nozzle is a flexible bearing nozzle using a steel/polyisoprene flex seal, which allows for thrust vector control.

Specific literature

1. Mitchell, R., Thomas, J., and Levinsky C.
   ASRM: Turning in solid performance, Aerospace America, July 1992.

2. Isbell, D.
   ASRM price tag nears $3 billion, Space news, 8-14 July, 1991.

7. VEGA P80 motor

The VEGA P80 motor is developed under ESA authority. Industrial Prime Contractor is FIAT Aviazione. Principal features of the P80 are shown in Fig. 7.1.

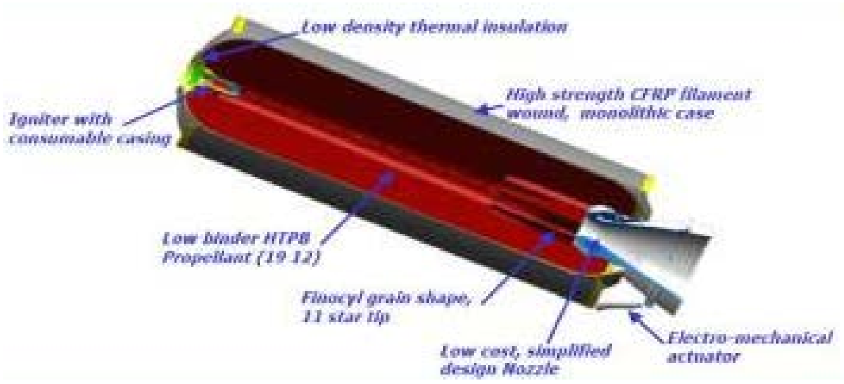


Figure 7.1: P80

The motor is tailored and is directly applicable to the VEGA small launcher. However, the motor is also tailored for validating technologies applicable at a later stage to a new generation of Ariane-5 solid boosters.

The motor case is of a monocoque design and is made of a high strength composite graphite epoxy that is filament wound. The case is thermally protected by a lightweight, low-density type of EPDM rubber.

As propellant is used a HTPB type of propellant with a high aluminium content of 19% (12% HTPB binder and the remainder is ammonium perchlorate. The HTPB-based 1912 propellant grain has a cylindrical perforation that evolves into a 12 star shaped grain close to the nozzle (finocyl shape).

Figure 7.2: P80 Nozzle

The nozzle has a throat diameter of ~500 mm and an expansion ratio of 16. The divergent part is made of carbon phenolic materials. To allow for TVC, the nozzle is equipped with an advanced self-protected flexible joint. Actuation of the nozzle is through electromechanical actuators. Nozzle deflection angle is up to 6.5 degree..

The igniter has a consumable composite case of a carbon fibre structure.

Figure 7.3: P80 Igniter

Vacuum Total Impulse, (MNs)	241.2 (e)
Av. Vacuum Thrust, (kN)	2261
Vacuum Specific Impulse, (s)	279,5
Burn Time, (s)	106,7
Maximum pressure (bar)	95
Motor Diameter, (m)	3,0
Motor Length, (m)	10,557
Skirt to skirt length, (m)	8,63
Inert mass, (ton)	7,408
Usable Propellant Mass, (ton)	88,383
Thrust Vector Control	Yes

Table 7.1: P80 Rocket Motor Characteristics

P80 development cost is estimated at $110 million (SN, Oct. 30, 2000). France, Italy, Belgium and Netherlands together provide the necessary funds. About 50% is funded directly by FIAT Aviazione.

Specific literature

1. L. Battocchio
VEGA - Vettore Europeo di Generazione Avanzata; For the April 15, 2002 issue of SpaceEquity.com

2. R. Barbera & S. Bianchi
VEGA: The European Small launcher Programme; ESA Bulletin 109, February 2002.

8. MAGE series of Single-Case SRM's

The MAGE (Moteur d'Apogée Géostationnaire Européen) motors developed by SEP (France) in close cooperation with BPD (Italy) and MAN (Germany) form Europe's main series of single-case solid propellant apogee kick motors. The series consist of 2 different motors, respectively MAGE 1 and MAGE 2, which mainly differ in length and on board propellant mass. Some principal features of these motors are shown in the Figs. 8.1 and 8.2. Characteristic motor data are given in Table 8.1.

Motor Version	1	1S	2
Vacuum Total Impulse, (MNs)	0,970(e)	1,186(e)	1,412
Av. Vacuum Thrust, (kN)	19,4	23,73	32,3(e)
Vacuum Specific Impulse, (s)	295	295	292,6
Burn Time, (s)	50	50	43,7
Motor Diameter, (m)	0,766	0,766	0,766
Motor Length, (m)	1,125	1,288	1,525
Total Motor Mass, (kg)	369	448	530
Usable Propellant Mass, (kg)	335	410	490
Thrust Vector Control	No	No	No

