SPACE TRANSPORTATION SYSTEMS - PROPULSION, STRUCTURES & SUBSYSTEMS

(System related aspects)

DGLR – CEAS 1st European Air and Space Conference

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1.Abstract

A survey of ongoing developments concerning propulsion, structures and subsystems of space transportation systems is presented for consideration and argumentation by the DGLR Working Group on Space Transportation Systems (FAS4.1). As propellants determine to a high degree the properties of propulsion systems, a short survey lists the categories of propulsion systems according to the state of aggregation of the propellants used (liquid-, solid- and hybrid-propulsion), the number of their components (mono- and bipropellants) and the energy release as measured by specific impulse (Isp). Current propellant R&D focuses on solid high energy additives to liquids, which in turn requires stabilization by gelling. Special hydrocarbons and nitrogen compounds are promising candidates for new High Energy Density propellants. Conventional propellants may be greatly improved by introducing solids in the form of frozen liquids: Hydrogen (improved density as slush) and also conventional liquid bipropellants that are frozen to yield cryogenic hybrid (USAF) or solid bipropellants (AI/FhG Germany and SNPE, France). Environmentally benign ("green") propellants are mainly of interest, because of their capability to reduce handling costs. In terms of reducing the environmental load of rocket launches, the present launch rate is tiny compared with other anthropogenic emissions.

A depiction of the state of the art in liquid propulsion shows European Aestus 1 and 2 as pressure and turbine fed storable propellant Ariane upper stage engines, Vulcain 2 and Vinci as gas-generator and expander cycle cryogenic propellant engines. Staged combustion engines exist in USA (the venerable SSME), Russia (RD0120) and Japan (LE-7A). Current developments in liquid propulsion concern LOX/CH4 as new cryogenic bipropellant and full flow staged combustion.

The same depiction for solid propulsion has to consider small motors of orbital stages, where metallic steel and titanium as well as fiber wound cases are being used. Using a performance factor with the dimension energy over force (=length) reveals recent improvements in solid motor design. Where older designs such as the Ariane 4 PAP had values between 2,5 and 4,5kms, the younger AZK Orion motors obtained, in 2004, 7 to 8kms with high strength steel. Most modern design reach 15kms with filament wound cases. The present VEGA development in Europe is an example of monolithic carbon composite case design.

Hybrid propulsion remains an all time favorite of university student experiments but has also seen major advances by the successful suborbital flight of "Space-Ship1" and the introduction of fast-burning paraffin in NASA sponsored research.

A glance at non-rocket or non-chemical high thrust propulsion reveals the absence of any replacement for chemical rockets in the foreseeable future.

The propellant/propulsion part of the article concludes with a tabular, rather exhaustive survey of international rocket systems. All system in use since Jan. 2000 or later were considered. As of June 2007, a total of 467 launch attempts were made by 8 nations (counting Europe as one).

Propulsive structures have fully turned towards mature compact design with a minimum of hollow spaces as they are used with storable (e.g. Ariane 5 EPS upper stage formed by the Gore-panel technique) or cryogenic propellants (Ariane 5 ESC-A). Ariane 5 EPC is also a typical example of a modern load carrying structure. Re-entrs structures need either complete thermal protection, or must be "hot" structures.

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2. Propulsion: Propellants

2.1 State-of-the Art Propellant Overview

The selection of the propellants is one of the most important steps in the design of an engine.

Propellants are categorized into liquid and solid ones according to the state of aggregation in which they are used, i.e. not the one which they assume at ambient temperature. Propellants may thus require cooling in order to keep them in liquid or solid state. Such propellants are said to be cryogenic. In general, rocket propulsion requires two propellant components, an oxidizer and a fuel. Any such pair is called a bipropellant. When the components of a bipropellant are mixed (provided that is feasible) one obtains what is called a (heterogeneous) monopropellant. Some energetic materials, that can decompose without needing an external reagent, are called homogeneous monopropellants.

2.1.1 Fully Cryogenic Bipropellants

In an order of decreasing energy release per unit total mass, fully cryogenic bipropellants yield the highest performance. A widely used measure of this performance is called Isp [sec], the weight specific impulse. Their components are liquefied gases that must be stored at temperatures below the critical point, where their vapour pressure has some sufficiently low value for pressurized liquid storage. Most frequently used is liquid hydrogen (LH2) as the fuel and liquid oxygen (LO2 or LOX) as the oxidizer (Standard Isp = 391sec). LH2 remains a liquid (of very low density: 0,071 kg/l) at -253o C and LOX (1,140kg/l) at -183o C. Methane, also major constituent of liquefied natural gas (NLG), is a cryogenic hydrocarbon. With higher density (0,4645 kg/l at -164oC) than Hydrogen, it delivers an Isp of 310sec with LOX, but has as yet never been used to power space vehicles

2.1.2 Semi Cryogenic Bipropellants

Using LOX with storable propellants yields semi cryogenic bipropellants. A large number of hydrocarbons have been investigated as fuel candidates, e.g., Propane and Kerosene (Isp 300sec). Kerosene, a blend of different hydrocarbons, is a well-known launcher propellant. The high density (0,81715kg/l) allows a compact design of launcher stages. At higher temperature, kerosene is sensitive to coking and sooting. The coking limits of methane are higher than those of Kerosene.

2.1.3 Storable Liquid Propellants

Liquids of acceptable vapour pressure at room temperature are called storable. Widely used storable bipropellants are based on Nitrogen Tetroxyde N2O (NTO) with Hydrazin-derivatives, e.g. MMH/NTO (Isp 288sec) or UDMH/NTO (Isp 286sec). Engines utilizing storable propellants stand ready for launch on short notice (e.g. in ICBMs). However, storables are also used on longer duration flights such as for the US Space Shuttle or the European Ariane 5 upper stage engine AESTUS. An advantage of these bipropellants is their hypergolicity, i.e. the capability of the components to self ignite on contact, omitting the need for an ignition mechanism. A major diadvantage is that they are poisonous and corrosive, hence storage requires special containers and safety facilities.

Compared with cryogenic propellants, hypergolics are also less energetic. That is, their Isp, as shown above, is lower than the one of semi-cryogenics.

All propellant examples discussed so far were liquids. All propulsion systems using liquids as propellants need at least one each of all of the following subsystems: storage in tanks, a feed system based on pressurization or pumps, pipes, valves and a combustion chamber, that again requires a number of essential components, most notably a heat protection system. All liquid propellants resistant against heat decomposition can be used as cooling fluids ("regenerative cooling").

2.1.4 Storable Solid Propellants

Without a liquid, solid rocket motors rely on interior insulation and high temperature resistant materials for heat protection. There is no feed system. The propellant mixture resides inside the combustion chamber. Ammonium-Perchlorate (NH4ClO4 or AP) is the most widely used solid oxidizer and by far the best. A polymer binder (HTPB) serves as fuel, Aluminum powder is added for energy and density enhancement. A typical mixture contains AP, Binder and Metal in a mass ratio of about 68/17/15 and burns as a heterogeneous monopropellant. The geometrical shape of the burning surface and its evolution during combustion determines the thrust-time history. Isp is low, typically around 260sec.

2.1.5 Hybrids

Hybrids are bipropellants with one liquid and one solid component. The liquid is stored in a tank, as in liquid propulsion, the solid resides inside the combustion chamber, as in solid propulsion. After liquid injection and ignition, combustion takes place in a boundary layer. There is no mixing and consequently a very low risk of explosions. Hybrids with LOX and a solid polymer fuel (P) are of course semi-cryogenic. Storable are H2O2/P and N2O/P, "Green Propellants" where P is either Polyethylene or HTPB.

