Finishing Technology Method for the Lightweight Reaction Motor NERVA

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NERVA research is a continuation of a limited experimental liquid rocket motors program (**MRE** in Romanian) started by the authors in 1962 at the Polytechnic Institute in Bucharest, Romania, with the encouragement of the famous pioneer of Astronautics Prof. Hermann Oberth, a Romanian native. The **MRE** program was part of a larger ADDA project of developing sounding rocket vehicles, first aimed to develop a series of small thrust, liquid propellant rocket engines and their remotely controlled test stand, as early Romanian academic preoccupations in space propulsion. The successful first test firing of the MRE-1 motor was recorded on April 9th, 1969. This early endeavor was 100% successful and was continued long after with a series of engine developments including the test firing of the first Romanian air-breathing rocket engine in 1987 and the present development of the NERVA liquid motor for a Mach-10 re-entry research vehicle. The design principles and proprietary manufacturing technologies developed by ADDA are demonstrated with respect to the double wall, cooling jacket of the engine thrust chamber and nozzle. A cooperative Romanian-Greek (Patras University) research project under a NATO sponsorship is now envisaged to solve this problem. Emphasize is placed on the surface and channel finishing procedure for enhancing the fatigue behavior of the double wall structure in regard to the medium-pressure level envisaged in the NERVA powering unit. Cooperation with US *SpaceX* is also expected.

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Romanian state-of-the-art in MRE

From early 1910, when the Romanian Hermann Oberth, by then a high school boy, began to practice his first ideas on rocket propulsion, and up to 1962 after the WW-2, a long eclipse in this field turned up. Beginning that year 1962 a team of young students, led by the author, initiated a limited research program in liquid rocket propulsion[1] at University "Politehnica" in Bucharest (**UPB**). The team began by ignoring the restrictions imposed after the war through the Paris peace treaty to the aeronautical industry in Romania, once a remarkably well developed aircraft producer. Conversely, the proprietary technology program A.D.D.A. (from "*Association Dedicated to the Development in Astronautics*") consisted in designing and building a low thrust (200-N) experimental liquid propellant rocket motor and the accompanying test bed for hot, static investigations. The intention was to prove the feasibility of such engines as scale models for a future 10 times larger motor (later ML-26) to equip a liquid propellant sounding rocket vehicle and the result was a success, experimentally proved by the first test firing in 1969.

The study program was also backed by adequate numerical prediction of propellant performances through equilibrium thermochemical computerized calculations of the deliverable specific impulse. A large series of propellant combinations of practical interest was investigated and a surprisingly high fit with the experiments was recorded after the method of "frizzing temperatures" was introduced[2]. The NERVA rocket motor is under current development using the previous experience of the research team at **UPB**.

Earliest MRE concept

At the level of the year 1960 any experimental knowledge of the problems of rocket propulsion in Romania was lacking, despite the considerable amount of international literature on the matter and the existing soviet rocket armament on the national territory. To answer the numerous challenges, natural precautions were engaged, specifically to minimize the risks of the preliminary research, as they were engaged by all the teams of forerunners in the risky field of rocket technology. As a result the following, general working parameters and conditions for the engine and the test stand were chosen[3]:

1. Ground level thrust F=200-N;
2. Chamber pressure level p_c=7-bar abs.;
3. Exit pressure level p_e=1-bar;
4. Longest allowable burn t_a=1-min;
5. Non-cryogenic liquid propellant;
6. Low level propellant toxicity;
7. Non-hypergolic di-propellant;
8. Third-party hypergolic ignition;
9. Remotely-controlled firing (70-m);
10. All-portable energy/water supply.

These general conditions were imposed due to the compulsory safety requirements, prevailing over all other technological aspects of the project. Low combustion pressure was considered a good precaution that could not affect the conclusions related to the efficiency of combustion, where the remaining criteria were considered for study and enhancement.