Table 8.1: MAGE Rocket Motor Characteristics

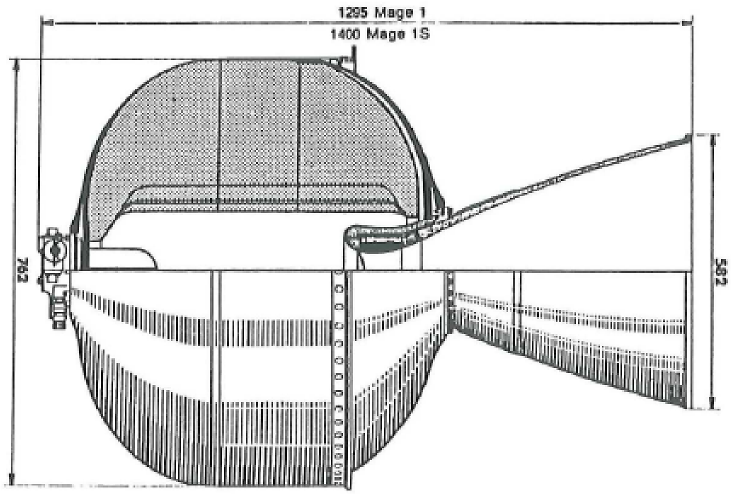


Figure 8.1: MAGE 1/1S SRM

The MAGE 1 rocket motor is available in 2 versions, being 1 and 1S. MAGE 1S is, basically, the same as MAGE 1, except that it is capable of providing a slightly higher total impulse, and is intended for spin-stabilized platforms. The higher total impulse level is accomplished by an increased propellant load, which is made possible by extending the length of the motor case of the MAGE 1 motor from 1,125 m to 1,288 m. The basic MAGE 1 provides an estimated vacuum total impulse of 0,97 MNs compared to an estimated 1,186 MNs for the 1S version. The total mass of MAGE 1 is 530 kg of which 490 kg is propellant (MAGE 1S has a total mass of 448 kg and a propellant mass of 410 kg). This gives a propellant fraction of 92,5% for MAGE 1 compared to 91,5% for MAGE 1S. The burn time for both motors is 50 s. Off-loading of propellant mass is a standard option for MAGE 1/1S.

MAGE 2 provides a vacuum total impulse of 1,41 MNs. The total mass of the motor is 530 kg of which 490 kg is propellant (propellant fraction of 92,5%). Its burn time is 63,7 s, which is longer than for MAGE 1/1S. Based on the vacuum total impulse and the burn time, an average vacuum thrust of about 32,3 kN is estimated. Off-loading of propellant mass by 90 kg is standard option.

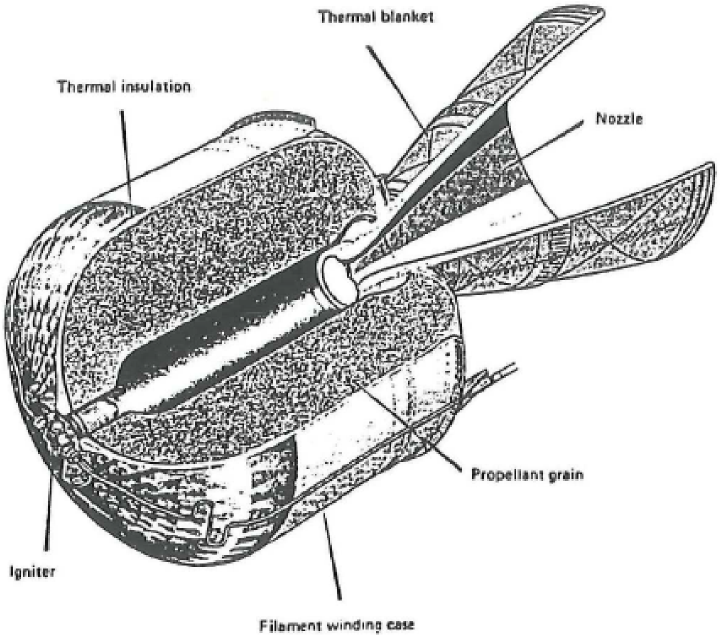


Figure 8.2: MAGE 2 SRM

The MAGE motor cases all use filament wound Kevlar 49, embedded in an epoxy matrix, as the case material. The case is a thin-walled pressure vessel designed to withstand about 71 MPa. Average operating pressure is 35 MPa and the maximum operating pressure is 48 MPa. Some important case dimensions are given in Table 7.2. A mounting flange is present for motor handling and mating to the spacecraft. To provide thermal protection, the internal surfaces of the motor case are insulated with Ethylene Propylene Diene Polymer (EPDM) rubber (MAGE 1/1S) or by asbestos fibre-filled rubber (MAGE 2).

Motor Version	1	1S	2
Mass, (kg)	9	11,4	13,7
Volume, (l)	212	260	309
Motor Diameter, (m)	0,766	0,766	0,766
Motor Length, (m)	0,628	0,728	0,845

Table 8.2: Some MAGE Motor Case Characteristics

The composite propellant uses Carboxy-Terminated PolyBbutadiene (CTPB) as the binder fuel (12%), which is loaded with 16% Aluminium (Al) as additional fuel and 72% Ammonium Perchlorate (AP) as the oxidiser. The propellant grain is of a tubular shape. Only for the MAGE 2, the propellant grain is not bonded to the case. This is to avoid grain stress. Both motors can be off-loaded to some extent.

