ARTICLE NO. 78-1036R

Performance Characteristics of the HM7 Rocket Engine for the Ariane Launcher

Marcel Pouliquen* and Georges S. Gill†
Société Européenne de Propulsion, Vernon, France

This paper presents the main performance characteristics of the 60,000 N thrust HM7 rocket engine, designed to power the LOX/LH_2 third stage of the European launcher Ariane. The development test phase of the engine has almost been completed, and qualification testing is scheduled before the end of 1979. During development testing under altitude-simulated conditions, a mean specific impulse of 440.8 s was demonstrated compared to a specification value of 430 s. The first vacuum tests resulted in ignition delays of about 0.5 s and produced hard starts. Reducing the helium purge on the oxidizer side of the injector from 34 g/s to 8 g/s and simultaneously increasing the opening rate of the main LOX valve reduced the ignition delay to 0.13 s and eliminated the hard starts. The ability of the engine to meet the mixture ratio accuracy specification of ±1% has been demonstrated.

Introduction

THE 60,000 N thrust HM7 rocket engine was designed to power the third stage of the European launcher Ariane, the first flight of which is scheduled for November 1979. The qualification tests of the thrust chamber assembly have been successfully completed. Development tests of the engine have been accomplished on four sea-level engines (MS1, MS2, MS3, and MS4) and three flight-type engines (MV1, MV2, and MV3) under altitude-simulated conditions. Final developmental tests are scheduled for completion on MV3 in June 1979.

The main performance characteristics of the engine have been demonstrated during these development tests. The compatibility of the engine with the Ariane third stage has been verified during 24 propulsion bay tests and 8 propulsion system (propulsion bay + flight tanks) tests.

Engine Main Features

Since the detailed features of the HM7 engine have been presented in previous papers,[1-3] only the main characteristics are indicated in Table 1. The general arrangement of the engine is shown in Fig. 1 and the flow diagram in Fig. 2.

Table 1 HM7 main features

Thrust (vacuum)	60,000 N
Propellants	LOX/LH_2
Mixture ratio	4.5:1
Specific impulse	
At 30 bar chamber pressure	440.8 s
At 35 bar chamber pressure	443.0 s
Dry weight	150 kg
Dimensions	
Length from gimbal	1713 mm
Diameter at nozzle exit	938 mm
Flight burn duration	570 s

Presented as Paper 78-1036 at the AIAA/SAE 14th Joint Propulsion Conference, Las Vegas, Nev., July 25-27, 1978; submitted Oct. 5, 1978; revision received May 18, 1979.

Index category: Liquid Rocket Engines and Missile Systems.

*Ariane Third Stage Technical Manager.
†Technical Consultant.

Specific Impulse

At the beginning of the Ariane program, the specified values for altitude specific impulse were 434 s for the thrust chamber and 430.5 s for the integrated engine. The loss assumed for the turbine flow was 3.2 s with the gas generator cycle shown in Fig. 2. The specific impulse of the thrust chamber assembly was determined on the horizontal altitude simulation test stand[4] and that of the integrated engine in vacuum tests on a vertical altitude simulation facility.[5]

Thrust Chamber Performance

Specific impulse of the HM7 thrust chamber assembly has been improved in three phases:

Preliminary Development Program

A standard coaxial injector element was used in conjunction with a prototype dump cooled nozzle extension. The mean specific impulse value demonstrated on three injectors was 438 s, and the minimum value obtained was 437 s.

Peformance Improvement Program

The thrust chamber assembly was modified slightly to improve performance. The three modifications described below were incorporated into the final production design, resulting in a thrust chamber assembly vacuum specific impulse of 444 s.

1) Improved injector element: Element-controlled propellant mixing rate, and thus element-controlled performance, was increased by utilizing the concentric tube ϕ mixing rate correlation.[6] This correlation predicts minimum element-controlled mixing rate when the momentum head ratio between element hydrogen annulus gas and the central liquid oxygen jet is equal to 1:1. Variations in the standard element (such as swirlers, tapered LOX posts, increased ΔP, etc.) had not produced any measurable change in performance. The ϕ correlation analysis indicated that the standard element was operating in a region in which any performance gain produced by improved mixing was nullified by a simultaneous decrease in oxygen atomization. This resulted because the standard element had a propellant momentum head ratio of slightly less than 1:1.