At the same time a low combustion level leads to a larger volume of the combustion chamber and to larger dimensions of the injector nozzles, easing the manufacturing technology of both. The above mentioned parameters were found to well met both costs and hazards which are conveniently reduced, provided the following design specifications are followed:

1. White-fuming nitric acid (96%) as oxidizer at α=0.62 excess ratio (markedly low value);
2. Romanian kerosene STAS 5639/57 jet-engine as the main fuel (accessibility considerations);
3. "*Aerobee*" 75% aniline+25% furfurol as starting, hypergolic fuel (apparently well proved);
4. Voluminal gear pump, controllable propellant supply system (to avoid POGO instability);
5. Low pressure, external water cooling of the entire motor (high cooling capacity);
6. Water purge of proximity devices after each firing (to reduce chemical contamination);
7. Lead, car-type, 24V d. c. electric power supply (accessibility/costs considerations).

A low-pressure, voluminal gear-pump, dyergolic main propellant feed system was selected to accomplish the safety requirements. The following main issues were targeted as research objectives of the preliminary program:

a) Study of the injector head performance and related thrust chamber optimal length and volume;

b) Study of ignition delays and transients in a triple-component, hypergolic start fuel system;

c) Extensive testing of nitric acid and nitrogen tetroxide based propellants;

d) Real cooling system confrontation with the computational model, previously run on computer;

e) Behavior of the overall rocket engine system and its controllability;

f) Overall efficiency of the **MRE** as compared to the numerical thermochemistry simulations;

g) Materials behavior and technology enhancements;

h) Reliability of the liquid propellant rocket propulsion system, in parts and as a whole;

To acquire the largest possible amount of data during the experiments, an automated monitoring system was introduced. Nevertheless no recording capacity was considered and the problem of data storage was solved by a continuous filming of the control panel. Again to fairly reduce costs, the following main set of engine's parameters was accordingly chosen for being measured, displayed and recorded on the control panel:

1) Turning rate of pumps shafts for oxidizer and fuel;
2) Main propellant delivery pressure at exit of pumps;
3) Main propellant flow rate of components at engine entrance
4) Feed pressure for the auxiliary starting fuel
5) Unsteady axial gas-dynamic thrust;
6) Pressure level in the thrust chamber;
7) Ignition delay;
8) Cooling water temperature.

The main constraint, voluntarily assumed in the early design, consisted in decidedly employing existing Romanian commercial equipment mainly in the test stand systems. It was the intention to demonstrate in that manner the capability of rocket engine manufacturing in the frames of existing Romanian industrial infrastructure of the years 1960. Thus far a minimal number of devices remained to be specifically designed and manufactured by the research team.

Nevertheless, although minimal, this number of transducer and control devices finally proved to be a considerable challenge for the research team: very little existing components were found and most of the tasks in designing new apparata were carried out starting from zero.

The first MRE-1B engine

As shown in the nearby Fig. 1, the experimental rocket motor is designed as a cylindrical, thick self-sustained double wall construction with a flat injector head and a smooth de Laval short and low expansion ratio nozzle. The gas-dynamic profile was designed according to equilibrium, steady flow thermo-chemical computations of the combustion and expansion of gas-phase products, performed by the author on existing IBM-360-30 machines. Details are given in reference list.