Ignition occurs using a head-end pyrogen igniter. Initiation is provided by a remote Safe and Arm (S&A) device, which fires a detonation cord, which runs along the exterior of the motor. The detonation cord in turn ignites the Through-Bulkhead Initiators (TBI's), which finally ignite the head-end pyrogen charge.

The MAGE 1 and 2 motors feature a semi-submerged carbon-carbon nozzle with a 4D throat. The nozzle is supported by a titanium adapter, which also provides attachement to the case. Thermal insulation of the nozzle is provided by carbon-phenolic and EPDM. Thermal radiation from the hot nozzle to the environment is minimised using carbon felt on the outside of the nozzle. An aluminium diaphragm at the nozzle entrance protects the loaded case against contamination. The expansion ratio of the MAGE 1 and MAGE 2 nozzle is 45:1 and 65,1:1, respectively and the exit diameter 0,582 m and 0,605 m.

Specific literature

1. An.
   MAN in space (D 184070).

9. STAR 30 and 37 FM series of single-case SRM's

The STAR 30 and 37 FM SRM's belong to Thiokol's (USA) series of Spherical Thiokol Apogee Rocket (STAR) motors[21]. This series of motors has been especially developed for use as spacecraft kick and launcher upper stage motors, and has shown a succes ratio of 0.999 in nearly 2100 flights. Some principal features of the STAR 30 and 37 FM motors are shown in Fig. 9.1. Characteristic motor data are given in Table 9.1.

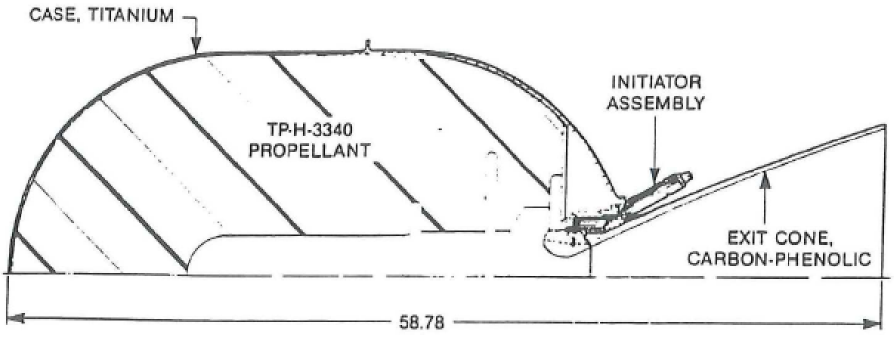
Motor Version	30B	30C	30E	37 FM
Vacuum Total Impulse, (MNs)	1,46	1,65	1,78	3,05
Av. Vacuum Thrust, (kN)	27,04(e)	-	-	47,85(e)
Vacuum Specific Impulse, (s)	293,0	285,2	290,1	291,8
Burn Time, (s)	54	51,1	49,3	63,7
Motor Diameter, (m)	0,762	0,762	0,762	0,933
Motor Length, (m)	1,51	1,49	1,68	1,68
Total Motor Mass, (kg)	537	620	667	1148
Usable Propellant Mass, (kg)	505	585	621	1066
Thrust Vector Control	No	No	No	No

Table 9.1: STAR 30 and 37FM Characteristics

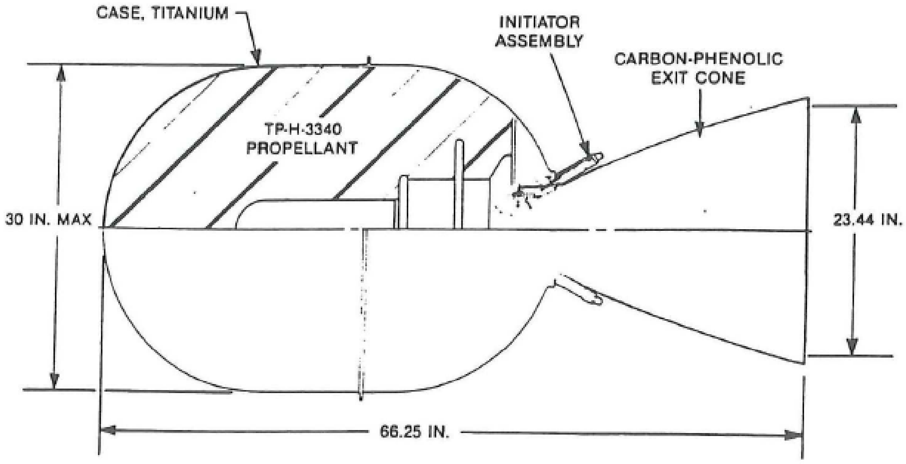
The STAR 30 has three distinct versions (designated by 30B/C/E), where the C and E versions provide an improvement in total impulse over the B version (1,46 MNs) of 13,1% and 21,8% respectively. This improvement is achieved through an increase in propellant load, which in turn is accomplished through stretching of the motor case. The total mass of the STAR 30 B/C/E is 536,8 kg, 620 kg, and 665,6 kg. The propellant mass is 505 kg, 585 kg, and 621,4 kg, which gives for the propellant fraction 94,1%, 94,3% and 93,4%, respectively. A typical mass breakdown of the STAR 30E motor is given in Table 9.2.