The element was redesigned (within existing element geometrical restraints) by means of the ϕ correlation technique to produce a momentum head ratio much greater than 1:1, while simultaneously improving the oxygen atomization characteristics. This element type was called "high-shear." The use of the high-shear element with an oxygen swirler produced a specific impulse gain of 4.5 s.

Details of the element improvement program and the use of the ϕ mixing rate correlation are contained in Ref. 6.

2) Reduction in nozzle extension hydrogen dump coolant flowrate: It was experimentally determined that the hydrogen dump coolant flowrate could be safely reduced from the design value of 150 g/s to 126 g/s.

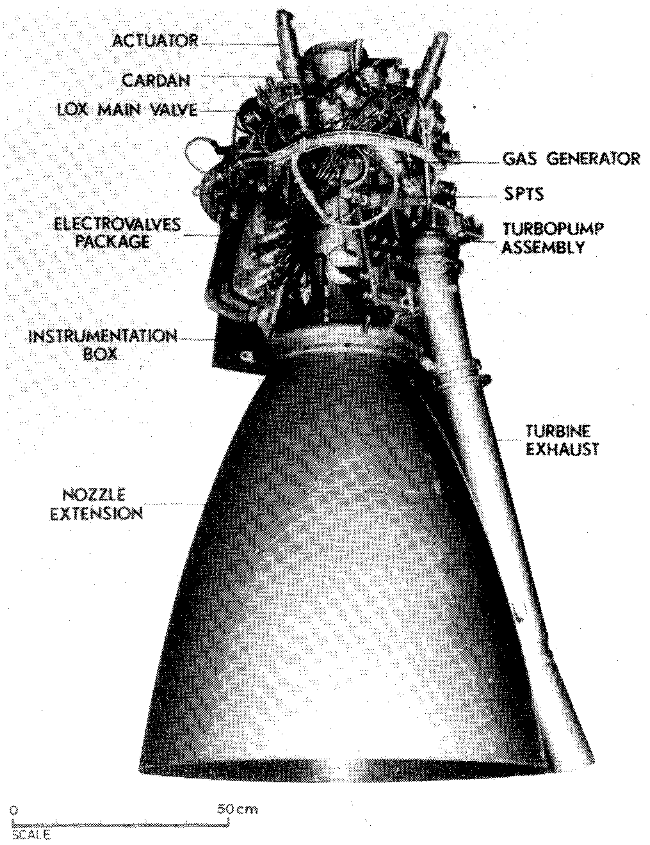

Fig. 1 HM7 rocket engine.

3) Increase in hydrogen dump nozzle area expansion ratio: There are 484 very small nozzles at the end of the hydrogen dump cooled nozzle extension, which originally had an area expansion ratio of 1:1. These nozzles were redesigned to provide an area expansion ratio of 5.7:1. The combination of decreasing the hydrogen dump coolant flowrate and increasing the dump coolant nozzle area expansion ratio increased the specific impuse about 1.5 s.

Increased Nozzle Area Expansion Ratio

A new combustion chamber operating at the same thrust level but at a chamber pressure increased from 30 to 35 bar and with a smaller throat diameter is used. This resulted in a geometric nozzle area ratio increase from 63.5:1 to 74.1:1. The experimental performance gain was 2.2 s. Thus the maximum specific impulse demonstrated in the thrust chamber altitude simulation facility was 446.2 s. The standard deviation of this facility is 2.1 s. All performance values are for propellants corresponding to MIL specs and for nominal mixture ratio.

The main tests performed on the thrust chamber to evaluate performance are summarized in Table 2.

HM7 Engine Specific Impulse

Due to the gas generator cycle of the engine, the engine specific impulse is lower than the thrust chamber assembly specific impulse. The gas generator gases used to drive the turbine exit through a 6:1 area ratio nozzle, with a calculated specific impulse of 255 s. The effective loss in performance is calculated at 3.2 s.

Because specific impulse depends essentially upon thrust and flowrate measurements, a detailed analysis has been made to evaluate and improve the accuracy of these parameters.[5]

Propellant Flowrates

Two turbine flowmeters are used in each main propellant feedline. At the present time, flowrate measurement accuracy has been completely evaluated, and improvements presented in Ref. 5 have been applied. The accuracy of propellants mass flowrates measurements is now estimated at 0.5% for LOX and 0.9% for LH_2.