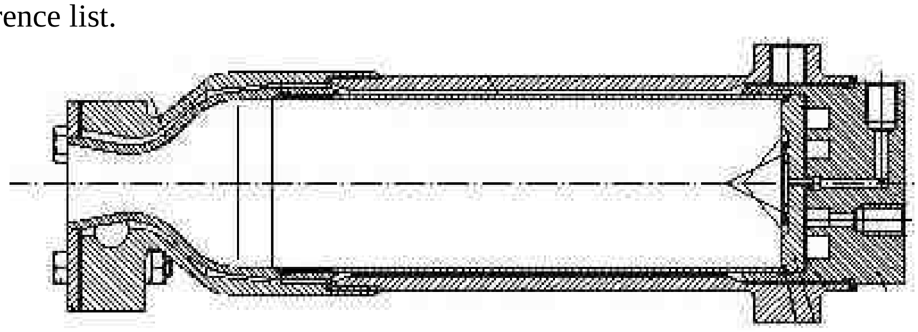

Fig.1. Unbound, double wall configuration of **MRE-1** *(1962)*

The cooling channel presents an annular shape with small, almost constant width and fairly unsupported construction, due to the very high self-stiffness of the motor inner wall. The cooling water flows between the two envelopes at the current, 2 bar over-pressure of the public water feed lining.

As a result, the outer wall is discharged from any important pressure load and consequently it is manufactured from an aluminum type, anti-corrosion, low strength alloy. The cooling water enters the nozzle exit side, sweeps the inner wall along the engine and exits the side of the injector head, where the temperature is measured by a thermocouple device. This gives the simplest information about the global heat transfer along the engine during the hot runs. The computed temperatures of the inner and outer sides of the motor wall are given in Fig. 2.

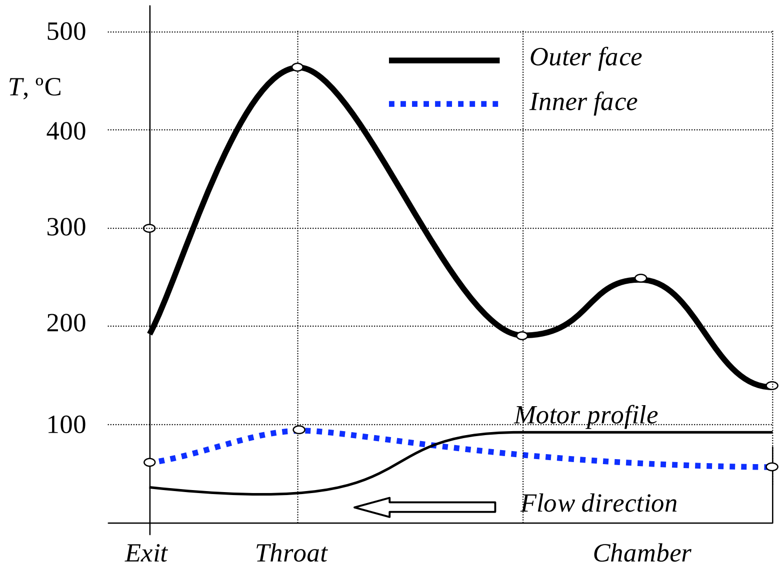

Fig.2. Wall temperature profile

Two temperature maxima are observed: the throat common maximum is of 460 ºC, while the neighboring one of 240 ºC is induced by the fixture disposal (screw) between the nozzle and the thrust chamber, where the wall is doubly thick. Thrust chamber wall behaves at 135ºC only. The maximal temperature gradient, registered in the throat area of the nozzle, equals 300 degrees only, meaning a really low temperature stress.

The small aria ratio of the exhaust nozzle is a direct consequence of the very small pressure ratio of the engine, as chosen, for safety reasons, by the designer (7:1). Special attention was devoted to the heat transfer simulations of the entire engine, worsened by the inner wall of visibly over-dimensioned thickness, result of the imposed safety conditions.

As a further precaution, a special chromium-nickel, non-ferrous alloy *INOXITERM-5* was manufactured at the “Malaxa” works in Bucharest and used to build the nozzle of MRE-1B (Fig. 3). The alloy is given in table 1 and manifests high anticorrosive and temperature resistance qualities, although a little smaller heat transfer coefficient as compared with the common steels.