The STAR 37 FM is a slightly improved version of the STAR 37 motor. Its total mass is 1149,4 kg of which 1066 kg is propellant (propellant fraction of 92,7%). Its burn time is 63,7 s and the average vacuum total impulse is 3,05 MNs, which gives an estimated average vacuum thrust of about 47,85 kN. Off-loading of propellant mass by 10% is standard option. Table 9.3 gives a typical mass breakdown.

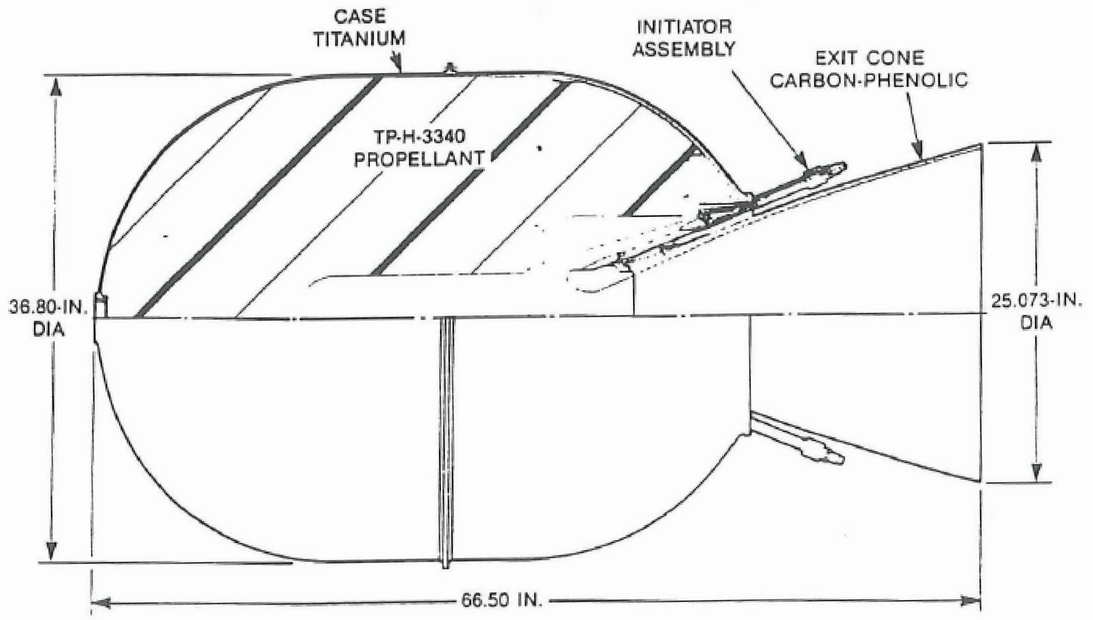
[21] Besides the STAR 30 and 37 FM, the STAR series also includes STAR 6, 27, 31, 48, 63, and 75. STAR motors are available in diameters varying between about 0,15 m (6 inches; STAR 6) to about 1,90 m (75 inches; Star 75) with propellant masses varying from as little as 2 kg to over 10.000 kg (5 lb_m - 20.000 lb_m). TVC has been proven on the STAR 27 and is being tested on STAR 48. STAR 48V features a moveable nozzle, which vectors up to 4 degrees.



STAR 30C rocket motor assembly



STAR 30E (TE-M-700-17) rocket motor assembly



STAR 37FM (TE-M-783-1)

Figure 9.1: STAR 30C/E and Star 37 FM

The STAR 30 and 37 FM motors have been designed from a common approach and use a 6Al-4V titanium case. This case is a thin-walled pressure vessel designed to withstand 1,25 times the Maximum Expected Operating Pressure (MEOP) and is formed from forward and aft hemispheres, which are joined to a cylindrical mid-section. The latter allows for easy adaptation of the length of the motor. Joining of the different parts is through circumferential welding. A mounting flange is present for motor handling and mating to the spacecraft. The internal surfaces of the motor case are insulated with silica-filled EPDM rubber to provide thermal protection to the case.

case (kg)	17,45
case insulation (kg)	9,30
nozzle assembly (kg)	16,98
liner (kg)	0,72
igniter propellant (kg)	0,25
propellant (kg)	621,43
hardware (kg)	0,21
initiator assembly (kg)	0,28
TBI (kg)	0,10
mounting assembly (kg)	0,19
expended inerts (kg)	5,85
total mass (kg)	666,92

Table 9.2: STAR 30E mass characteristics

The propellant consists of Hydroxy-Terminated PolyButadiene (HTPB) as the binder fuel, 18% Aluminium (Al) fuel and Ammonium Perchlorate (AP) as the oxidiser. The STAR 30 and 37 FM propellant grain features a full head end web, radial slots, and cast in place STAR points. The average combustion pressure for the STAR 30C and 37 FM is about 3,69 MPa and 4,14 MPa. The propellant is case bonded using a liner.