2.2 Current propellant R&D

2.2.1 Solid propellant research

Amoniumperchlorate (AP) based composite solid propellants are part of a very mature area of propellant technology. Nevertheless there are at least two new developments worth mentioning. Nano-sized particles (<100 nm diameter) of AP or metal-additives posses very favorable combustion characteristics in terms of fast and complete combustion (also applicable for hybrid fuel grains). The second trend concerns ammonium dinitramide (ADN) as "green" high energy solid oxidizer. High energy fuel research has seen progress with encapsulated metal hydrides (e.g. hexahydridoborane). Smokeless or reduced smoke solid propellants are based on what is called double base formulations, i.e. various mixtures of nitrocellulose and nitro-glycerol. They are of great interest for military applications, including gun powder and hypervelocity missiles.

2.2.2 Gelled propellants and energetic matter

Are subject of current R&D because they promise to increase the shelf life (US Army target 2001: up to 20 years) of liquid propellant missiles with solid and/or meta-stable additives including nano-particles (US Army RDT&E 2003). A storable high density gelled fuel STS-deployable upper stage was studied in 2003 by Northrop Grumman for NASA MSFC. Typical gelling additives are mixtures of silicone-polyether block copolymers and free polyethers. An example of near term high energy hydrocarbon fuels is Bicyclopropyliden (C3H2=C3H2, Isp with LOX: 313 s). This research is done by USAF, where pilot plant size has been achieved in 1999.

2.2.3 Reduced Toxicity Monopropellants

Meant for use as high performance storable satellite auxiliary propulsion, they are also subject of USAF sponsord investigations. An example is Hydroxyethylhydrazinium-nitrate (C2H5ON4H4NO3), that is 50% denser, has 25% higher Isp and much lower toxicity than Hydrazine.

2.2.4 High Energy Density Matter

USAF has investigated HEDM since the 1990ies. Near term candidates are the Polynitrogens, of which the N5+ Pentazole Kation has been synthesized at the USAF Edwards AFB Lab in 1999 and by ERC,Inc. HEDM R&D is also the main subject of ERC incorporated, a Huntsville, Ala. based US small business that has been growing fast, working for Argonne National Lab since 1988 (to 80million US$ revenue in 2006). Work includes Polynitrogens and Polyoxygens as well as new non-toxic hypergolic fuels (see also above).

2.2.5 Slush and doped Hydrogen

As solid H2 is 25% denser than LH2. Hydrogen in both forms has been considered for cryogenic stabilization of HEDM. The USAF has trapped Boron atoms in sH2. Slush hydrogen was a requirement for concepts such as the US National Aerospace Plane NASP. The technology development was done by Sierra Lobo (Freemont, Ohio) at NASA's Plum Brook Station.

2.2.6 Cryogenic Solid Propellants CSP

Since 2000, CSP has been the subject of DLR and ESA sponsored R&D by AI:Aerospace Institute, a Berlin based German small business. Work focused on solid propellants using hydrogen peroxyde in innovative solid propellant grains (frozen disk stacks and sponges). Test firings were done by the Fraunhofer Institute for Chemical Technology in Pfinztal, Germany. Similar work was begun at SNPE, France, in 2005.

2.3 Propellants: Environmental Concerns

Concerns about the environmental impact of rocket propellants come in two categories. The first refers to the properties of the propellants themselves, which might be toxic, corrosive and explosive as opposed to "green" and safe. The quest for "green" propellants is driven by the desire of reducing the cost of handling and operations. The second category relates to the combustion products and the products of their interaction with the atmosphere. Typical products of liquid and CSP propellants are H2O, CO, CO2 and NOx. Conventional solid propellants release also HCL and metal-oxides (Al2O3). Any of these compounds and their products of fragmentation by UV radiation is detrimental to the natural stratospheric ozone layer that peaks around 35km altitude. Every space launch causes a temporary hole in the ozone layer, that may have several tenths of kilometers in diameter and last for up to 24h. However, while space travel is the only source of pollution above airline traffic levels, its bulk amount is negligible in comparison to other sources. Assuming a present rate of 100 launches per year, the maximum propellant consumption per year is roughly 100000 tons. In 2004, global air traffic consumed 14,9 billion gallons (10.7 million tons) of jet fuel, which were combusted into 47,4 million tons of H2O and CO2, yielding a factor of about 500 from space- to air-traffic (air traffic in turn was topped by highway by a factor of 11). Comparing anthropogenic emissions with natural ones yields surprises. While annual coal combustion releases about 200 times the amount of HCl released by solid rockets (1,8Mt compared with about 0,01Mt), volcanoes emit 7,8MT every year and the oceans 300Mt.

3. Propulsion: Technologies

3.1 Liquid propulsion

3.1.1 General Principles

Liquid-propellant systems carry the propellant in external tanks. Most of these engines use a liquid oxidizer and a liquid fuel which are stored in separate tanks, and are fed to a combustion chamber where they are mixed, combusted and expanded through a nozzle to produce thrust.

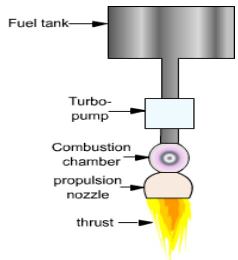


Fig. 1: Basic principle of liquid propulsion.

The propellant supply system raise the pressure well above the operating pressure of the engine, and the propellants are then injected into the engine in a manner that assures atomization and rapid mixing. Liquid-propellant engines have certain features that make them preferable to solid systems in many applications, e.g. (1) higher attainable exhaust velocities or Isp, (2) higher mass fractions (propellant mass divided by mass of inert components), and (3) control of operating level in flight (throttleability), sometimes including stop-and-restart capability and emergency shutdown. In some applications it is an advantage that propellant loading is delayed until shortly before launch time, a measure that the use of a liquid propellant allows.

3.1.2 Performance Comparison

The table compares the vacuum specific impulse (Isp, vac) of some rocket propellants.

Vacuum expansion, Pc=69 bar		
Propellant	Isp,vac (sec)	Mixture ratio O/F
Lox/H2	455	4,8
Lox/Kerosene	358	2,8
Lox/CH4	369	3,5
N2O4/MMH	341	2,4

Tab. 1: Performance and mixture ratio comparison

3.1.3 Propellant Tank, Feed System and Rocket Motor

Liquid propulsion engines use liquid propellants that are fed under pressure from tanks into a thrust chamber (assembly of the injector, chamber, and nozzle extension). The liquid bipropellant consists of a liquid oxidizer (e.g. oxygen) and a liquid fuel (e.g. hydrogen). A monopropellant consists of a single liquid that contains both oxidizing and fuel species. They decompose into hot gas when properly catalyzed.

Liquid rocket engines are either pressure-fed or pump-fed, depending on the mission requirements. Turbopumps are typically used in applications with larger amounts of propellants and higher thrust (e.g. launch vehicles), whereas gas pressure-fed systems are mostly used on low-thrust engines (e.g. satellites).

The propellants are injected, mixed, ignited and burned in the thrust chamber. The very hot combustion gases are then accelerated and ejected at a high velocity through the supersonic nozzle. Thrust is produced equal to the product of mass flow times exhaust velocity and momentum is transferred to the vehicle.

3.1.4 Engine Cycles

An engine cycle describes the propellant flowpaths through the major components of the engine. It is the method how to provide hot gas to one or more turbines, and how to discharge the turbine exhaust gas.
Liquid bipropellant rocket engines can be categorized into: the gas generator cycle, the staged combustion cycle and the expander cycle. The engine cycle terminology refers to the source of energy to drive the turbine.

3.1.4.1 Pressure-Fed Cycle

Pressure-fed cycle is the simplest and least expensive engine design. Without pumps it relies on tank pressure for to feeding the propellants into the main chamber. This cycle is limited to relatively low chamber pressures in order to avoid all too heavy vehicle tanks. The cycle can be reliable, given its lower part count and lower complexity compared with other systems.

