HM60 Cryogenic Rocket Engine for Future European Launchers

M. F. Pouliquen*
Société Européenne de Propulsion, Vernon, France

In order to improve the performance of the Ariane launchers family and to reduce specific cost of payloads in LEO and GEO, tradeoff studies were conducted in 1981 and 1982 that led to the need for a new LOX/LH2 rocket engine named HM60, generating 900 kN thrust in vacuum. Tradeoff studies were made at the engine level also; based on these studies, the gas generator cycle was selected in order to reduce production costs and development risks. At present the nominal characteristics of the engine are: combustion pressure 100 bar, mixture ratio 5:1, specific impulse 445 s, and single gas generator feeding in parallel two independent turbopumps.

Introduction

STUDIES made in Europe concerning launching systems (Fig. 1) needed for the nineties have shown the necessity of developing a high-thrust LOX/LH2 rocket engine. As early as 1978, preliminary studies were started on a 500-kN thrust LOX/LH2 engine in order to evaluate the technology effort needed for such an engine in Europe.

In 1979 and 1980 a two-stage Ariane V launcher (Fig. 1) concept was arrived at, composed of the first stage of Ariane IV and a new LOX/LH2 H60 second stage (Fig. 2), able to carry 15,000 kg payloads (representing half the shuttle capacity) into low-Earth orbit at a reasonable operational cost; the reduction of cost per kilogram in orbit being the main criterion for designing the whole Ariane V vehicle and its subsystems. The first specifications of the HM60 engine to arise from the first concept of Ariane V resulted in a nominal thrust of 800 kN in vacuum, but with a possibility of reducing thrust to 600 kN in flight, in order to limit acceleration for manned missions.

This was followed by tradeoff studies of the engine and subsystems in order to start a technological program covering the most critical items in 1981. The development of the engine is scheduled to start in 1984 and be completed by 1991 to allow a first flight of the Ariane V in 1993-94.

This paper gives the main results of the tradeoff studies and presents the reference configuration of the engine.

Tradeoff Studies

The main specifications selected to make the comparisons were:

a) Specific impulse: higher than 443 s in vacuum.

b) Nominal thrust: 800 kN in vacuum.

c) Potential growth: 1300 kN in vacuum. The possibility of increasing the thrust to 1300 kN was put forward in order to be conservative for using this engine on a first stage for future launching systems. In addition, the first version of the engine at 800 kN could be developed with low risks, taking into account the "over-design" of the engine.

d) Overall dimensions: length, 4.0 m and maximum diameter, 2.4 m. These dimensions allow a conservative separation of the stages in flight. Optimization studies showed, however, that an interesting increase of payload could be obtained by increasing the area ratio of the nozzle. This possibility was

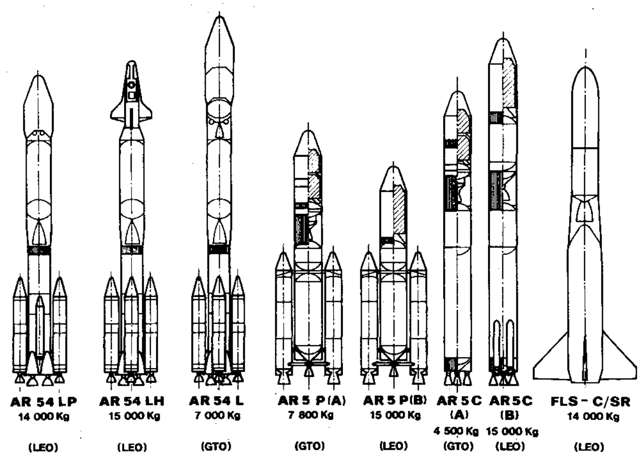


Fig. 1 European launchers, existing and under study.

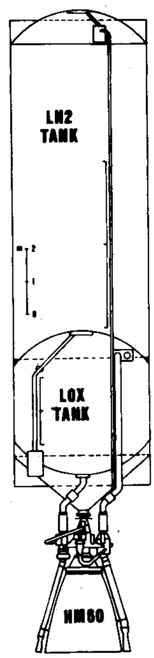


Fig. 2 H60 stage.

Presented as Paper 82-1250 at the AIAA/SAE/ASME 18th Joint Propulsion Conference, June 21-23, 1982; submitted Sept. 24, 1982; revision received Oct. 13, 1983.

*HM60 Project Manager. Member AIAA.

Table 1 Comparison of HM60 to SSME

	HM60 Gas generator		HM60 Staged combustion		SSME
Vacuum thrust (kN)	800	1300	800	1300	2092 (100%)
Sea level thrust-sea level engine (kN)	624	1054	654	1104	1669
Engine mixture ratio	5.12	5.12	5.58	5.58	6.0
Thrust Chamber:					
Combustion pressure (Bar)	100	160	125	203	205
Nozzle area ratio	103.7	103.7	124.4	124.4	77.5
Gas generator/precombustor:					
Combustion pressure	50.6	115.6	194	355	356
Mixture ratio	0.9	0.9	0.68	0.9	0.81
Turbopumps, LH2/LOX					
Discharge pressure, BAR	143/122	243/218	225/153(257)	415/248(486)	13/296(480[a])
Shaft speed, RPM	30000/11700	40500/16140	25000/21900	35000/31100	34700/27500
Turbine power, MW	7.6/2.0	21.2/5.6	10.8/2.8	32.4/8.6	45.5/18.6

[a] Discharge pressure of the second stage of the LOX pump.