Ignition occurs using a toroidal ignition system (Fig. 8.2), which is integral to the titanium nozzle aft closure. Such an ignition system eliminates head end ignition and allows for the incorporation of an head end web, thereby increasing the propellant fraction. The ignition system is initiated from a remotely located safe and arm (S&A) device and is initiated by Through Bulkhead Initiators (TBI). The TBI's ignite a small transfer propellant grain, which in turn ignites the main igniter propellant in the toroidal chamber. The hot gases generated in this chamber exhaust on to the actual propellant grain from 12 ports in the face of the housing.

case (kg)	32,11
case insulation (kg)	14,74
Nozzle and igniter assembly (kg)	33,24
liner (kg)	0,68
Igniter propellant (kg)	0,54
propellant (kg)	1056,73
miscellaneous hardware (kg)	0,73
initiator assembly and TBI (kg)	0,28
expended inerts (kg)	7,35
total mass (kg)	1148,33

Table 9.3: STAR 37FM mass characteristics

The STAR 30 and 37 FM motors, except the STAR 30B, all feature a tape-wrapped carbon phenolic exit cone with a 3D carbon-carbon throat insert (Fig. 9.3). The nozzle is supported by a titanium adapter, which also provides attachment to the titanium aft closure. Thermal insulation of the nozzle is provided by carbon-phenolic and EPDM. The expansion ratio for the STAR 30E and 37 FM is respectively 65,8:1 and 50,9:1 and the exit diameter 0,60 m (23,44 in.) and 0,64 m (25,073 in). No data are available for the STAR 30B and C. All STAR motors are equipped with a slightly submerged nozzle, except the STAR 37FM, which has a semi-submerged nozzle.