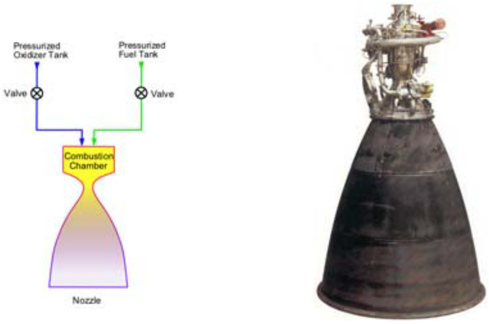


Fig. 2: Left: Basic principle of a pressure-fed cycle. Right: Ariane 5 upper stage engine Aestus built by EADS Astrium at Ottobrunn.

The Aestus rocket engine was developed at Ottobrunn during the period 1988 - 1995. The pressure-fed engine was used on the upper stage of Ariane 5. It is capable of multiple re-ignitions and a highly reproducible restart - necessary for the precision injection of Ariane's multiple payloads. The first operational flight of Aestus was on 30 October 1997 on Ariane 5 flight 502. Aestus features a novel, highly efficient coaxial injector element. By varying the number of injector elements, the engine thrust level, and size, can be adapted to precisely match specific customer requirements.
A more powerful turbopump version of Aestus has also been developed, known as the RS 72 or Aestus 2.

3.1.4.2 Gasgenerator Cycle

The gasgenerator engine cycle has been widely used since early in the history of liquid-fuelled rocket developments. In a gas generator cycle engine, also called open cycle, the turbine flow is in parallel with the thrust chamber. The hot gas of the turbine is either dumped overboard or fed into the main nozzle downstream. Sufficient propellants to drive the turbine are removed from the pump discharge, combusted in the gasgenerator, and expanded through the turbine to atmospheric pressure.

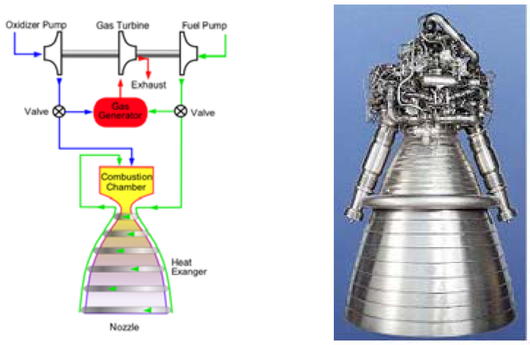


Fig. 3: Left: Basic principle of a gas generator cycle. Right: Ariane 5 Vulcain 2 core stage engine.

The gas generator must burn propellants at a less-than-optimal mixture ratio to keep the temperature low for the turbine blades. Thus, the cycle is appropriate for moderate power requirements but not high-power systems, which would have to divert a large portion of the main flow to the less efficient gas-generator flow. Vulcain 2 is the generator cycle rocket engine for the Ariane 5 core stage, featuring design enhancements and innovative production technologies. The thrust was increased up to 135 tones - more than 30% over its predecessor. Vulcain 2 has sunstantially increased the payload capacity of Ariane 5. Development of the Vulcain family started in 1988 under the leadership of Snecma Moteurs.
A further European gasgenerator engine is the cryogenic upper stage engine HM7, that was already used on the Ariane 4 launcher. EADS Astrium Space Transportation at Ottobrunn facility was responsible for the thrust chamber assembly.

3.1.4.3 Expander Cycle

In this cycle, the turbine flow is also in series with the thrust chamber and the fuel is heated before combustion. However, the turbine available energy is limited only by the fuel flow preheated in the thrust chamber coolant passages instead burned with liquid oxygen in the preburner. It maximizes the engine's specific impulse for a given chamber pressure by passing all the propellants through the thrust chamber, i.e., no flow is dumped.

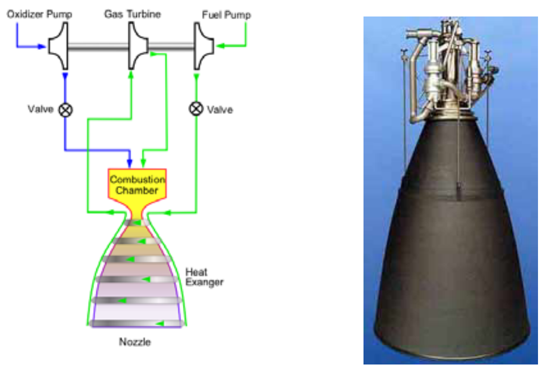


Fig. 4: Left: Basic principle of an expander cycle. Right: Ariane 5 upper stage engine Vinci

Turbine available energy limits the expander cycle engine to relatively low combustion- and moderate pump dis-

charge pressures, low turbine operating temperatures. In practice, this engine type is often used to power upper stages.
In an open cycle, or "bleed" expander cycle, only some of the fuel is heated to drive the turbines, which is then vented to atmosphere to increase turbine efficiency. While this increases power output, the dumped fuel leads to decreased efficiency. This can achieve higher chamber pressures than the closed expander cycle although at lower efficiency because of the overboard flow.
Vinci is an advanced expander cycle cryogenic propellant rocket engine currently under development. It will be the first European re-ignitable cryogenic upper-stage (designated ESC-B - Etage Supérieur Cryotechnique B) engine for the Ariane 5 launcher. On 20 May 2005, the Vinci engine performed its first flawless ignition and hot-fire test at Lampoldshausen's P4.1 test stand.
The biggest improvement from Vinci's predecessor, the HM-7B (which powers the ESC-A upper stage), is the capability of restarting up to five times. It is also the first European expander cycle engine. It features a carbon ceramic extendable nozzle in order to have a large, 2.15 m diameter nozzle extension with minimum length: the retracted nozzle part is deployed only after the upper stage separates from the rest of the rocket.
Further flight proven e. c. engines are: RL-10 (Pratt&Whitney Rocketdyne, USA), LE-5 (MHI, Japan).

3.1.4.4 Staged Combustion Cycle

In the staged combustion cycle engine, the turbine flow is in series with the thrust chamber. In a this cycle, also called a closed cycle, the propellants are burned in stages. A preburner generates the gas for a turbine by tapping off and burning a small amount of one propellant and a large amount of the other, producing a very oxidizer- or fuel-rich hot gas mixture. This hot gas is then passed through the turbine, injected into the main chamber, and burned again with the remaining propellants. All the propellants are burned at optimal mixture ratio in the main chamber and no flow is dumped overboard. The staged combustion cycle is often used for high-power applications.
It maximizes the pump discharge pressures, minimizes the pump flow rates and maximizes the turbine operating temperature. The higher the chamber pressure, the smaller and lighter the engine can be to produce the same thrust.
Development cost for this cycle is higher because the high pressures complicate the development process. Further disadvantages are harsh turbine conditions, high temperature piping required to carry hot gases and a very complicated feedback and control design.
Staged combustion cycle engines yield a performance higher by about 5 to 7% than a gas-generator cycle engine of the same chamber pressure and nozzle area ratio.
Staged combustion was developed independently by Germany and Soviet engineers at the beginning of the 60th.

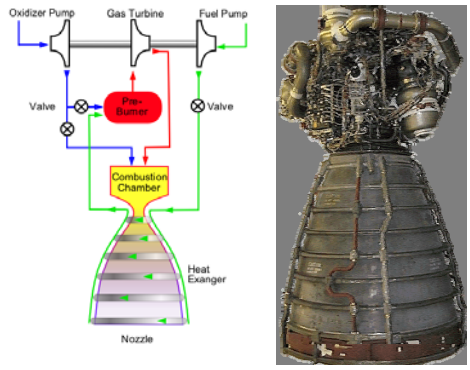


**Fig. 5: left: Basic principle of a staged combustion cycle.
Right: Space Shuttle Main Engine (SSME)**

The Space Shuttle Main Engines was the only high-pressure closed-cycle reusable cryogenic rocket engine ever flown. Three engines are used in a Space Shuttle orbiter, and they provide the main portion of the total impulse to the flight. The SSME is a reusable, staged-combustion cycle engine, and can be throttled between 67% and what NASA calls "109%" of its rated thrust.
Using a staged combustion cycle, the propellants are partially burned at high pressure and relatively low temperature in two preburners, and then completely burned at high pressure and high temperature in the main combustion chamber.
All three main engines receive the same throttle command at the same time. Normally, these come automatically from the orbiter general-purpose computers (GPCs) through the engine controllers. SSME throttling reduces vehicle loads during maximum aerodynamic pressure and limits vehicle acceleration to a maximum of 3 g's during ascent.
Further cryogenic staged combustion engines are: RD-0120 (KBKhA, Russia), LE-7A (MHI, Japan). Popular Lox/Kerosene staged combustion engines are: RD-180, RD-170, RD-191 from NPO Energomash, Russia.