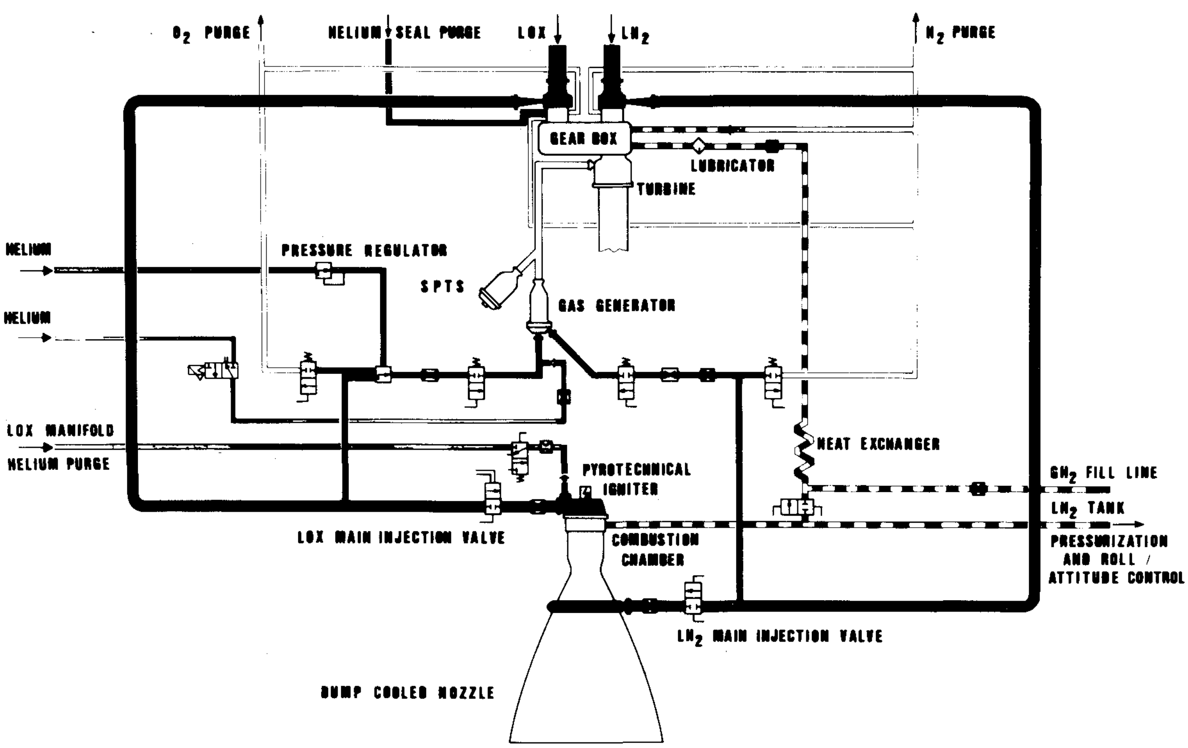

Fig. 2 HM7 engine flow diagram.

Table 2 Thrust chamber altitude simulation tests

Phase	Injector type	Combustion pressure, bar	Nozzle extension	Number of tests/Cumulated duration	Specific impulse, s
I	Standard	30	Standard	83/3137	438
II	High shear LOX + swirlers	30	Dump cooled flow nozzles	46/2226	444.0
III	High shear LOX + swirlers	35	Dump cooled flow nozzles	24/942	446.2

Thrust Measurement

The measurement of integrated engine thrust is a difficult problem compared to thrust chamber assembly thrust measurement because of 1) the large number of fluid mechanic couplings between engine and facility, and 2) the turbine exhaust, which creates an asymmetric thrust and which (in the case of HM7) has a separate hot-water altitude simulation system.

The altitude simulation tests on the MV1 engine were made to complete facility acceptance and to study engine transients. The major results on flow measurement and thrust were obtained on MV2, and these results are now being improved during MV3 tests.

During the first three tests with the MV2 engine, hydrogen leakage occurred during the low temperature encountered during engine cooldown, and the altitude facility failed to start. After this leakage was eliminated, the altitude system started, a reduced pressure (30 mbar) was obtained in the altitude cell, and a specific impulse of 441 s was obtained.

Before MV3 tests, improvements were made on: 1) propellant inlet lines (to reduce mechanical friction), 2) thrust management load cell, and 3) gauging procedure, by making calibrated thrust steps immediately before and after the engine firing (engine cooled and vacuum set).

The main results obtained during MV2 engine testing are presented in Table 3.