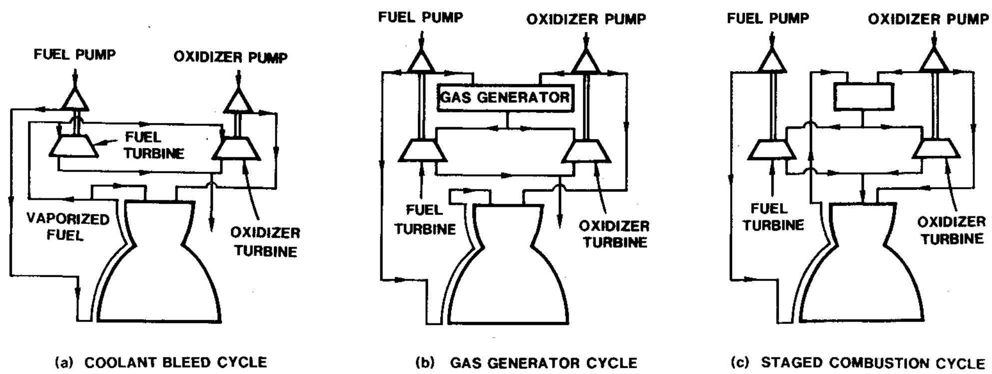


Fig. 3 HM60 cycle candidates.

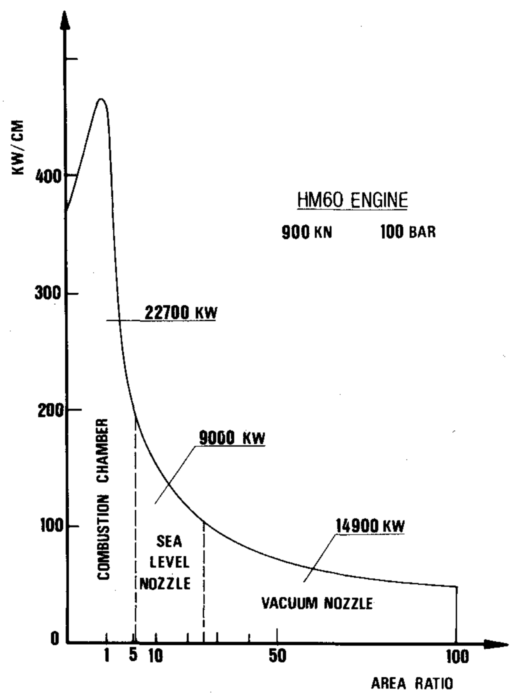


Fig. 4 Thermal energy in the thrust chamber walls.

considered as a potential improvement in the same way as the use of a deployable or extendable nozzle.

e) Propellant conditions at the inlet of the pumps: critical value of Net Positive Suction Pressure (NPSP) of 1.5 bar for LOX and 0.5 bar for LH2 in order to preclude the use of boost pumps, although some studies are presently being made to evaluate their performance.

f) Engine usable on a first stage (after some modifications of nozzle, turbine exhausts, etc.).

g) One reignition must be possible in flight.

h) An engine recoverable and reusable (engine to be used on future recoverable launchers).

Cycle

Three cycles were studied (Fig. 3): 1) heat exchanger (coolant bleed cycle), 2) staged combustion, and 3) gas generator.

The expander cycle was not selected for the study because its combustion pressure level is limited to low values for high thrust engines, because of the limited power supplied from heated fuel and the low pressure ratio of the turbines. These same types of limitation will be demonstrated for the coolant bleed cycle engine.

Coolant Bleed Cycle (Fig. 3a)

A small percentage of the LH2 flow rate is heated to a high temperature (900 K) in a heat exchanger portion of the main thrust chamber. The heated fuel is expanded through high-pressure-ratio turbines and is exhausted separately from the main thrust chamber. The principal advantages of this cycle are its simplicity, leading to low production cost of the engine, and the relatively low pressure of the pumps needed for a given chamber pressure.

However, this cycle was only briefly pursued, because it appeared that the thermal energy available to drive the turbines for a chamber pressure of 100 bar was not sufficient. This is the major problem of the coolant bleed cycle, because the heat transfer capabilities are a limitation on the power of the turbines. At 900 kN thrust and 100 bar with optimistic values of turbine efficiencies and no margins for regulation, it is necessary to use all the heat transfer surface for the heat exchanger, including the nozzle extension and the combustion chamber. Figure 4 gives the maximum thermal energy released from the combustion chamber to the exit of the nozzle. After this first evaluation, studies on the coolant bleed cycle were not pursued.

Staged Combustion (Fig. 3c)

The detailed flow schematic of the staged-combustion HM60 is represented in Fig. 5.

The thrust chamber utilizes a technology similar to the Space Shuttle Main Engine (SSME), with a copper alloy combustion chamber, milled channels closed by electroformed nickel, and a welded tube construction nozzle extension. Of the LH2 flow rate, 20% is used to cool the combustion chamber, and 6.5% is used in the dump cooled nozzle. The thrust chamber is fed by two separated turbopumps, driven from hot gases produced by a single precombustor integrated to the LH2 turbopump.