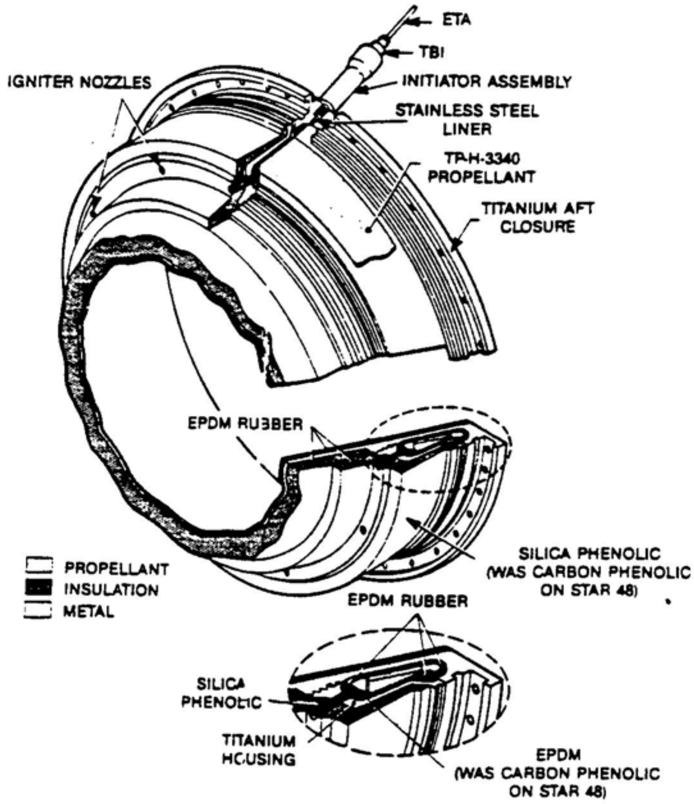


Figure 9.2: Toroidal igniter

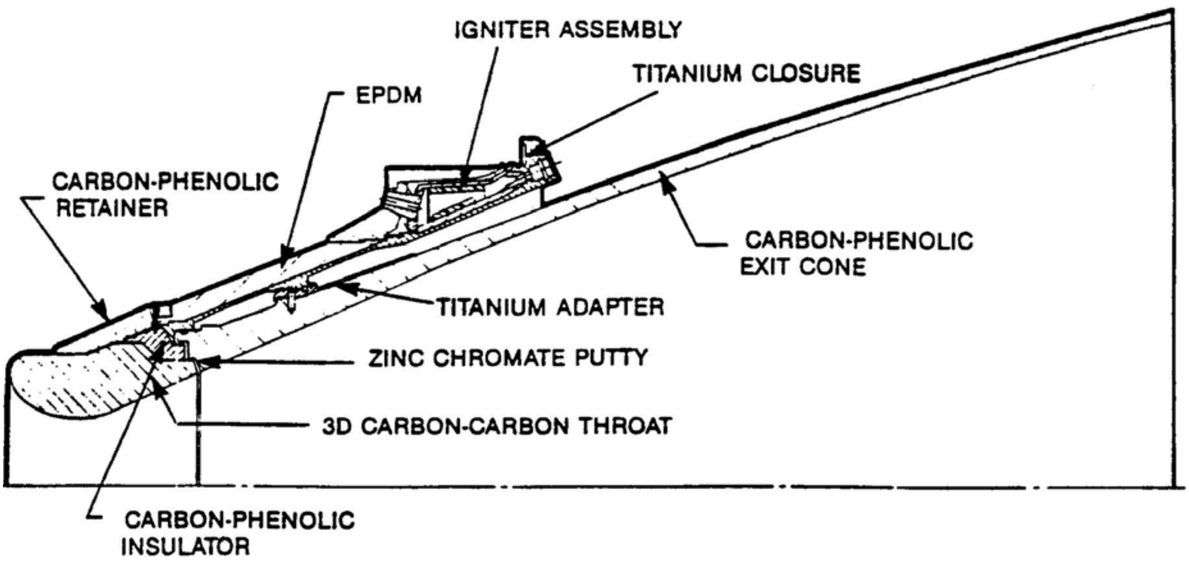


Figure 9.3: Nozzle assembly

Specific literature

1. Walstrum, D.W., and Stransky, M.D.
   STAR 30 C/E and STAR 37 FM improved performance in upper stage solid rockets, AIAA-86-1374.

2. An.
   Thiokol brochure TK3110.

10. Orbus series of single-case SRM's

The Orbus series of single-case SRM's have been derived from the two motors developed for the Space Shuttle Inertial Upper Stage (IUS) programme. These two motors are Orbus 21 and Orbus 6E[22], see Figure 10.1. Characteristic motor data of both motors are given in Table 10.1.

Development of the motors started in 1978 and ended August 1983 after 30 static test firings. Total development cost has been reported to be US$ 32 million (1978 figure), which is approximately equal to 400 man-year.

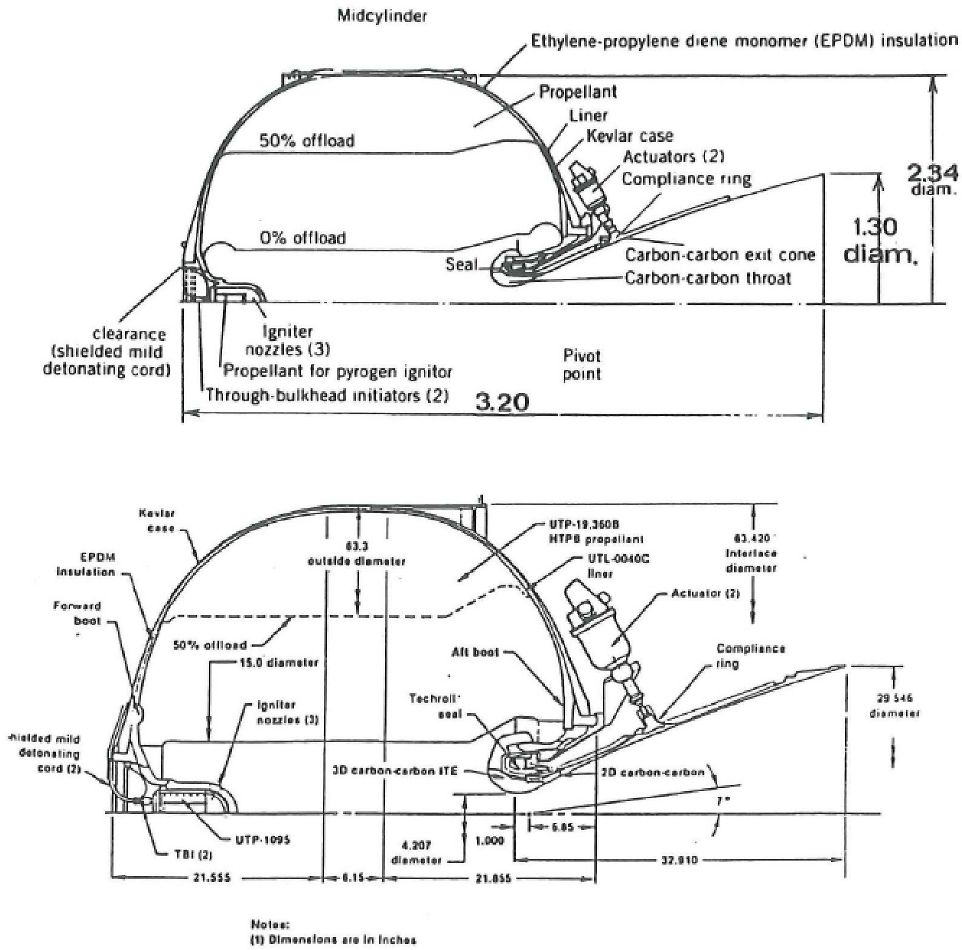


Figure 10.1: Orbus 21 (top) and 6E (bottom)

The 3,2 m long and 2,3 m maximum diameter Orbus 21 has a mass of about 10.398 kg (fully loaded) of which 9700 kg is propellant, which gives a propellant fraction of 93,2%. During operation, Orbus 21 generates an average vacuum thrust of 195,7 kN at a vacuum specific impulse of 295,5 s. The average burn time is 146 s allowing for a total impulse in vacuum to be delivered of 28,1 MNs.