Mixture Ratio

To fulfill Ariane third-stage specifications, the overall mixture ratio of the HM7 engine must be controlled within ±1% accuracy. This will be done during operational engine production by two calibrating engine tests (50 s duration each) followed by a 100 s acceptance test. These tests will be performed under sea-level or altitude-simulated conditions.

At the start of the HM7 development program, mixture ratio variations from the target value occurred, both between engines and between tests with the same engine. The main causes of this variation were identified as follows:

1) A small, erratic leakage around the cavitating venturis feeding the gas generator. This problem was solved by a slight hardware modification.[3]

2) The design and the fabrication quality of the main calibrating orifices. The design has been altered, and improved fabrication techniques have been introduced to obtain a more reproducible orifice geometry (dimensions, sharp edges, smooth surface) and thus a more reproducible orifice flow coefficient.

Several tests have been performed on two sea-level engines, with the result that the variation in mixture ratio due to calibrating orifice flow coefficient variation appears to be small compared to the flowrate measurement accuracy. These results have also been verified during MV2 and MV3 altitude simulation tests, and the 1% accuracy has been demonstrated.

Table 3 MV2 engine altitude simulation tests

Test number	Firing duration, s	Nozzle extension	Test objectives	Result
MV2-01	3.0	Ablative	3 s engine calibrating	H_2 leakage made altitude simulation fail to start
MV2-02	2.69	Flight type	3 s engine calibrating	
MV2-03	3.0	Ablative	3 s engine calibrating	
MV2-04	3.0	Ablative	3 s engine calibrating	Good starting of altitude facility + engine calibrating
MV2-05	3.0	Ablative	3 s engine calibrating	
MV2-06	50	Flight type	Performance	Specific impulse measured = 441 s
MV2-07	498	Flight type	Performance	

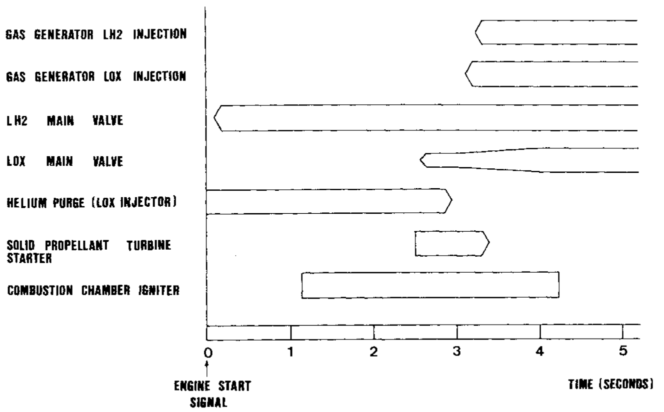

Fig. 3 Initial starting sequence.

Engine Ignition and Starting in Vacuum

While no problem was encountered during engine starting and ignition under sea-level conditions, the first altitude simulation tests on MV1 in mid-1977 resulted in very hard starts. For these initial vacuum tests, the same sequence as for sea-level tests was used. The starting sequence (Fig. 3) includes 1) precooling of combustion chamber for 2.5 s, 2) simultaneous solid-propellant turbine starter ignition and main LOX valve opening signal, and 3) LOX and LH_2 gas generator valve opening signals. The main events and results obtained during MV1 engine testing are given in Table 4.

Theoretical Considerations

The main parameters having an influence on main combustion chamber ignition delay may be assumed to be: combustion chamber pressure level before ignition, local

Table 4 MV1 engine tests

Test number	Duration, s	Objectives	Results
01	...	Test of altitude facility with combustion chamber precooling	0.1 bar in altitude cell during engine precooling
02-1	...	Engine calibration/starting (3 s)	Starting inhibited by measurement error
02-2	0.45	Engine calibration/starting (3 s)	Safety system automatic shutdown (hard start)
03	1.72	Engine ignition test (3 s)	Safety system shutdown, engine regulation shifting
04	1.17	Engine ignition test (3 s)	0.5 s ignition delay, nozzle extension destroyed
05	0.50	Engine ignition test (0.5 s)	0.30 s ignition delay
06	0.50	Tank head ignition test (0.5 s)	No ignition
07	0.50	Ignition test (0.5 s) igniter with two exhausts at 30 deg angle	0.20 s ignition delay
08	0.50	Igniter with one exhaust at 45 deg angle	0.22s ignition delay
09	0.50	45 deg angle igniter, fast opening LOX injection valve	0.22 s ignition delay
10	0.30	Standard igniter, reduction of helium purge flowrate	0.185 s ignition delay
11	0.30	New helium flowrate reduction	0.16 s ignition delay
12	3.00	Engine starting/ignition (3 s)	0.17 s ignition delay
13	20		0.17 s ignition delay
14	120	Engine starting/nominal conditions	0.15 s ignition delay
15	570	Nozzle extension life duration	0.13 s ignition delay

mixture ratio near the igniter, combustion chamber and injector element geometry, igniter energy, injection temperature of propellants, and helium purge flow rate.