In the study, the configuration of the turbopumps chosen was very similar to the SSME, but with inducers to meet specified NPSP (no boost pump upstream of the main pumps). Regulation of mixture ratio is obtained by a valve at the outlet of the LOX pump. The igniters are pyrotechnic, and starting is obtained by using the tank pressure head. The general arrangement of the engine is given in Fig. 6a.

In order to prevent propellant accumulation problems in the precombustor during starting and after shutdown, the turbine of the fuel turbopump is on the upper side (with the integrated precombustor in a vertical position). The inlets of the pumps are connected to the stage by vertical feed lines with large deflection bellows for engine gimballing.

Gas Generator Cycle (Fig. 3b)

The detailed flow schematic of the gas generator cycle is given in Fig. 7.

The thrust chamber technology is similar to that of the staged combustion, but a fraction of the total LH2 flow rate

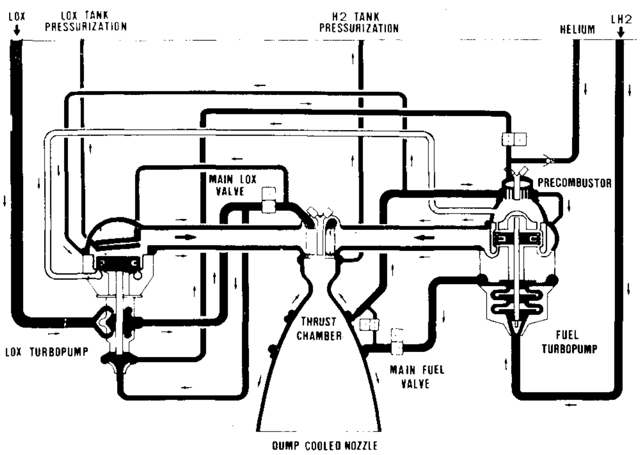


Fig. 5 HM60 staged combustion engine.

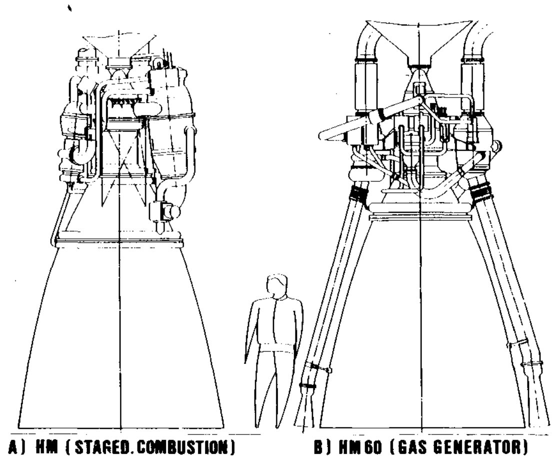


Fig. 6 HM60 engine arrangement.

Table 2 HM60 Engine Characteristics

	HM60	SSME
Thrust		
Vacuum, kN	900	2090
Sea level (sea level version), kN	715	1700
Specific impulse		
Vacuum, s	445	455.2
Sea level version (at sea level), s	349	363
Sea level version, vacuum, s	424	-
Mixture ratio	5.1	6.0
Chamber pressure, bar	100	207
Area ratio, kg/s	110.5	77.5
Flowrates		
Engine,	206	468
Gas generator	7.06	248
Dump cooling	1.93	-
Pump discharge pressure, bar		
LOX	125.7	319(528)
LH2	150.5	426
Length, m	4.0	4.24
Nozzle exit diameter, m	252	2.39
Combustion time, s	291 (p/H60)	480
Mass, kg	1300	3002

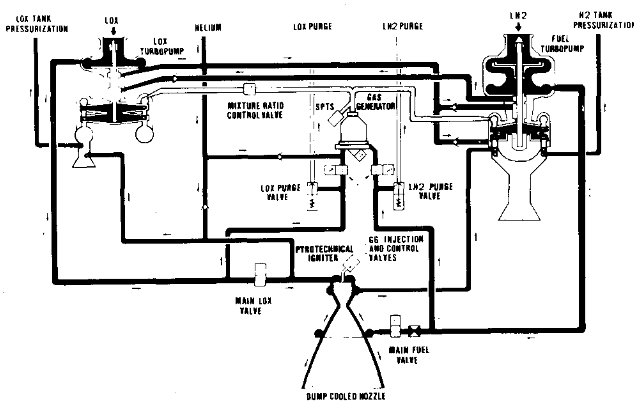


Fig. 7 HM60 gas generator cycle flow diagram.