Table 1. **Soft INOXITERM-5 alloy**

Specification	*Value*	*Units*
Carbon	<0.15	%
Chromium	24.00	%
Nickel	21.00	%
Iron	16.57	%
Manganese	0.80	%
Silicon	2.00	%
Flow limit	34.0	daN/sqmm
Rupture stress	55.0	daN/sqmm
Rupture elongation	12.0	%
Hardness	65.0	HB

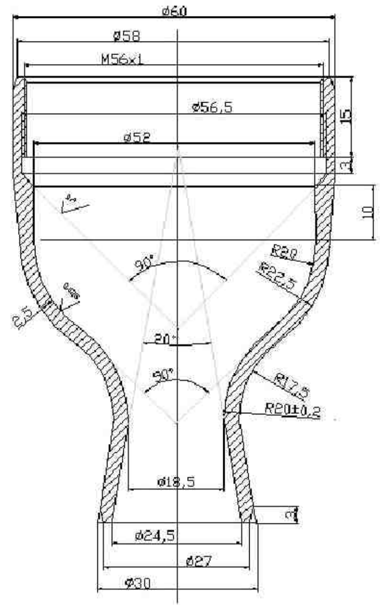

Fig.3. Thrust nozzle of **MRE-1** *engine.*

The injector head is a circular flat plate provided with two concentric rows of injectors, due to the very small mass flow rate of the engine of 127 grams/second only (Table 2). Each injector circle contains four equally spaced, inclined impinging injectors of constant diameter

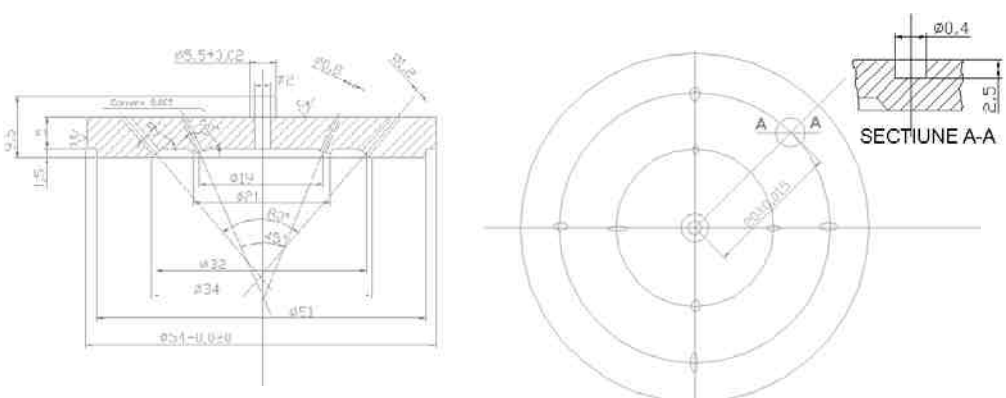

Fig.4. Injector head

The thrust injectors of 0.8 and 1.4 mm diameter for the fuel and oxidizer respectively (2-bar overpressure) are crossing their jets in four pairs, as shown in Fig. 4. This solution is a compromise, which serves to spread the components into the chamber and to protect the walls at the same time.

The ninth, separate small injector is provided for the admission of the starting fuel through a lateral, auxiliary access line, as shown further in the block diagram of the propellant system.

The central hole in the injector head provides the means for measuring the stagnation pressure in the thrust chamber by an isolated pressure transducer. The material for the injector head is the same *INOXITERM-5*.

The two main components of the propellant access the injectors through a double distributor, on which the plate with the injector is pressed by the chamber and simultaneously sealed by a ceramic fiber (asbestos) seal.

The geometry of the rocket engine resulted from thermo-chemical computations of the steady combustion. The following main parameters of the engine were thus delivered:

MRE-1B characteristics

Table 2.

Parameter (condition)	*Value*	*Units*
$T_{chamber}$	2483	ºC
T_{exit}	1528	ºC
w_{exit} (computed)	1633	m/s
D_{crit}	18.49	mm
D_{exit}	24.44	mm
Thrust (ground level)	200	N
Flow-rate	127	g/s
Oxidizer/Fuel ratio	3.5	-

Two design solutions were adopted to allow the required modification of the thrust chamber length (and consequently volume) be made, in order to study the influence of this parameter upon the combustion efficiency, this point standing as a main goal of the research program.