3.1.4.5 Full Flow Staged Combustion Cycle

The so-called full flow staged combustion cycle (FFSCC) is a variation on the staged combustion cycle where all of the fuel and all of the oxidizer pass through their respective power turbines. A small amount of fuel and oxidizer is swapped and combusted to supply power for the turbines.
The turbines run cooler in this design since more mass passes through them, leading to a longer engine life and higher reliability. The design can provide higher chamber pressures and therefore greater efficiency.
This cycle is currently under development in the "Integrated Powerhead Demonstrator Program". A cooperation between NASA, US Air Force, and AFRL with Pratt&Whitney Rocketdyne and Aerojet as industry partners.

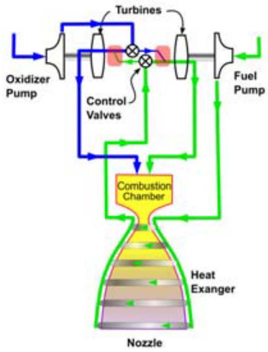


Fig. 6: Basic principle of a FFSCC.

1.4.6 Engine Data Comparison

Tables 2a/2b show selected main data of various flight engines:

Engine	Country	Cycle	Stage	Propellant
Aestus	Europe	PF	US	N2O4/MMH
HM7-B	Europe	GG	US	Lox/H2
Vulcain 1	Europe	GG	CS	Lox/H2
Vulcain 2	Europe	GG	CS	Lox/H2
RS68	USA	GG	CS	Lox/H2
RS27	USA	GG	CS	Lox/Kerosene
Vinci	Europe	EC	US	Lox/H2
RL10 B-2	USA	EC	US	Lox/H2
LE 5B	Japan	EC/B	US	Lox/H2
SSME, IIA	USA	SC	CS	Lox/H2
LE 7A	Japan	SC	CS	Lox/H2
RD-0120	Russia	SC	CS	Lox/H2
RD-180	Russia	SC	CS	Lox/Kerosene
RD-170	Russia	SC	CS	Lox/Kerosene

Tab. 2a: Some characteristic data of flight engines

Engine	Thrust, vac(kN)	Isp, vac (Ns/kg)	Pc (bar)	Mass (kg)
Aestus	29,5	3.170	11	111
HM7-B	62,3	4.370	35,5	155
Vulcain 1	1.145	4.228	109	1.665
Vulcain 2	1.355	4.254	116,4	1.846
RS68	3.317	4.021	97,2	6.800
RS27	1.055	2.962	48,3	1.148
Vinci	180	4.562	61	480
RL10 B-2	110,2	4.575	44,4	290
LE 5B	141,7	4,385	35,7	269
SSME, IIA	2.284	4.430	205,5	3.396
LE 7A	1.080	4.325	119	1.800
RD-0120	1.957	4.462	218	3.450
RD-180	4.156	3.313	256,7	5.545
RD-170	7.903	3.315	245	9.750

Tab. 2b: Some characteristic data of flight engines

3.1.5 Current developments and R&D

3.1.5.1 NASA´s Next Generation Launchers

Currently, it seems that propulsion systems for the so-called next generation launchers are based upon reliable, conservative than real brand new or even innovative technologies. A first step has been done by NASA with the new launcher family Ares I (crew launch vehicle) and Ares V (heavy lift cargo launch vehicle). They are based upon already existing technologies or stages (so-called building block concept). It is, on the first view, a step back and not ahead. However, the devil is in the details and not in the principles of the launcher architecture.

The Ares I second, or upper, stage is propelled by a J-2X main engine fuelled with liquid oxygen and liquid hydrogen. The J-2X is an evolved variation of the well-known J-2 engine that propelled the second and third stage of the Apollo Saturn V launcher.

The Ares V launcher uses, in addition to the two solid boosters, five RS-68 engines for powering the core stage. The RS-68, as further building block engine was developed at Pratt&Whitney Rocketdyne (PWR) to power the Delta IV Evolved Expendable Launch Vehicle (EELV). It was the first new US booster engine developed in over 25 years after the SSME. The second stage, the so-called EDS (earth departure stage) would feature again a single J-2X rocket engine.

3.1.5.2 Lox/LNG

The upper stage of the Japanese Galaxy Express launcher shall be powered by a completely new rocket engine. The stage would be the world's first LNG (or Lox/Methane) orbital rocket stage. Lox/Methane is often in discussion as launcher propellant, but to date no worldwide launcher is using that propellant combination.

By late 2006, JAXA and contractor Ishikawajima-Harima Heavy Industries (IHI) had encountered enough problems with second stage development to substantially delay the program. Costs for developing the engine alone had grown to about $300 million, 3.5 times the initial estimate, which meant that the second stage project cost had grown to something like $600 million. Problems with the CFRP tanks had finally forced a switch to stainless steel tanks. The planned pressure fed engine design was also being reconsidered.

3.1.5.3 Falcon Launcher

The El Segundo, Cal. based SpaceX company follows a commercial approach with its Falcon launcher family. n June, 2002, dot-com multimillionaire Elon Musk established SpaceX Corporation. He poured nearly $100 million of his own money into the company to develop not only the Falcon 1 space launch vehicle and its engines. It is the first time after the Apollo era that an American company has developed a Lox/Kerosene flight engine named Merlin. It is a 40 t gasgenerator cycle booster engine. The upper stage is powered by the new Lox/Kerosene pressure-fed engine Kestrel.
Engine upgrades are in development for Falcon 5 and 9.

3.1.5.4 Air Breathing Propulsion

Several programs are underway to explore revolutionary airbreathing propulsion systems in response to the challenge of reducing the cost of space transportation. Concepts being investigated include rocket-based combined

cycle (RBCC) and turbine-based combined cycle (TBCC) engines.
The primary objective of NASA´s GTX program is to determine whether or not air-breathing propulsion can enable reusable single-stage-to-orbit (SSTO) operations. The approach is based on maturation of a reference vehicle design with focus on the integration and flight-weight construction of its air-breathing rocket-based combined-cycle (RBCC) propulsion system.
In Europe, some basic work is done in the program LAPCAT which is funded from the EU´s 6th framework program. A 36 month study that will focus on propulsion systems that could might make such long distance hypersonic flight practical.

3.1.5.5 Advanced Propulsion

Conventional rockets and fuel simply aren't practical to explore space, especially deep space. Only some priority concepts are presented below.
Solar sail propulsion is a way of moving around in space by allowing sunlight to push a spacecraft. NASA has a program in place to develop solar sail technology to a point where it can be used to implement important space exploration missions.
The source of the electrical energy for electric propulsion is independent of the propellant itself and may be solar or nuclear. Nuclear propulsion is another, more practical method that uses propellant, such as hydrogen, heated to extreme temperatures and ejected at high velocities as in a conventional rocket. Unlike a conventional rocket, the propellant's energy would come from direct or indirect nuclear energy, and thus be extremely more powerful. Nuclear electric propulsion is similar to nuclear thermal propulsion, except that instead of heating the fuel to accelerate it, it ionizes the fuel and then sends it through an electric field to propel it at extremely high velocities.
NASA announced in January 2003 the Prometheus program to develop nuclear propulsion and nuclear power generating systems that could ultimately be used to take a manned mission to Mars and to power other ambitious deep space vehicles.