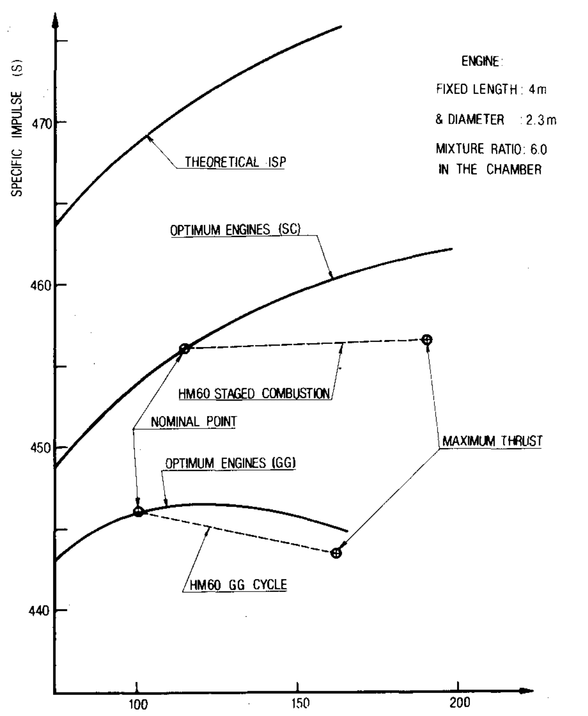


Fig. 8 Vacuum specific impulse.

bypasses the regenerative circuit in order to limit the height-to-width ratio of the channels to a value of 4. The dump cooled nozzle is cooled by 3.8% of the LH2 flowrate. The turbopumps, described further on, are fed in parallel from a single gas generator.

Ignition of the gas generator and combustion chamber is obtained by pyrotechnic igniters, starting of the turbopumps by a solid propellant turbine starter. Regulation of mixture ratio is similar to the staged-combustion engine. The general arrangement of the engine is given in Fig. 6b.

Comparison

The main characteristics of the two engines are given in Table 1 and are compared to the SSME at 100% power level. It can be seen that the pressure levels are always lower on the HM60 than on the SSME. The performances are compared in Fig. 8, where vacuum specific impulse as a function of combustion pressure is given for the two engines between the nominal thrust level (900 kN) and maximum thrust level (1300 kN). The upper curves for the staged combustion (SC) and gas generator (GG) cycles correspond to optimized engines at the combustion pressure considered and with a fixed length of the engine of 4.0 m (so with variable area ratio). The lower straight curves give the performance of the HM60 engines considered at fixed area ratio, operating at variable combustion pressure (uprating). The decrease of specific impulse for the GG cycle corresponds to the increased flow rate fraction in the gas generator cycle. In the two cases, the HM60 engines were optimized at the minimum combustion pressure point, corresponding to 800 kN thrust.

The weights of the engines have been determined from the drawings of the integrated engines and subsystems and are compared with existing engines in Fig. 9. The detailed development plans were established for the two engines, resulting in a total duration of 7.5 years for the gas generator engine and 8.75 years for the staged combustion engine (including the building of the facilities). The development cost for the staged combustion engine was found to be 25% higher than for the gas generator engine, and the production cost 20% higher for the 800 kN version. The gas generator was finally selected, essentially because of the higher development risk and higher production cost of the topping cycle.

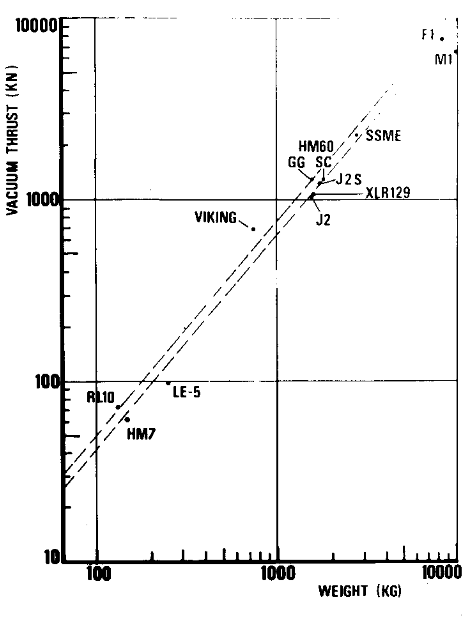


Fig. 9 Weight of existing engines.

Present Status of HM60 Definition

The Main Specifications

The modifications to the specifications used in the comparison are:

a) Nominal thrust: 900 kN in vacuum.

b) In addition to generation of main thrust for the stage, an engine capable of: 1) Allowing gimballing for pitch and yaw control; 2) Furnishing GOX and GH2 for pressurization of the tanks; 3) Furnishing a flow rate of 0.150 kg/s for the Roll Control System.

c) Design specifications: Fig. 10 gives the conditions (thrust and mixture ratio) for the design and operation of the engine.

d) Production cost of the engine minimized; and Design to Cost used intensively.

e) Maintenability of the engine taken into account for multiuse ability on reusable launchers.

f) Engine capable of use for manned flight, after minor modifications.

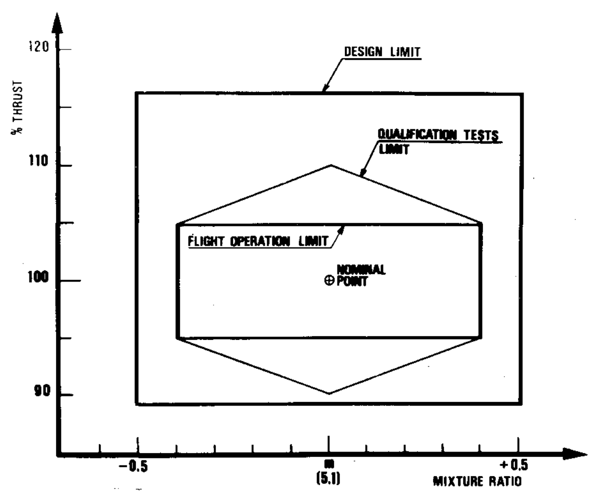


Fig. 10 HM60 design and operation limits.