[22] The numerical motor designator corresponds to motor thrust in pounds-force to the nearest 1000 lbf and E denotes the use of an extendable nozzle.

Orbus 6E has a mass of about 3018 kg (fully loaded) of which 2720 kg is propellant (propellant fraction is 90,1%). When operative, Orbus 6E generates an average vacuum thrust of 81,0 kN at a vacuum specific impulse of 303,5 s. The average burn time is 103 s, allowing for a total impulse in vacuum to be delivered of 8,1 MNs. The length of the motor is either 2,0 m or 3,2 m with the nozzle fully extended. Its maximum diameter is 1,6 m.

	Orbus 21	Orbus 6E
Vacuum Total Impulse, (MNs)	28,1	8,1
Av. Vacuum Thrust, (kN)	195,7	81,0
Vacuum Specific Impulse, (s)	295,5	303,5
Burn Time, (s)	146	103
Motor Diameter, (m)	2,3	1,6
Motor Length, (m)	3,2	2,0/3,2
Total Motor Mass, (kg)	10398	3018
Usable Propellant Mass, (kg)	9700	2720
Thrust Vector Control	7 deg. Pitch/yaw	7 deg. Pitch/Yaw

Table 10.1: Orbus Rocket Motor Characteristics

Both rocket motors have been designed from a common approach and use a kevlar-epoxy case. This case is wound over an insulated mandrel utilizing both helical and hoop layers to achieve the desired homogenity in the case wall. The fiber content is between 60 and 65 %. After curing at approx. 150^{o}C the mandrel is removed and the case hydro-tested to pressures up to 9.7 MPa. Nominal Maximum Expected Operating Pressure (MEOP) is 5,71 MPa for Orbus 6E and 5,79 MPa for Orbus 21.

The composite propellant is made up of Hydroxy-Terminated PolyButadiene (HTPB) as the binder fuel and Ammonium Perchlorate (AP) as the oxidiser. The total solids loading of the binder is 86%. Ballistic characteristics such as burn rate, slope and temperature sensitivity are controlled through the selection of AP particle size and the use of burn rate catalysts, if required. Both rocket motors can be 50% off-loaded, affording a wide variety of total impulse.

The nozzle of both motors consists of a three dimensional carbon/carbon integral throat and entrance design and a two dimensional carbon/carbon exit cone. Nozzle control, necessary for thrust vector control is realized by means of the Techroll system. This is a nozzle suspension system consisting essentially of a fluid-filled (silicon oil) bladder, which serves as the joint between the nozzle and its mounting, see Fig. 10.2. It has the advantage of very low actuation torques and is capable of high deflection angles (up to 7^{o}). Actuation of the nozzle is by orthogonally placed electromechanical actuators allowing for both pitch and yaw control.

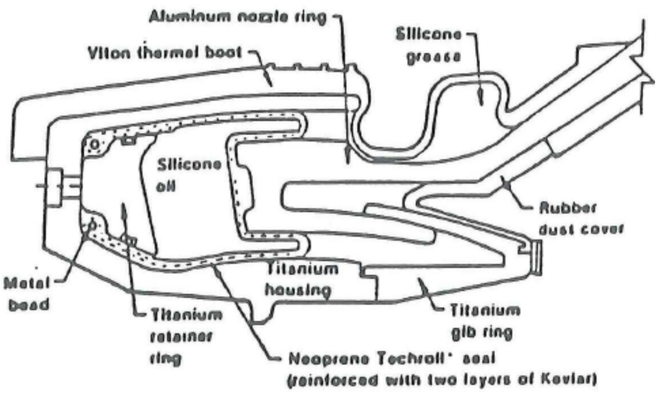


Figure 10.2: Techroll nozzle suspension system

The nozzle of the Orbus 6E can be extended with an Extendible Exit Cone (EEC) to increase the geometric expansion ratio from 47,3:1 to 181,1:1. This EEC is a system of two 61 cm long nested two-dimensional carbon/carbon cones, one with a 1,09 m diameter exit and the other with a 1,44 m diameter exit.

Specific literature

1. Laan, F.H. van der, and Timnat, Y.M.
Chemical Rocket Propulsion, Lecture Series D-35, Delft University of Technology, Faculty of Aerospace Engineering.

11. Ariane 4 stage separation motors

Ariane 4 is equipped with 4 types of stage separation rocket motors (SSRMs). These motors, respectively named type A, B, C and D provide the impulse needed for the separation of the stages of the Ariane 4 launch vehicle. The motors have been developed and flight qualified by FIAT AVIO (Italy). Characteristic motor data of the type A, B, C and D motors are given in Table 11.1.

Qualification of the motors took place in 1978 (types A-C) and 1986 (type D). Typical operation range is from -30 to + 40 °C.