Generally, it is assumed in the literature that a flammability limit curve can be drawn when pressure is represented as a function of mixture ratio. In the upper area above the flammability limit curve, combustion is supposed to occur when sufficient igniter thermal energy is transmitted to the mixture, while in the lower area under the curve, combustion is not obtained even if very high thermal energy is transmitted to the mixture. In reality, combustion phenomena are not so simple, and the limit curve location is not so clear, particularly in a combustion chamber in which there are significant variations of local mixture ratio, propellant temperature, and ignition source energy. The shape of the flammability limit curve, however, can be used to predict a trend—when pressure and mixture ratio are increased, ignition delay is reduced. This is observed in Fig. 4 where MV1 test results are plotted.

In Fig. 4, the experimental points on the left correspond to high ignition delay times and the points on the right to low ignition delay. The mixture ratio indicated on this figure has been calculated at the pump inlets so that it is not the true absolute mixture ratio in the combustion chamber. The intent is only to compare mixture ratios of the different tests. The curve indicated is not an absolute flammability limit but is presented to indicate the shape of "constant ignition delay curves."

The combustion pressure in the chamber during starting for MV1 tests is presented in Fig. 5. It can be noticed that, during the LH_2 precooling phase (before turbine starting) the combustion pressure is not high enough to induce a choked nozzle. Chamber pressure is only 0.12 to 0.20 bars. Immediately after turbine startup, however, the chamber nozzle is choked so that ignition conditions are representative of what will occur in flight.

During the tests MV1-01 to MV1-04 (see Table 4) a development injector configuration (without the final element swirlers) was used. The starting sequence developed on sea-level tests (Fig. 3) was used on the MV1-02 altitude test, resulting in a high ignition delay and a hard start. For MV1-03 and MV1-04 tests, the opening of the main LOX valve was delayed in order to increase the chamber pressure (induced by the increased hydrogen flowrate) at ignition (see Fig. 4). Tests results revealed that ignition delay was not improved by this change, since mixture ratio was also reduced at LOX injection time. The severe hard start which occurred during MV1-04 destroyed the chamber nozzle extension. In order to develop a satisfactory starting sequence: 1) plastic reinforced battleship nozzle extension was used in place of the dump nozzle extension; and 2) the developmental injector elements were replaced with the injector elements using LOX swirlers and high shear hydrogen flow (in order to solve the ignition problems with the final injector configuration).

Short "ignition only" tests were made, with the duration of LOX injection flow limited to 0.5 s (and later 0.3 s) in order to

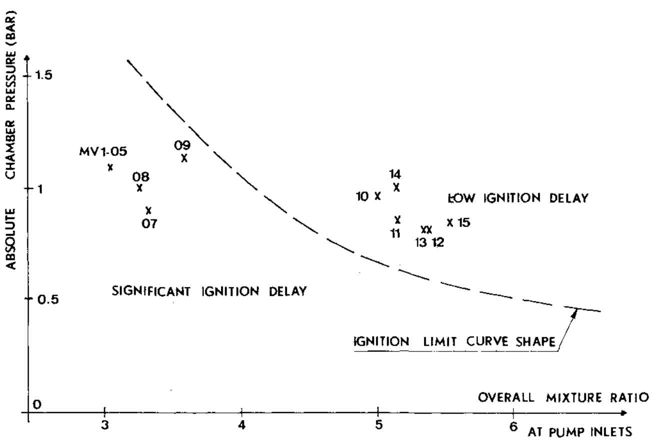

Fig. 4 Ignition characteristics.