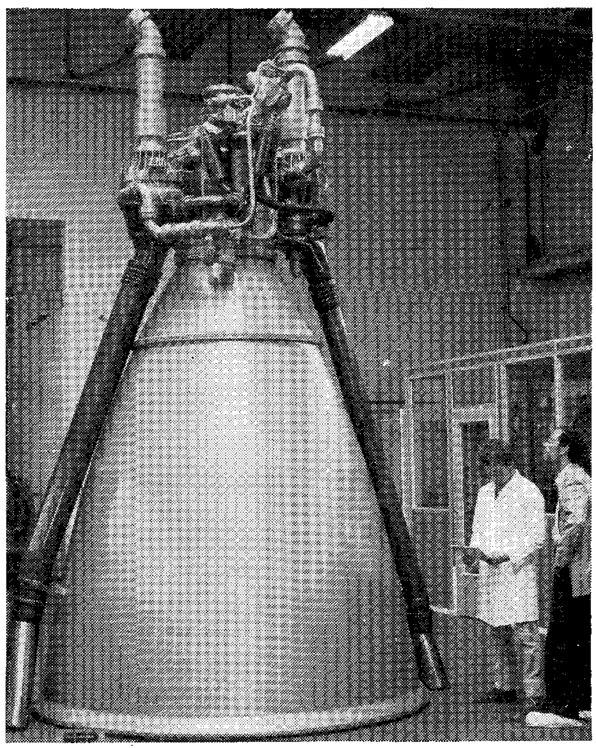


Fig. 11 HM60 full scale mockup.

Main Characteristics of the Engine

The flow schematic of the engine is represented in Fig. 7. The thrust chamber has a regeneratively cooled combustion chamber and a dump cooled nozzle extension. The two independent turbopumps are driven in parallel from a single gas generator using LOX/LH2 injection. The exhausts of the turbines are separated from the main thrust chamber. Starting of the turbines and ignition of the gas generator and main chamber are obtained from pyrotechnic systems similar to those presently used on the HM7 ARIANE 1 engine. [1-3] Mixture ratio of the engine is controlled through the use of a hot-gas control valve which varies the power of the LOX turbopump turbine. Thrust of the engine and mixture ratio of the GG are controlled through the use of the injection/control valves of the gas generator. The engine computer must insure all the checking and control functions, using transducers mounted on the engine and in the tanks. The main characteristics of the engine are given in Table 2. Figure 11 shows the full-scale mockup of the HM60.

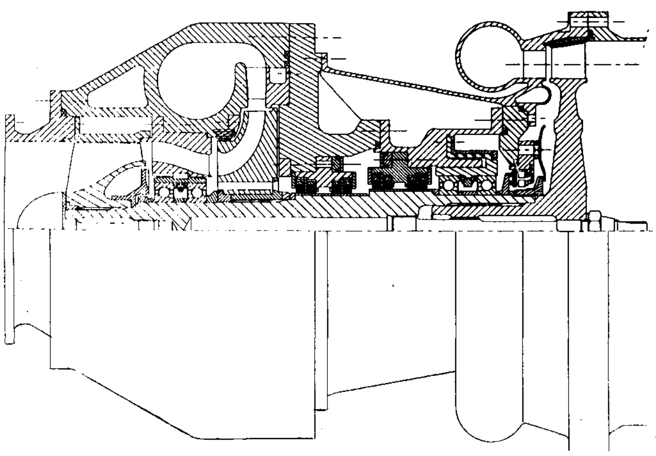


Fig. 12 Oxygen turbopump.

Turbopumps

The LOX turbopump (Fig. 12) has an inducer, one-stage centrifugal pump, and an impulse turbine. The inducer and the impeller are made from aluminium alloy and the turbine from INCO 718. The bearings are located between the inducer and the impeller and between the pump and the turbine disks. The pump bearings are lubricated by LOX and the turbine bearings by LH2. The absolute sealing between LOX and LH2 is obtained by floating ring dynamic seals and a helium barrier. A liftoff seal is used to prevent any LH2 leakage in the turbine during precooling (before starting). Axial load balancing is obtained by calibrating the LOX flow on the back side of the impeller. The main characteristics of the LOX turbopump are given in Table 3. The hydrogen turbopump (Fig. 13) has an inducer, a two-stage centrifugal pump, and a two-stage total injection impulse turbine. Due to the power transmitted by the shaft, the ball bearings are located outside the pump/turbine section, in order to have reasonable DN