3.2. Solid Propulsion

3.2.1 Design of Solid Propellant Rocket Motors

Solid propellant rocket motors (short: SRM) have been developed in parallel to liquid propellant motors, as they offer advantages for specific applications: storable over long periods without the need of any servicing, no need for movable parts to fulfil their function, and high trust due to high mass output are regarded as their main highlights.
The following chapters summarize some main items of the different design lines of SRM, whereas the sequence of the chapters reflects the history of development, too.

3.2.2 In Orbit Motors

Developed as the first line, the physical size of these kind of motors ranges from several inch diameter casings (for position adjustment) up to almost 2 m diameter for apogee motors used with satellite platforms.

ATK – Star, Ø 17 inch

MT-Aerospace – MAGE/ IRIS, Ø 760 – 1300 mm

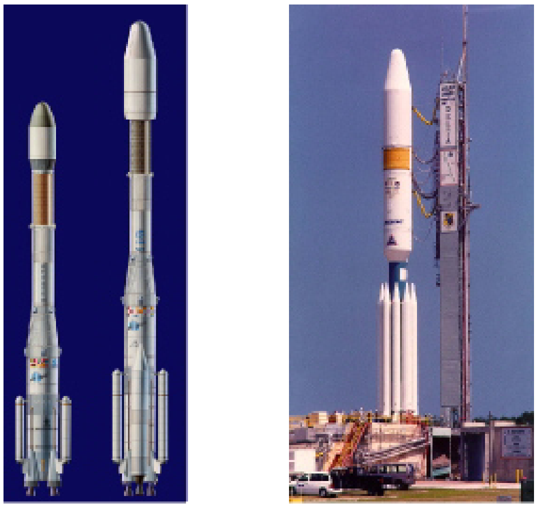
Fig. 7: Solid Propellant In-Orbit Motors

Metallic casing have been made from high-strength steel or Titanium, composite casings used Carbon or Aramide fibres. Common for all this kind of motors are:
- monolithic case, whereas
  - ° composite motors were filament wound over a “lost mandrel”;
  - ° metallic cases are composed by welding of pre-formed elements
- strength-oriented design
- pressure level up to 6 MPa
- loads transmitted by attached skirts

For “lost” mandrels mainly plaster has been used because of its low costs and its ability to be machined. Today, a couple of different chemical mixtures are available which offer easier removal of the material. As usually the openings have been small and the amount of identical casings was low, a dismountable mandrel would have been too expensive.
Today, the majority of in-orbit motors are designed as liquid propellant platforms.

3.2.3 Small In-Line Motors / Strap-on Boosters

The next generation of SRM has been designed by far larger, with higher thrust, thus serving either as a central stage for a small rocket, or as an additional strap-on booster assisting the first phase of the ascent of a larger launch vehicle. This kind of motors have been used up today, for example on the European launch families Ariane 3 and 4, Boeing’s Delta family, Lockheed-Martins Atlas I to V line, Japanese H II-A, and India’s PSLV.
The size of these motors ranges from Ø 1,0 to 1,8 m, the length can be up to 15 meters for some applications.
During the course of their use, these motor types have seen continuous improvements: Early designs have been made from common steel alloy plates, rolled and longitudinal welded to cylinders, the domes welded on both sides, thus forming a monolithic case.
Due to this concept they have been relatively heavy; the achieved performance factors ((pressure x volume) / (mass x gravity const.)) were in the range of 2.5 to 4.5 km.
Examples are: ATK’s Castor-motors and Ariane 4 PAP.
Improved designs, like ATK’s Orion motors (2004), are using high-strength steel alloys, but keeping the same manufacturing principle. By that, the performance factor is in the range of 7 to 8 km.



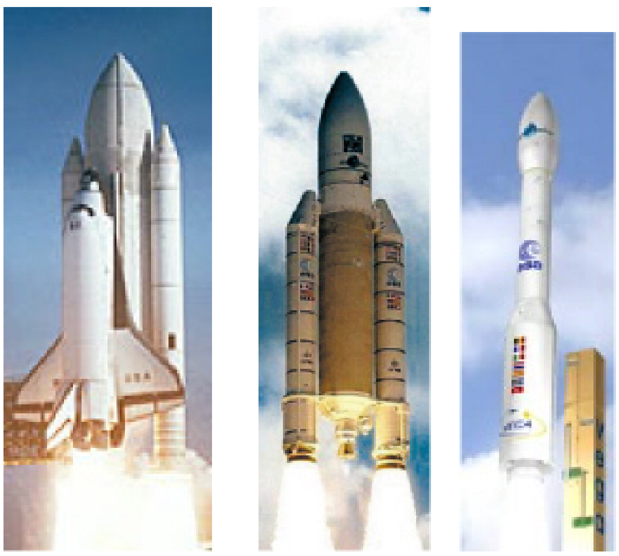
PAP of Ariane 3 and 4 steel casings made by AVIO — GEM 46 Composite case by ATK for Boeing's Delta III

Fig. 8: Solid Propellant Strap-on Booster

Up-to-date designs are made from Carbon fibre composite CFRP. The Delta's GEM motors or the Japanese' H II-B boosters are the actual representatives of this motor type. Filament wound over a dismountable mandrel, the motors offer a performance factor in the range up to 15 km due to less weight compared to steel.

3.2.4 Solid Propellant Stages

The next step in progress of SRMs enabled them to be used as self-standing stages for large launch vehicles. But there are overlaps with the previous described motor type:



US space shuttle — Ariane 5 — VEGA

Fig. 9: Solid Propellant Stages

Although the new European launcher VEGA does not belong to the GTO-capable large-sized launchers, its 1st stage motor P80 fulfils the categorization of a large-scale motor. For the launch of the US space shuttle, the two SRMs are today the largest and most powerful SRMs, but used as strap-on boosters. The previous pictures reflect typical representatives of this kind of SRM in the diameter class of > 3 m; similar designs are used with Atlas V, Titan 3c and 4, and will be used on the H II-B as well. While the US shuttle, Ariane 5, Atlas and earlier Titan types use high strength steel cases, Titan 4B, H II-B and Vega have Carbon composite cases.
Except for the monolithic case of VEGA's P80 stage, the general items of a large SRM case, shown in the following figure, are common for these motors.

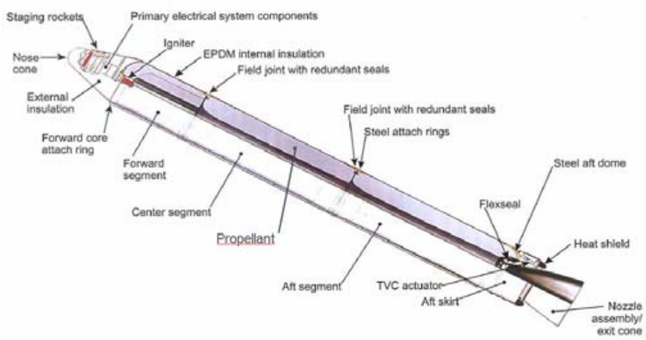


Fig. 10: Typical cross section of a large SRM

All segments of the metallic cases are composed of individual cylinders and domes, joint by a twin-lapped shear bolt concept called "clevis-tang.