	Type A	Type B	Type C	Type D
Total Impulse, (kNs)	26,6	30,8	28,0	26,6
Average Vacuum Thrust, (kN)	25,0	6,5	3,0	25,0
Delivered Specific Impulse, (s)	242	227	220	242
Burn Time, (s)	1	4,8	9	1
Max. Chamber Pressure (MPa)	14	5,3	4,5	14
Motor Diameter, (m)	0,231	0,231	0,231	0,231
Motor Length, (m)	0,54	0,465	0,467	0,525
Total Motor Mass, (kg)	23,0	23,5	22,6	22,8

Table 11.1: Ariane 4 Stage Separation Motor Characteristics

The 4 types of Ariane 4 SSRMs have a burning time in the range of 1-9 s and a mean vacuum thrust level of 25 kN (type A and D), and 3 - 6,5 kN (type B and C).

All 4 types use a HTPB-type of composite propellant with the grain shaped in such a way to slow down the 1st and 2nd stages (type A and D) just at the end of their combustion and speed up the 2nd and 3rd stages (type B and C) after the separation from the lower stages. The case of all 4 types of rocket motors is made out of steel. As insulation material Kevlar fiber filled EPDM rubber is used. The nozzle is made of either AISI 4130 steel or silica phenolic material. Ignition of the motors is through a pelleted pyrotechnic igniter.

Specific literature

1. An.
   Rocket Motors, for Stages Separation of Ariane 4 Launcher, by FIAT Aviozione, Italy.

12. IMP SRM's

The British Black Arrow and Black Knight rocket launcher both used a range of small special purpose propulsion units capable of providing the type of impulse needed by vehicles for spin up and separation. One class of motors were termed IMP motors, which were developed by the Rocket Propulsion Establishment established in Westcott (currently part of Royal Ordnance PLC.). Characteristic motor data of IMP VI and X used on the British Black Knight and Black Arrow launcher (IMP VI only) are given in Table 12.1[23].

The IMP VI has a burning time of 0,52 s at 293 K and a mean thrust level of 490 N. IMP X has a somewhat shorter burning time (0,34 s), but a larger mean thrust level (5026 N), which leads to a somewhat larger total impulse of about 2000 Ns (compared to 280 Ns).

	IMP VI	IMP X
Total Impulse, (Ns)	280	1957
Av. Thrust, (N)	490	5026
Specific Impulse, (s)	195	230
Burn Time, (s)	0,52	0,34
Motor Diameter, (m)	0,035	0,070
Motor Length, (m)	0,147	0,229
Total Motor Mass, (kg)	0,334	1,360
Usable Propellant Mass, (kg)	0,142	0,839
Thrust Vector Control	No	No

Table 12.1: IMP's special purpose SRM Characteristics

The IMP XV motor, used on the Black Arrow rocket offers about half the total impulse of the IMP VI. It has about half the propellant quantity and half the propellant web thickness.

All IMP motors use a plastic[24] propellant in the form of a radially burning charge. This charge is case-bonded. The case is made out of high strength extruded steel tubing with threads cut at each end to take the head end closure and the nozzle. The igniter uses a magnesium/potassium nitrate pyrotechnic initiated by a suitable fuse-head. The ignition leads through rubber grommets in the nozzle.

[23] It is not clear from the text of the original publications, for what conditions the data provided are valid.

[24] From an article by Maxwell, it follows that this propellant possibly is a mixture of ammonium perchlorate and polyisobetene. Note that at the time of development (1950's-1960's), rubbery propellants were not yet available in Europe.

Specific literature

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2. Harlow, J.
   Black Knight upper stages, Journal of the British Interplanetary Society, Vol. 43, pp. 311-316, 1990.

13. PSLV special purpose SRM's

The Indian Polar Satellite Launch Vehicle (PSLV) employs a number of SRM's to achieve stage separation and propellant settling. These motors are all of nominal 0,200 m diameter and employ propellant in the mass range of 14-38,5 kg. The motors used for separation purposes (marked 'retro') have low burn time, whereas the motor used for propellant settling (marked 'ullage') has a burn time of 5,5 s. All the motors have canted nozzles to ensure that the motor exhaust does not affect the main vehicle structure. Typical canting angles are in the range of 6 to 29 degrees. In Table 13.1 some detailed characteristics of these motors are given. Specific impulse has been estimated by dividing average thrust by the product of propellant mass flow rate and gravity induced acceleration, where the mass flow rate has been determined by dividing the propellant mass by the burn time. The total impulse has been calculated by multiplication of average thrust and burn time.

	Retro-1	Retro-2	ullage-2
Total Impulse, (kNs)	63,05	36,16	84,15
Av. Thrust, (kN)	48,5	22,6	15,3
Specific Impulse, (s)	239,9 (e)	265,3 (e)	222,9 (e)
Burn Time, (s)	1,3	1,6	5,5
Motor Diameter, (m)	0,209	0,209	0,207
Motor Length, (m)	1,414	0,835	1,271
Total Motor Mass, (kg)	-	-	-
Usable Propellant Mass, (kg)	26,8	13,9	38,5
Thrust Vector Control	No	No	No

Table 13.1: PSLV special purpose SRM Characteristics

Specific literature

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3. An.
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4. Koelle, D.E.
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