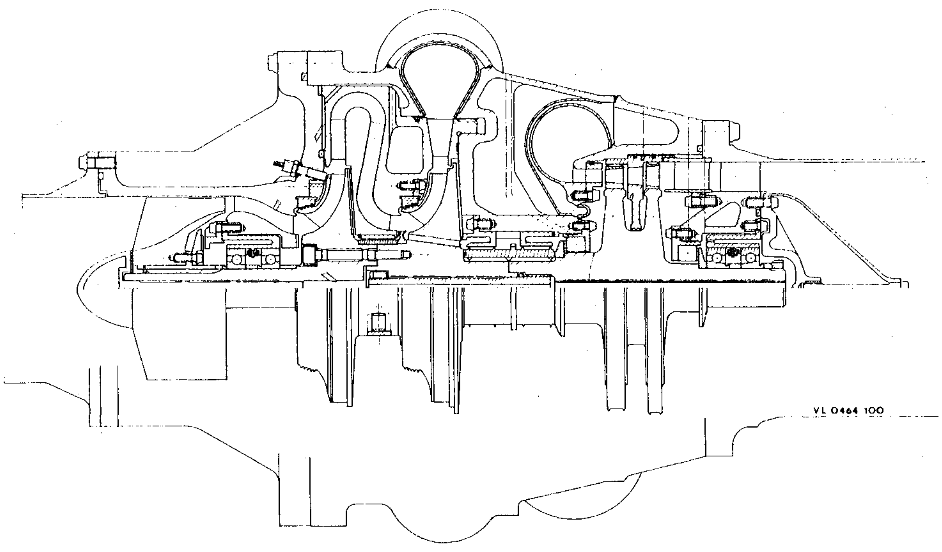


Fig. 13 Hydrogen turbopump.

Table 3 LOX Turbopump characteristics

Shaft speed	14,500 rpm
Flowrate	173.4 kg/s
Pump discharge pressure	125.7 bar
Shaft power	2331 kW
NPSP*	<1.5 bar
Pump	
Number of stages	inducer + 1 impeller
Diameter	205 mm
Specific speed	0.545 (1490)
Efficiency	0.79
Turbine	
Number of stages, type	1 impulse stage
Diameter	230 mm
Pressure ratio	17
Efficiency	0.29
Bearings	
Maximum D × N LOX 0.82 10^6	mm × rpm
Maximum D × N LH2 1.06 10^6	mm × rpm

Table 4 Hydrogen turbopump characteristics

Shaft speed	37900 rpm
Flowrate	34.07 kg/s
Pump discharge pressure	150.5 bar
Shaft power	8660 kW
NPSP*	<0.42 bar
Pump	
Number of stages	inducer + 2 stages
Diameter	205 mm
Specific speed	0.534 (1460)
Efficiency	0.77
Turbine	
Number of stages, type	Velocity compounded two stages
Diameter	201 mm
Pressure ratio	20.5
Efficiency	0.50
Bearing	
Maximum D × N	1.5710^6 mm × rpm

Table 5 Main injector characteristics

	J2S	RL10	SSME	HM7	MBB, exp. t.c.	HM60
Total injector mass flow, kg/s	242	18,5	469	13,9	45	195,8
Chamber diameter, mm	470	262	450	180	182	415
Number of inj. elements	614	216	600	90	90	516
Mass flow per element, g/s	375	85.6	782	70.7	470	380
H_2-injection temperature, K	105	180	850	136	190	95
c*-efficiency	0.98	0.985	0.99	0.986	0.98	0.989

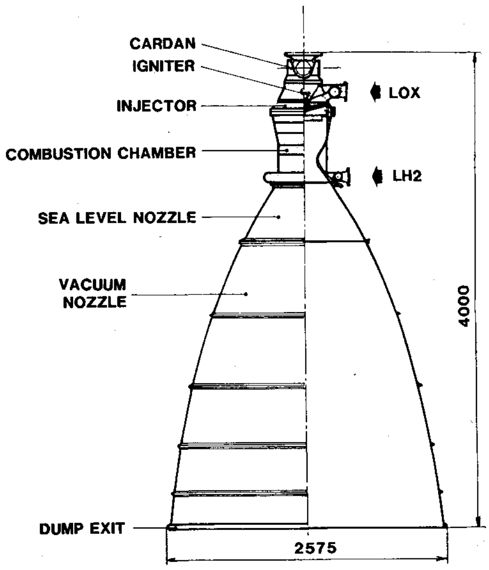
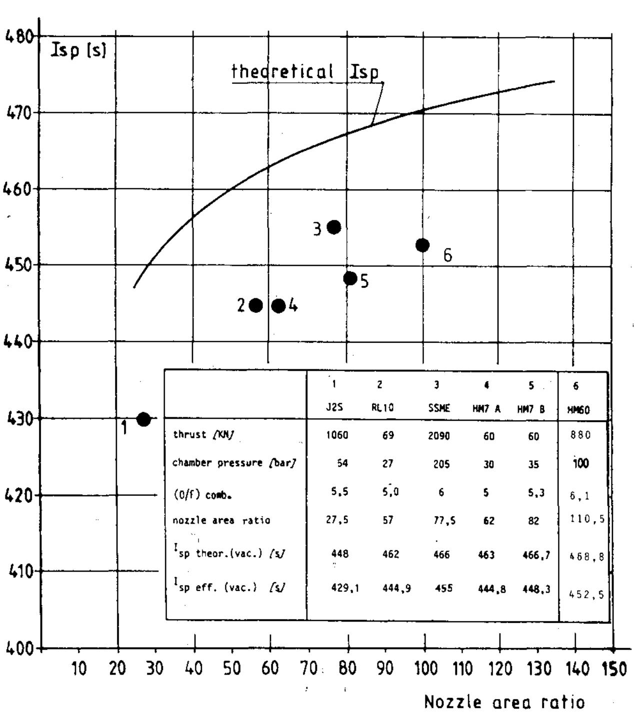


Fig. 14 HM60 thrust chamber.