First, the simplest solution consisting of an interchangeable set of thrust chambers and chamber-cooling jackets of different length was put into practice. Obviously these parts had to be changed prior to each run when the experiment occurred. The second solution consists of a variable length inner volume and constant total length thrust chamber of specific construction, actuated, even during the hot runs, by a step-by-step electrical motor.

The thrust chamber is sealed through two O-ring elastic seals, which permit the motion of the propellant distributor into the cylindrical chamber, varying the volume. Note that the internal pressure is acting in the direction of increasing the volume and this action helps the work of the actuating device. Consequently, the volume of the thrust chamber can be easily increased during the hot runs of the engine, with a small effort.

In either alternative the thrust of the engine F is discharged outside (to the thrust frame of the test bed) in the region of the head part of the chamber-cooling jacket, through four fixing screws, releasing the motor envelope from the axial stress of the thrust.

Loading capacity of the envelope

A peculiar method of computation was used to assess the total strains induced in the motor's hood, involving the juxtaposition of the separate pressure and temperature strains. This means the following equations are matched by the axial ε_x, circumferential ε_y and temperature ε_{temp} strains affecting the inner envelope of the motor:

$$\begin{cases} \varepsilon_{x_{tot}} = \varepsilon_x + \varepsilon_{temp}, \\ \varepsilon_{y_{tot}} = \varepsilon_y + \varepsilon_{temp}, \end{cases} \tag{1}$$

where the "equivalent" stress-strain dependence for the envelope material, at the actual temperature of the wall, must be taken into account,

$$o_i = o_i(\varepsilon_i, t). \tag{2}$$

A rupture theory must be first applied, e.g.

$$\begin{gathered} \varepsilon_i = \frac{2}{\sqrt{3}}\sqrt{(\varepsilon_x)^2 + \varepsilon_x\varepsilon_y + (\varepsilon_y)^2}, \\ o_\mathrm{i} = \sqrt{(o_x)^2 - o_x o_y + (o_y)^2}. \end{gathered} \tag{3}$$

The axial N_x and circumferential N_y specific loads are respectively given by

$$\begin{gathered} N_x = \int_{-h/2}^{h/2} o_x(z)\,dz, \\ N_y = \int_{-h/2}^{h/2} o_y(z)\,dz. \end{gathered} \tag{4}$$

The local shape of the envelope forces a supplemental relation between the two loads be complied with

$$K = \frac{N_x}{N_y}, \tag{5}$$

where the ratio K is, for the particular case of the cylindrical thrust chamber,

$$K = \frac{p_c \pi R^2 - F}{2 p_c \pi R^2}. \tag{6}$$

Thus a convenient formula relies the total strain produced by the simultaneous action of the internal pressure and thermal dilatation and the internal chamber pressure itself.

$$\varepsilon_{x_{tot}} = \frac{1-2K}{K-2}\varepsilon_{y_{tot}} + A, \qquad (7)$$

where

$$A = 3\frac{1-K}{2-K}\frac{\int_{-h/2}^{h/2}\frac{o_i}{\varepsilon_i}\varepsilon_{temp}dz}{\int_{-h/2}^{h/2}\frac{o_i}{\varepsilon_i}dz}. \qquad (8)$$

These relations are valid for the self-supported inner envelope of the MRE-1B motor and are easily expandable to the case when a double wall construction is used[4]. The following loading diagrams for the chamber and nozzle throat regions resulted with the specific parameters of MRE-1B engine (Fig. 5).