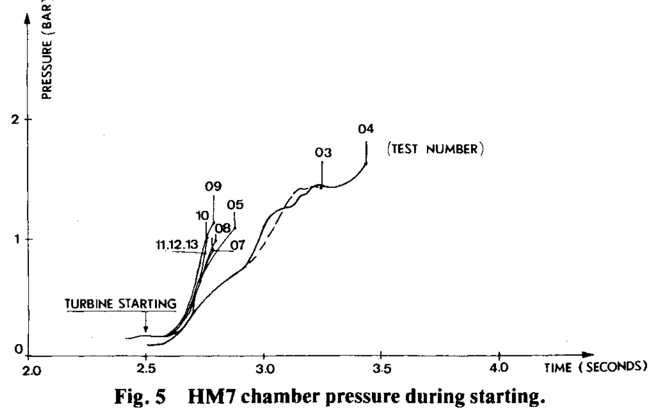

Fig. 5 HM7 chamber pressure during starting.

prevent a large accumulation of mixed LOX/LH_2 in the facility before ignition. The solid propellant turbine starter was used, but propellants were not allowed to enter the gas generator.

A special test was conducted (MV1-06) to determine the feasibility of combustion chamber ignition under "tank head" conditions. No ignition occurred because of the very low chamber pressure. During these vacuum ignition tests, all the six parameters (previously listed) presumed to have a significant influence on ignition delays were studied.

Combustion Chamber Pressure before Ignition

No ignition had been obtained on the HM7 at less than 0.8 bar in the mixture ratio range tested. The conclusion was that to reduce ignition delay (i.e., time between start of LOX injection and start of chamber pressure increase) it is better to delay LOX injection but, due to LH_2 flowrate increase, to have a fast oxygen valve opening rate in order to maintain (or increase) mixture ratio. Thus, for MV1-10 to MV1-15 the preliminary (partially open) LOX valve step was suppressed.

Local Mixture Ratio Near the Igniter

Ignition can occur when sufficient thermal energy is transmitted (by hot gas or particles) to a location in the combustion chamber where the oxygen-hydrogen mixture is within the flammability limits. The first ring of injector elements surrounding the igniter are the first elements to be ignited, and every mixture ratio value can be found locally in each element. This explains why for tests 07, 08, and 09 the igniter hot jet (one or two jets) was directed toward one or two elements. This resulted in an ignition delay reduction to 0.2 s, but the necessity of modifying the igniter made this solution uninviting. Cold flow tests made with water showed that, during starting, LOX flow distribution is not uniform because the LOX inlet is not at the center of the LOX injector manifold.

Combustion Chamber and Injector Element Geometry

No significant ignition delay difference was found between development nonswirler elements and final high-shear elements with swirlers, even though the swirler elements should direct some oxygen toward the hot gas igniter for the center ring elements. The angle between the element centerline and the igniter jet, however, had a significant influence (see tests 07, 08, and 09).

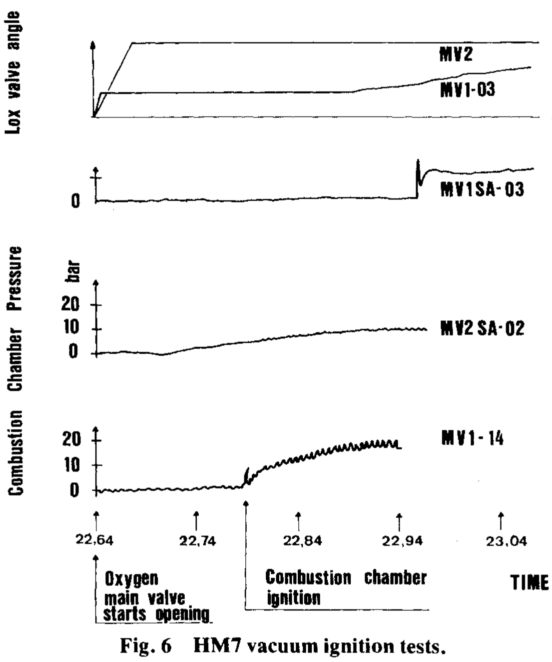

Fig. 6 HM7 vacuum ignition tests.

Igniter Energy

This is an important parameter, which must be related to the flowrate of the injector elements near the igniter. As explained previously, to obtain a low ignition delay, the chamber pressure must be as high as possible. The pressure level in the chamber before ignition is a function of 1) the hydrogen flowrate, or 2) the igniter energy, which can be used to heat the hydrogen and thus to increase the chamber pressure.