	1 J2S	2 RL10	3 SSME	4 HM7 A	5 HM7 B	6 HM60
thrust [KN]	1060	69	2090	60	60	880
chamber pressure [bar]	54	27	205	30	35	100
(0/f) comb.	5,5	5,0	6	5	5,3	6,1
nozzle area ratio	27,5	57	77,5	62	82	110,5
I_{sp} theor.(vac.) [s]	448	462	466	463	466,7	468,8
I_{sp} eff. (vac.) [s]	429,1	444,9	455	444,8	448,3	452,5

Fig. 15 Thrust chamber characteristics.

(bore diameter × rotational speed) values. All the bearings are lubricated by LH2. As on the LOX turbopump, a liftoff seal is used on the turbine side. The axial load balancing system is integrated to the back side of the second impeller. The inducer is made from aluminium alloy, the impeller from titanium alloy TA5E-ELI, and the turbine and the shaft from INCO 718. The characteristics of the LH2 turbopump are given on Table 4.

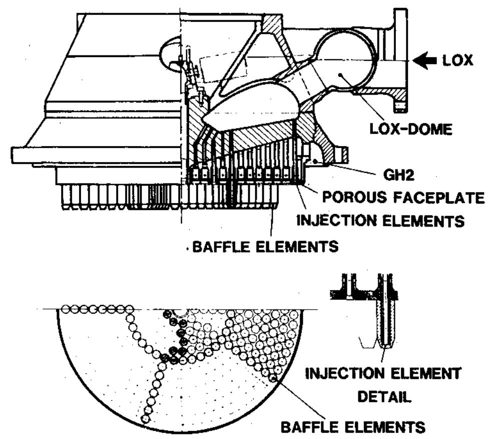


Fig. 16 Main injector.

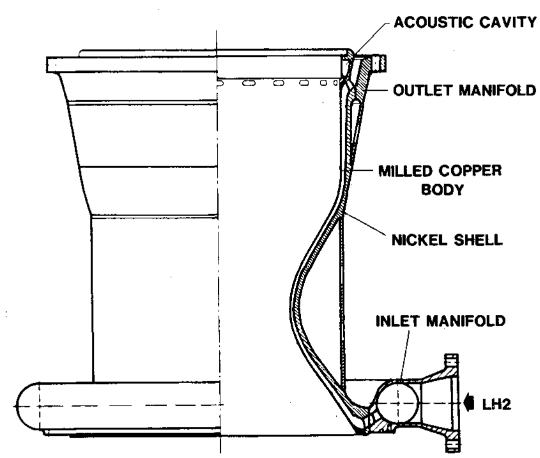


Fig. 17 HM60 combustion chamber.

Thrust Chamber

The general arrangement of the thrust chamber is given in Fig. 14 and is composed of the gimbal joint, main injector, solid propellant igniter, combustion chamber, and nozzle extension. The performance and main characteristics of the HM60 thrust chamber are given in Fig. 15, and are compared to existing thrust chambers. The components are designed on the basis of existing technologies in Europe which were developed during previous programs. The main injector (Fig. 16) uses 516 coaxial elements mounted on a porous face plate, cooled by transpiration of hydrogen representing 5% of the total flow rate. Comparison with other cryogenic injectors design data is given in Table 5. Acoustic baffles are obtained by the increased length of the elements.

The combustion chamber (Fig. 17) includes a divergent part to an area of 5.8 and is regeneratively cooled by hydrogen. The inner part of the chamber is made from copper alloy, in which the cooling channels are milled and then closed with electroformed nickel. The manifolds are made from INCONEL and are welded to the nickel part. Comparison with other cryogenic combustion chamber design data is given in

Table 6 Combustion chamber characteristics

	J2S	RL10	SSME	HM7	MBB, exp. t.c.	HM60[a]
Chamber inner diameter, mm	470	262	450	180	182	415
Characteristic length, m	0.62	0.98	0.8	0.7	2.3	0.85
Contraction ratio	1.58	2.95	2.96	2.78	6.95	2.99
Coolant temperature rise, k	60	150	254	100	140	61
Coolant pressure drop, bar			98	5.7	100	23.3
Maximum wall temperature, K			~740	625	690	600
Maximum heat flux, W/cm^2			12,800	2,900	16,800	6,400
Chamber pressure, bar	54	27	205	35	280	100

[a] Values for OFHC copper.