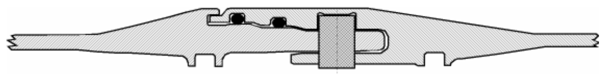


Fig. 11: Clevis-Tang joint of Ariane 5 booster case

In this case, the tightness of the joint is guaranteed by O-ring sealings. The opening tendency of the joint under applied inner pressure during operation is prevented by a retaining nose. Not present in the first shuttle design, this extremely-relevant security feature has been added after the first shuttle failure.
In the last decade, attempts to improve the shuttle boosters towards a welded and a CFRP composite design have been made, but finally failed due to budget cut-offs.
In Europe, however, the evolution of the Ariane 5 boosters, to weld the cylinders and domes to segments, has been qualified and successfully flown in 12/2006. This new design reduces significantly mass and costs, thus contributing to a better performance and a higher payload capacity of the launcher. An even more improved design was offered by ATK with its filament wound booster cases for Titan 4B. This concept finds a continuation in the Japanese H II-B boosters.
The new European launcher VEGA will be propelled by threes solid stages. Each motor, the largest is the 1st stage P80, is made in prepreg-winding technique by Avio. While the P80 has a capacity for 80 tons of "standard" HTPB-propellant with Amonium-Perchlorat oxidizer, the largest motor of this kind (US-shuttle) houses more than 300 tons of propellant. Further elongated by another segment, this design is the basis for the new NASA launcher family ARES I and ARES V.

3.3 Hybrid Propulsion

Hybrid propulsion has experiences a lasting surge that is driven by the quest for greener propellants and some spectacular applications such as that of an N2O/HTPB hybrid motor in Space Ship One (SS1). However the "Hybrid Propulsion Demonstration Program" at Stennis was not continued. As with liquid propulsion, vortex injection is a technological development aimed at performance improvement. Another goal of hybrid propulsion R&D is the development of scaleable design rules. Other investigations comprise cryogenic solid propulsion (CSP) in Germany and France (see chapter on research), and US research on high regression velocity LOX/paraffin hybrids. Without considering the many hybrid activities at university-student level, where hybrids remain an all-time favorite, the table below lists the currently most important professional activities. Prominent among these is, apart from S1, the break through implementation of paraffin as fuel for Oxygen (or N2O), which overcame the notorious regression rate problem of hybrids.

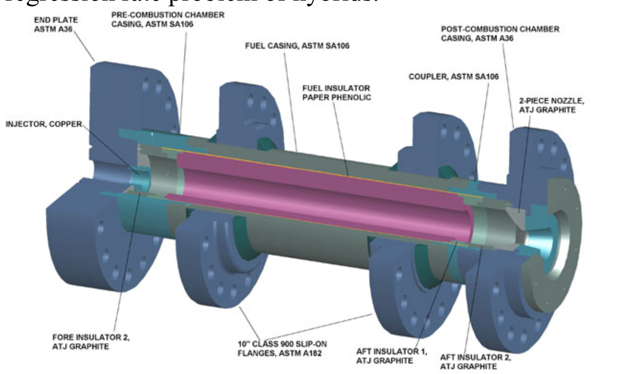


Fig. 12: SS1 hybrid motor schematic (Image NASA)

Fig. 13: SS1 hybrid motor in operation (Image courtesy J.Campbell & SC)

Propellant slumping can be prevented by molding the paraffin into a 3% by volume graphite sponge matrix. However, it remains to be seen, whether other shortcomings of hybrids (e.g. the shifting mixture ratio problem) can also be overcome in the future.

System	Prop.	State / year	Organisation	Remarks
White Knight / SpaceShip One	N2O/HTPB	Suborbital flights 2004	SC / engine by SD	Fuel evaporation required
250K Hybrid Rocket Motor	LOX / HTPB	Static test 01/ 23 2002	Consortium consisting of LMA, BR, TC, LMMSS, and UTC	Test at NASA Stennis SC
Hybrid Sounding Rocket	LOX / HTPB	Static test 07/ 21 2004	LMMSS	Test at NASA Stennis SC
UTAH-X propulsion system	N2O / HTPB	"3rd advanced hybrid experi-ment" 1996	Utah State University, Logan	Goal: small payload to an altitude of 130K ft
Flight Article Hybrid Motor	LOX/ Paraf-fin	Oct.2004 expts.	NASA Ames/ Portland State Univ.	He pressure fed

Tab.3: Current hybrid activities (LMA=Lockheed Martin Astronautics, BR= Boeing Rocketdyne, LMMSS= Lockheed Martin Michoud Space Systems, TC=Thiokol Corporation; UTC= United Technologies Chemical Systems Division; SC=Scaled Composites, Mojave; SD=SpaceDev)

Fig. 14: LOX/Paraffin hybrid test at NASA Ames (Img.:NASA)

3.4 Advanced Propulsion

Other advanced high thrust propulsion research concerns non-rocket and non-chemical propulsion.

The frontiers of air-breathing propulsion are characterized by keywords such as (US) ramjet missile propulsion, pulsed detonation engines, (EU) intelligent compressor development, (JAP) high temperature super alloy gasturbines, rocket-ramjet combined cycles and (AUS) scramjet development.

Non-chemical high thrust propulsion is limited to nuclear thermal propulsion, where NASA Marshall is leading the next wave of developmental efforts by the preparation of a Nuclear Thermal Environmental Simulator. Fusion propulsion, while waiting for progresses in terrestrial energy generation, is of course still limited to theory.

4. Propulsion: as Used in Current Space Transportation Systems

4.1 State of the art

As of June 2007, a worldwide total of 8 nations have the capability of transporting payloads into Earth orbit. These are, in alphabetical order: China (2/37-2), Europe (2/55-4), India (2/8-2), Israel (1/7-3), Japan (1/16-2), Russia (5/161-7), Ukraine (2/38-5) and USA (8/145-4), where figures in brackets are (number of different vehicle types / launch attempts since 2000, minus launch failures). The global percentage of successful launches was 94,1%. European launches after the turn of the century include 23 with Ariane 4 at 100% success rate. Use of Ariane 4 was terminated in Feb. 2003. Among 26 Ariane 5 launches were 2 complete and 2 partial failures. The total losses occurred at the maiden flight 6/1996 (caused by the Inertial Reference System) and at flight 14 in 12/2002 (caused by the Vulcain core stage engine). The partial failures occurred during the second flight in 10/1997 (caused by the nozzle of the core stage Vulcain engine) and during the 10th flight in 7/2001 (caused by mixture ratio problems in the upper stage). All subsequent flights 15 (4/2003) to 32 (5/2007) were successful.

About 30 different launch vehicles are currently in use, of which all but one, the US STS, are expendable and uncrewed. Single use, but man-rated, vehicles are available in Russia and China. Return stages and boosters exist with limited reusability. It is a lesson learned, that propulsion with man rated reliability is very expensive and that motor reusability pays off only at launch rates well above the present ones. All Earth-to-orbit propulsion and most of in-space propulsion is done with solid and liquid chemical propellants. Ion- and electro-thermal engines have been used for low-thrust transfer missions.

4.2 Survey of systems in use

This overview includes rocket systems and aircraft so far involved in space programs and being in active service at present.

Nation-ality	System	Stage(s)	intro-duced	remarks
China	CZ-3A	3rd	1994	
China	CZ-3B	3rd	1998	
China	CZ-5 (CZ-NGLV-A)	2nd + 3rd	Target 2010	Presented 2001
EU	Ariane 5G	1st	1996	
Japan	H-2A	1st+2nd	2001	Isp 440/447s
USA	US-STS Space Shuttle	External Tank + Orbiter	1982	Re-usable
USA	Atlas V	2nd Centaur	2002	Centaur with 1 or 2 engines
USA	Delta IV	1st + 2nd	2003	
Russia	Angara	2nd	in development	Chrunitschew launcher family

Tab. 4: Current rocket systems using fully cryogenic LOX/LH2 propulsion

Nationality	System	Stage	introduced	remarks
China	CZ-NGLV-A	Booster +1st	Concept 2010	Presented 2001
Russia	Soyuz-FG (ST)	1^{st},2^{nd},3rd	2001	FG = Fregat 4th stage
Russia	Angara	1st	In development	Chrunitschew launcher family
Ukraine	Zenit-3SL	1st+2nd +3rd	2000	Yuzhnoye
USA	Atlas V	1st "Common Core"	2005	Lockheed Martin
USA	Delta IV	core stage	2004	Boeing