If the second method is used to increase chamber pressure, the flowrate of the igniter must be very large (some significant percent of the nominal flowrate of the engine), or a high flowrate preburner "torch" must be used. For the HM7, the first method has been selected, that is, chamber pressure is increased by the increasing hydrogen flowrate during the turbopump starting. Oxygen is injected when sufficient pressure exists in the chamber. Ignition of a single element is then sufficient to initiate complete chamber ignition. The HM7 solid propellant igniter flowrate (14 g/s) was chosen high enough (based on previous results with the HM4 motor) to ensure chamber ignition when a satisfactory pressure-mixture ratio is reached. Increasing igniter flowrate by 50 or 100% would have very little effect on chamber ignition characteristics.

Propellant Injection Temperature

No specific tests were performed to vary propellant injection temperature. The possibility of reducing LH_2 combustion chamber precooling (which would increase hydrogen injection temperature and reduce mass flowrate) was considered. The major problem is that the reduction in mass flowrate would also reduce the chamber pressure during the ignition phase.

Helium Purge Flowrate

During engine start, it is necessary to purge the LOX injector to prevent precooling hydrogen from entering the LOX injection cavity. When the LOX valve is opened, the helium flow has two effects:

1) The fluid injected into the chamber is a mixture of helium and oxygen.

2) The LOX mass flowrate is reduced, resulting in a reduced mixture ratio. The effect of helium purge on injector flow conditions was studied during cold flow tests. During hot firing tests, reduction of helium purge flowrate from 34 g/s on MV1-02 to 8 g/s on the last MV1 tests reduced ignition delay from 0.2 to 0.13 s.

In conclusion, the most important HM7 ignition parameters were found to be chamber pressure, helium LOX purge mass flowrate, and chamber mixture ratio. These conclusions were confirmed by MV2 tests (Table 3), where the slightly different characteristics of the LOX valve improved the HM7 ignition (Fig. 6).

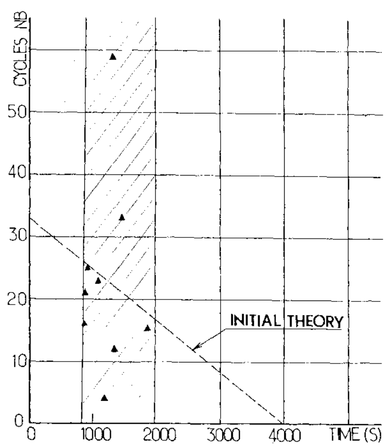

Fig. 7 HM7 thrust chamber life (results).

Table 5 LOX/LH$_2$ engine performances

	ASE	SSME	J2S	RL10 A3-3	HM7@ 30 bar	HM7@ 35 bar
Thrust, kN	89	2175	1064	69	60	60
Chamber pressure, bar	138	205	54	27	30	35
Mixture ratio	6	6	5.5	5.0	4.6	4.6
Area ratio geom	400	77.5	27.5	57	63.5	74.1
Theoret. performance, s	487.5	466	448.2	461.7	463.4	465.7
Engine performance, s	476	455.2	428.8	444	440.8	443.2

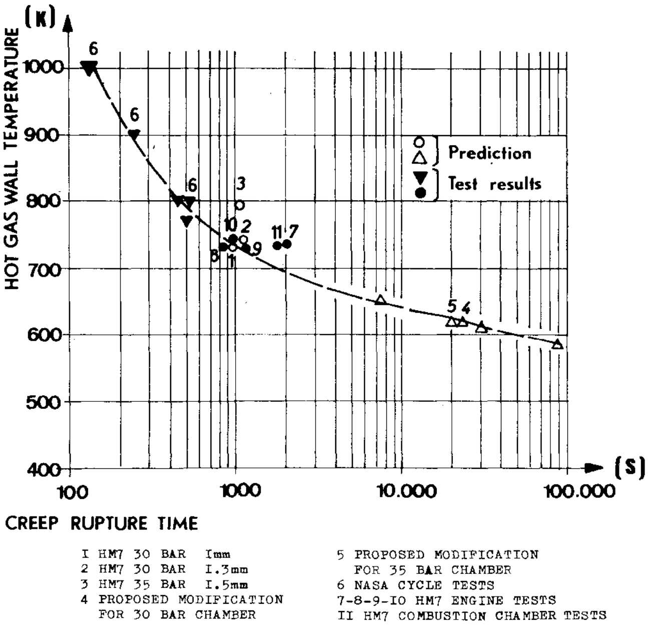

Fig. 8 Thrust chamber life as function of hot gas wall temperature.