Table 7 HM60 compared with existing engines

	HM 7 A	HM 7 B	LE-5	HM 60	J 2	J 2 S	RL,10 A3-3	SSME
Vacuum thrust, kN	61.6	62.7	100	900	1044	1180	67	2090
Specific impulse (VAC), s	442.4	445.9	442	445	425	435	444	455.2
Mixture ratio, -	4.43	4.80	5.5	5.1	5.5	5.5	5.0	6.0
Chamber pressure, bar	30	35	35	100	53.6	86	27	207
Area ratio, -	62.5	82.5	140	110.5	27.5	40	57	77.5
Engine total flowrate, kg/s	14.2	14.4	23.1	196.7	250	277	15.8	468
Length, m	1.71	1.91	2.7	4.0	3.38	3.38	1.78	4.24
Exit diameter, m	0.938	0.984	1.65	2.52	1.98	1.98	1.00	2.39
Combustion time, s	563	731	370	291	470	-	450	480
Weight (dry), kg	149	155	230	1300	1542	1556	132	3000
Beginning of the development phase	1973	1980	1977	1984	1960	-	1958	1972
Operational use	1979	1983	1984	1992	1966	Nonoper.	1963	1981
Use on stages	H8	H 10	H1 Second Stage	H 60	SII-SIVB	-	Centaur S IV	Space Shuttle

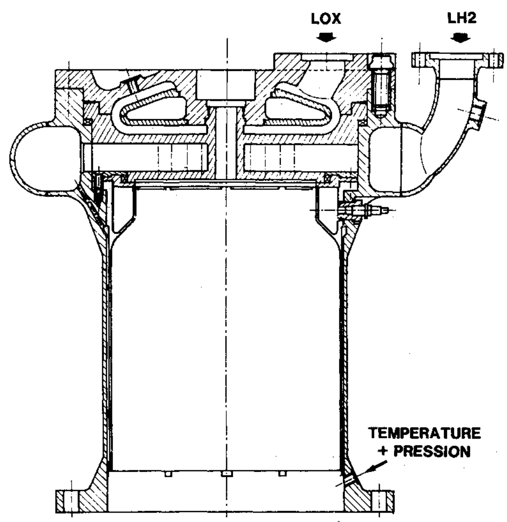


Fig. 18 HM60 gas generator.

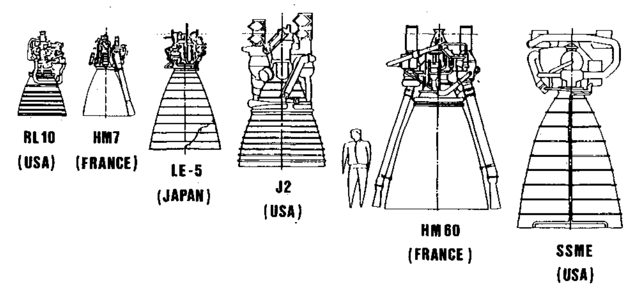


Fig. 19 Comparison of HM60 with existing cryogenic engines.

Table 6. The nozzle extension is dump cooled by 6% of the total hydrogen flow rate and is presently under study in order to select the best compromise between cost, weight, and performance.

The solutions being evaluated are brazed conical tubes, welded constant perimeter square tubes, and welded half-circle cross section tubes. For welding, automatic systems are under evaluation in order to improve on and reduce the cost of the present hand TIG (Tungsten Inert Gas) welding method used on the HM7 engine.[2]

Gas Generator

The gas generator, represented in Fig. 18, has a combustion pressure of 77 bar, a combustion temperature of 910 K, a mixture ratio of 0.9, and a total flow rate of 7.08 kg/s. The injector is composed of 120 four-to-one liquid/liquid elements having the same mixture ratio. Ignition is made from a pyrotechnic igniter situated in the center of the injector. The combustion chamber is cooled by LH2 flowing between two walls and then injected into the hot gases. Nine acoustic cavities are placed near the injector in order to damp combustion instabilities.

Accessories

The control valves and the engine gimballing systems are hydraulically actuated. One hydraulic pump driven from the LOX turbopump shaft is used for all the hydraulic power needed on the stage. The other valves on the engine are actuated from a helium pressure of 23 bar.

Conclusion

The HM60 is intended to be the main engine for European space transportation systems of the 1990's and the beginning of the next century; therefore, the choices made now concerning design and technology are fundamental for European competitiveness in space activities. A comparison of the HM60 to existing LOX/LH2 engines is given in Table 5 and Fig. 19. The cycle chosen for the HM60 engine is conservative, but Table 7 shows that the combustion pressure is

higher than for any existing gas generator cycle LOX/LH2 engine. This high combustion pressure was chosen in order to met Ariane V specified performance and engine overall dimensions and also to provide good performance if used later on a first stage.

Acknowledgments

The studies of the engine, turbopumps, and gas generator were conducted by Société Européenne de Propulsion under contract from Centre National d'Etudes Spatiales (France) in cooperation with MBB (West Germany) for the thrust chamber and VOLVO (Sweden) for the vacuum nozzle and the turbines.

References

[1]Kisselevsky, C. and Dederra, H., "Ariane Third Stage Engine System," AIAA Paper 74-1182, Oct. 1974.

[2]Brunner, W. and Buse, W., "An Advanced Tubular Nozzle Extension for Liquid Rocket Engines," AIAA Paper 77-967, July 1977.

[3]Pouliquen, M. F. and Gill, G. S., "Performance Characteristics of HM7, LOX/LH2 Rocket Engine for the Ariane Launcher," AIAA Paper 78-1036, July 1977.