Tab. 5: Current rocket systems using semi-cryogenic LOX/Kerosin propulsion

Nationality	System	Stage	Introduced	remarks
China	CZ-2F	1^{st},2^{nd},3rd		(99^{th} Long March launch in 5-2007)
EU	Ariane 5-2	EPS	1996	N2O4/MMH: Aestus Boeing Rocketdyne
India	GSLV	4x0 + 2nd	2001	N2O4/MMH: European Viking2 (4 Boosters) +V.4 engines
India	PSLV	2nd	1997	N2O4/UH25: Vikas engine
North Corea	Taepodong-1	1st , 2nd	1998	Nitric acid/UDMH: medium range ballistic missile
Russia	Soyuz	Ikar 4th stage		N2O4/UDMH: Melnikov 2.94kN engine. 30 flights as of 1998
Russia	Soyuz FG	Fregat 4th stage	1987	N2O4/MMH: 1,96t thrust S5.93 engine, 20 restarts
Russia	Proton K	1st, 2nd, 3rd	1967	N2O4/UDMH: 6xRD-253/163,5t +4xRD0210/59,4t+1xRD0212/62,5t engines, 55 flights 2000 - 2006
Russia	Dnepr	1st, 2nd, 3rd	1999	N2O4/UDMH: RD266/228/869
Russia	Rokot/Briz KM	1st, 2nd, 3rd+AC +RC	2000	N2O4/UDMH: Isp: 285SL>310vac/322/325,5s
Russia	Angara	Briz M 2nd stage		N2O4/UDMH: Briz Space Tug on Angara launcher family
USA	US-STS Space Shuttle	Orbiter OMS	1982	N2O4/MMH: Aerojet Orbital Maneuvring System
USA	Beal BA-2	1st	1995-2000	H2O2/Kerosene: Planned 1500t engine, tested 360t vac-thrust largest liquid rocket engine ever after F1

Tab. 6: Current rocket systems using storable propellants

Nationality	System	Stage	introduced	remarks
China	Kaituo zhe-1	1^{st} 2^{nd} 3^{rd} 4^{th}	2002	Commercial all solid LV, based on road mobile DF-31 ICBM. 100kg to LEO, by Solid Fuel Rocket Carrier Co. Ltd.
EU	Ariane 5	EAP Booster	1996	
India	PSLV	1^{st}+3^{rd}	1997	138t + 7,6t SP
India	GSLV	1^{st}	2001	S125
Japan	H-2A	SRB-A, SSB	2001	2x66t SP in SRB-A + (opt.) 2 or 4 CASTOR SSB
USA	Space Shuttle	SRB	1982	2x586,3t SP Thiokol
USA	Atlas V	SRB	2002	2x40,8t SP in Aerojet SRB
USA	Delta IV	SRB GEM60	2003	0 to 5 SRB
USA	Pegasus XL	1st +2nd	1994	3x Hercules Onion SRM; by 2006: 35 successful air-launch missions, 3 failures.

Tab. 7: Current Rocket Systems using solid Propellants

4.3 Current air breathing propulsion

Fuels considered were relevant for space related air-breathing jet or ramjet propulsion, such as stage return carrier aircraft, but also considered for concepts of systems with air-breathing propulsion. Most civil jet engines use "Aviation Fuel", a kerosene type hydrocarbon mixture with freezing point -40°C (in USA: JET-A (or „A-1",FP-47°C) and a combustion energy of about 43,5 MJ/kg. The aircraft mentioned below are used as space vehicle piggy-back transporters or as air-launch vehicle carriers:

4.3.1 Air / Kerosene

Presently in use in Russia: An-225, Tu-160, MiG-25 ; inUSA: NASA 747; L-1011; SR-71; F-6A; F-15; NB-52

4.3.2 Air / (LOX) / LH2

For lack of operational systems, the following historical concepts can be mentioned: EU: Sanger II-1 1986 / Ru: N1-MOK Liquefied air cycle engine 1974; Tu-2000 Air/Slush Hydrogen Scramjet Study 1993/ USA: Nova MM Heavy Lift LVs with air augmented plug nozzles; X-30; LACE winged orbital plane

5. Structures

5.1 Liquid Propellant Stages

5.1.1 Storable Upper Stages

The launchers servicing today for transportation to orbit mainly are equipped with a storable propellant upper stage, lifting the satellite platform to its designated orbit. Using hydrazine (MMH) and multi nitrogen oxides (MON) as propellant, the tanks for both agents are approximately of the same size. Usually, additional tanks for pressurization of the propellant tanks are implemented in the upper stage. The following figures show some examples of storable upper stages:

Fig. 15: Fregate upper stage of Soyuz

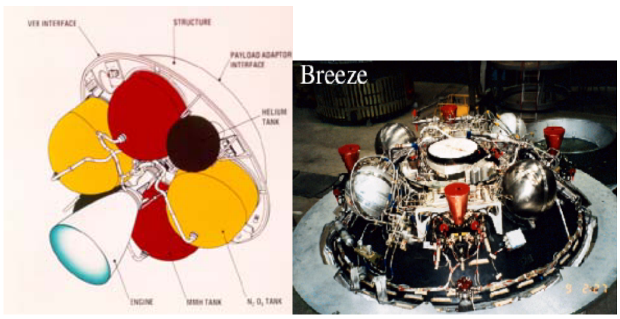


Fig. 16: Ariane 5-EPS and Rockot-Breeze upper stage *(courtesy IRS of Uni-Stuttgart)*

5.1.3 Cryogenic Upper Stages

Although the specific density of a LH2/LOX fuel system is less than that of storable propellants, the specific impulse balances this drawback by far and guarantees higher payload capacities. Today's insulation capabilities additionally allow a significant time offset between filling of the tanks and stage ignition. Moreover, the toxidity of the storable propellants is eliminated.

Therefore, new upper stage designs rely on the cryogenic techniques.

Also the improved ARIANE 5 launcher is already quailfied with the cryogenic upper stage ESC-A. In order to avoid re-construction of the relevant buildings at the launch site in Kourou, a configuration with a LOX-tank nested within the LH2 tank has been designed.

The complexity of a common bulkhead with the LH2-tank on top was regarded as not yet mastered; this subject is still submitted to extensive investigations.

The upper stage of Delta IV, however, offers a more convenient detached design, with the LH2-tank on top of the LOX tank, but having drawbacks for the overall length of the stage.

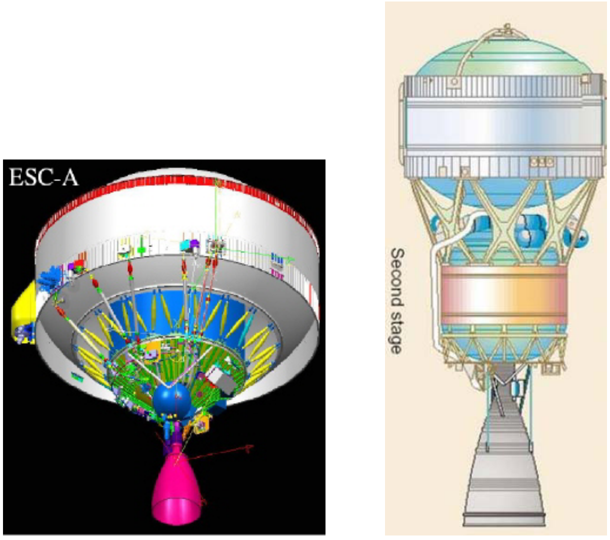


Fig. 17: Ariane 5-ESC-A nested design and Delta IV Ø 5,1 m version, detached design

5.1.2 Structural Tank Components of ARIANE 5 Main stage

Compared to liquid propellant stages of other launchers, the cryogenic 1st stage of Ariane 5 offers some specific items:

- As the booster loads are introduced by the front skirt on top of the first stage, and the launcher stands on the twin boosters on the launch platform, the structure of this stage has mainly to be designed for tensional loads. Only after separation of the booster, the Vulcain main engine's thrust has to be transmitted.
- Due to that lay-out, the walls of the cryogenic tanks could be kept thin, forming the outer boundary of the stage.
- The lower located liquid hydrogen tank (LH2) and the upper liquid oxygen tank (LOX) have a single-walled common bulkhead, avoiding a "wet" insulation at the LH2 side. Thus, an inter-stage structure is not necessary.