Combustion Chamber Life Duration

The minimum life duration necessary for the HM7 thrust chamber is 770 s in four hot firing tests: flight (570 s), engine acceptance test (100 s), and two calibration tests (50 s each). At the beginning of the program, it was expected that no chamber durability problem would occur before 2000 s and 11 tests. The first engine long-duration tests showed, however, that the combustion chamber thermal life was limited to a lower value, due to cracks appearing near the throat. The life calculations for the OFHC copper combustion chamber were based primarily on low cycle fatigue damage to which a creep damage estimation was added. In fact, the published data about stress/strain and creep characteristics of copper as a function of cycle characteristics, number of cycles, loading velocity, and temperature are incomplete and no really satisfactory model exists for a combined cycling and creep process.

Surprisingly, mean values of 1300 s life duration were obtained at the beginning of the thrust chamber and engine development (Fig. 7) which were independant of the number of cycles.

In an attempt to increase chamber life duration and to analyze the phenomenon, several tests and hardware investigations were made:

1) The hot gas wall temperature was measured. The experimentally measured value was 70°C higher than predicted by the heat transfer calculations.

2) The copper wall thickness was increased from 1.0 to 1.3 mm, with no improvement in life.

3) A 0.3 mm nickel layer was electroplated on the inside of the copper chamber. Short duration tests on the thrust chamber have shown that the nickel adhered well to the OFHC copper. The long duration tests scheduled for the engine to evaluate the nickel layer have not yet been made.

From the HM7 experimental program and theoretical studies concerning OFHC copper thrust chamber life duration, the main conclusions that can be stated are: 1) creep/rupture is the life governing mechanism, 2) the crack appears first on the hot gas side of the wall close to the throat, and 3) life duration of OFHC copper is very sensitive to hot gas side wall temperature. This is summarized in Fig. 8.

Based on these results, a design modification has been proposed for the future 35 bar combustion chamber pressure motor, for which the hot gas side wall temperature is reduced. For the thrust chamber presently used for the Ariane Launcher, the present design offers adequate life duration. In addition to these tests, it should be noted that an HM7-type chamber in which aluminum alloy was used instead of OFHC copper has been designed and tested. During these tests, the feasibility of this lightweight combustion chamber was demonstrated.

Conclusion

The HM7 LOX/LH$_2$ rocket engine developed for the low-cost Ariane launcher has demonstrated very good performance during development. Table 5 presents a comparison between the known cryogenic engines.

Acknowledgments

Messerschmitt-Bölkow-Blohm (MBB) of West Germany is responsible for the design and development of the thrust chamber and injector of the HM7 engine, and conducted tests at its Ottobrun facility. Société Européenne de Propulsion (SEP) of France is responsible for the complete engine design and development, and conducted tests at both its Villaroche and Vernon facilities.

References

[1]Kisselevsky, C. and Dederra, H., "ARIANE Third Stage Engine System," AIAA Paper 74-1182, AIAA/SAE 10th Joint Propulsion Conference, 1974.

[2]Dardare, J., "HM7 Oxygen/Hydrogen Rocket Engine Design, Operation and Development Status," *Proceeding of the 11th International Symposium on Space Technology and Science,* Tokyo, 1975.

[3]Pouliquen, M.F., "Development Progress of HM7 LOX/LH$_2$ Rocket Engine for the ARIANE Third Stage Propulsion System," Paper 77-91, IAF XVIII Congress, Praha, Czechoslovakia, 1977.

[4]Dederra, H. and Kirner, E., "Test Results with a Regeneratively Cooled LOX/LH$_2$ Thrust Chamber for the ARIANE Third Stage Propulsion System," AIAA Paper 75-1189, AIAA/SAE 11th Joint Propulsion Conference, 1975.

[5]Pouliquen, M.F. and Gill, G.S., "Altitude Simulation Testing of the HM7/LH$_2$ Rocket Engine for the ARIANE Launcher," AIAA Paper 78-968, AIAA/SAE 14th Joint Propulsion Conference, 1978.

[6]Gill, G.S., "A Qualitative Technique for Concentric Tube Element Optimization Utilizing the Factor (Dynamic Head Ratio-1)," AIAA Paper 78-76, 16th Aerospace Science Meeting, Huntsville, Ala., 1978.