Common with most other launchers, the bulkheads of the liquid tanks are made in gore panel technique: Up to 8 pre-formed panels are welded together to form the dome, the polar opening flange (if any) and the Y-ring as transition to the cylindrical part are welded as well.
The Japanese H II-A, however, is already equipped with net-shape spin-formed domes; this technique enables a reduction of highly accurate machining and welding assembly efforts.
This concept could also be used in the liquid stages of other existing launcher, but needs to be individually qualified.

Fig. 18: Large Tank Bullkheads:
Gore-panel technique of Ariane 5 EPC
Spinformed Dome for Japanese H II-A
(courtesy MT Aerospace)

5.2 Load Carrying Structures

In that sense as described above, the large tank components of a launcher are a load carrying structures by themselves ("self standing"). But there are also load carrying or load introducing structures between those large tanks, which have to fulfill very specific requirements. The Ariane EPC-front-skirt is a complex, but typical example. Located right above the cryogenic tanks of the main stage, it has to take over the axial forces (weight and thrust) of the so-called "lower composite" and the forces of the two boosters. The temperature boundary conditions of the interface to the cryogenic tanks contribute with very specific requirements. Moreover, different missions and payload introduce their loads during ascent from top of the structure.
Similar to many other large launchers, the A 5 front-skirt is made in Aluminum alloy, stiffened with rips and spans, elements joined by rivets, in a very classical - manufacturing process driven by aerodynamics.

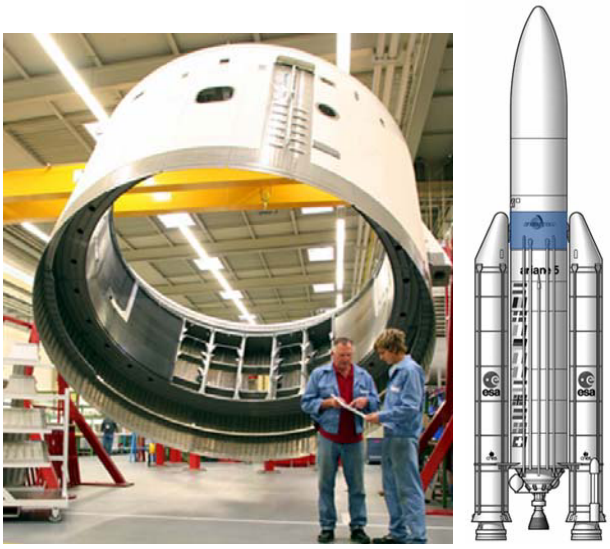


Fig. 19: Ariane 5 front skirt of the main stage
(courtesy MT Aerospace)

5.3 Hot structures / TPS

Thermal Protection Systems (TPS) are necessary to prevent thermal overloading of the external skin and thus destruction of re-entry vehicles during re-entry. TPS can withstand temperatures up to 1600 °C depending on the selected re-entry trajectories of manned re-entry vehicles. TPS consists of Ceramic Matrix Composite (CMC) material as depicted in the following figure. The material is a composite of woven carbon fibre fabrics infiltrated with Silicon Carbide, either in the gaseous or the liquid phase. Maximum heat fluxes appear at the nose cap and at the leading edges of the fins. The same material was used to develop the C/SiC body flaps, which are so called hot structures necessary to steer the re-entry vehicle according to the selected flight path. The hot structure Body Flaps are completely manufactured by C/SiC material - also the bearings - and therefore they are extreme light weight and high temperature resistant up to 1800 °C for short term temperature peaks.

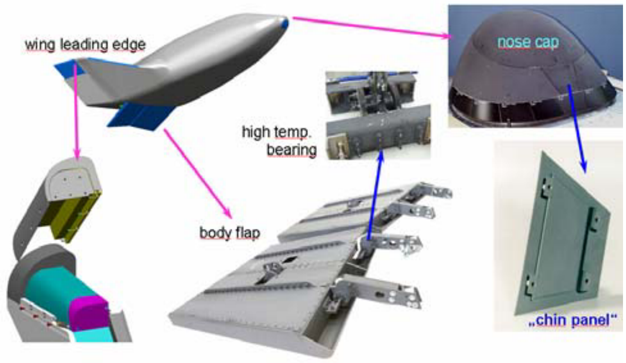


Fig. 20: Hot structural elements of a re-entry vehicle *(courtesy MT Aerospace)*

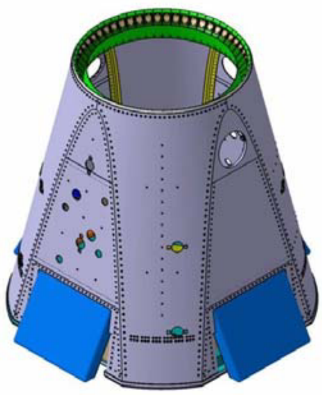


Fig. 21: XPERT hot metal TPS configuration with ODS metal cone, CMC nose and Flaps *(Image Courtesy Dutch Space - EADS Astrium)*

Another example, meant for a European re-entry demonstration vehicle , is the XPERT hot metal TPS, shown in Fig., a project of Dutch Space. Two versions of a generic TPS tile are considered, an ODS super alloy tile (850°C) and an ultra light y-TiAl tile version with temperature range up to 1250 °C.

6. Future Prospects

Many unknowns limit the scope of an outlook at the future of space transportation systems and their propulsion, propellants and structures. Foremost it is the course of space transportation itself, which opens an all too wide field of uncertainties preventing any more specific predictions of future propulsion needs. However, after the implementation of manned and unmanned elements of near Earth infrastructure will come Lunar development and an increasing demand of manned high thrust propulsion in near-Earth and cis-Lunar space and as beyond, if Mars is to be explored by astronauts. Short of unknown unknowns it is safe to predict that for many decades to come all Earth-to-orbit propulsion – the basic step into space! – will have to rely on chemical rockets. In doing so, environmental aspects will be of increasing importance. As a consequence, the replacement of AP based solid propellants by more benign ones will have to be a rather early change in chemical propulsion, for which there are several emerging solutions. Winged and re-usable orbital transportation will rely on advanced liquid rocket motors. All remaining traffic, the majority of missions, will use liquid, solid and hybrid propulsion as today. Suborbital and orbital tourism is likely to grow soon into an application of particularly high demand for reliable green propulsion, such as hybrids with non-toxic oxidizers. However, in other applications, hybrids and solids are going to make increasing use of cryogenic solid propellants, as these are the precondition for the implementation of very high energy propellants. While HEDM propulsion does have the potential of improving space transportation systems into realms of performance that were hitherto unthinkable, they too will hit on limitations posed by environmental concerns, in this case noise emission. While noise growths only with the square root of vehicle size (all other parameters kept constant), it grows with a rather high power of Isp. It may well be that air-breathing rockets will have to be implemented for that reason.

In space high thrust missions may become the true place for very high energy chemical propulsion, missions where it may be in competition with nuclear thermal fission propulsion. However, as in terrestrial applications, this kind of nuclear propulsion does have serious radiation problems that might prevent its use. The advent of nuclear fusion propulsion belongs to the known unknowns: it will be there 10 to 20 years after the appearance of applied terrestrial fusion power. On the other hand, nuclear energy provided by radio isotopic batteries is sure to find increasing application for long duration, low thrust deep space missions to the outer parts of the Solar System. Another innovative technology is likely to have an impact on short distance propulsion: beamed power transmission (as microwaves and laser light) to orbital users.

Of course, let one of the unmentioned global variables change, and everything might come completely different. Remember: nothing is more difficult to predict than the future!