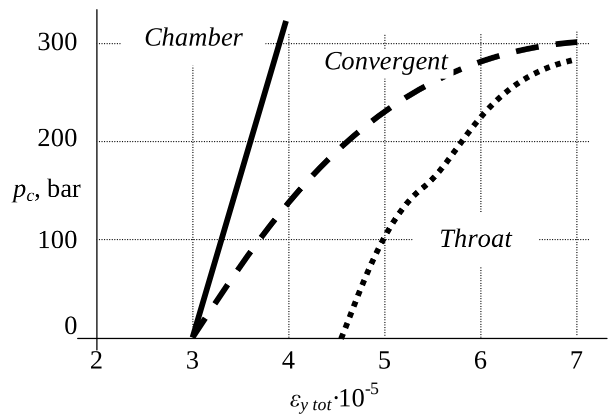

Fig.5. Strain-pressure profiles

The chamber proves a very high loading capacity, while the most loaded zone of the nozzle throat still has a strain reserve of 14.3.

*Fig.6. Module **M** with* **MRE-1T** *engine*

MRE-1 first hot test firing

The **MRE-1B** liquid propellant rocket engine was successfully fired for 20 seconds on April 9-th, 1969 at the main aerospace laboratory of the University "Politehnica" in Bucharest. The firing was observed with considerable interest, under the evaluation of a commission led by the late professor Elie Carafoli, a famous aerodynamicist in supersonic wing theory and former president of the International Astronautical Federation between 1968 and 1972. Professor George Baranescu, by then Rector of UPB and member of the Romanian Academy, later a scientist in US, was a main attendee. A short 2x8-mm color movie was shot to record the experiment. That very first time the ADDA research system proved reliable and the small, incipient Romanian space propulsion program thus started.

Considerable thermo-chemical numerical computations, performed by the author on an IBM-340-30 computer, based on a proprietary set of computational methods and of high temperature thermo-chemistry data about the combustion products, backed the experimental program. The main purpose of these investigations was aimed to determine the relative propulsive efficiency of different combinations of liquid rocket propellants, having in mind future applications in the construction of meteorological and sounding rocket vehicles. It resulted that the most attractive double liquid propellant in the class of storable and cost effective recipes is the *Nitrogen Tetroxide-Ethanol* system, able to deliver a high specific impulse with markedly moderate combustion temperatures.

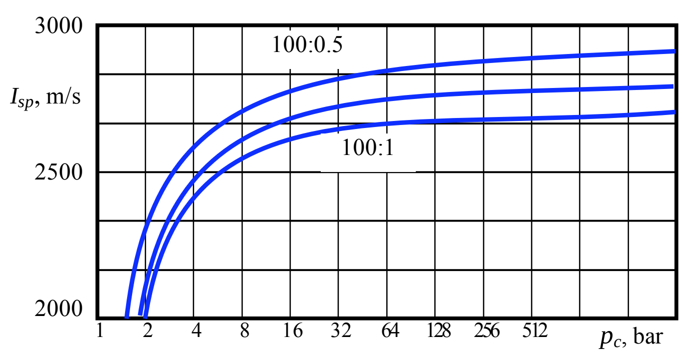

Fig.7. Effect of absolute combustion pressure

This observation was considered as important in regard with the flight deliverable qualities of the sounding vehicles in the ADDA program. As commonly known and shown in a previous research [2], for atmospheric ascent vehicles in the range of a mass ratios of μ=3÷4, the specific impulse I_v and the mass ratio μ manifest almost equal effects and both should be equally enhanced. Consequently a higher I_v under a moderately high combustion temperature, meaning moderate temperature strains in the shell, may appear attractive.

The self-sustained, double-wall structures were used further for the development of larger thrust but low chamber pressure rocket motors like ML-26, able to deliver a 2.6-kN of ground thrust at an optimum value of 10-bar combustion pressure [3]. The inner wall must in this case manifest enough stiffness to survive the compression stresses imparted from the cooling fluid that circles the cooling jacket. The more difficult geometry in ML-26 imposed a different solution for the nozzle area, while the entire motor shall remains identically self-supported. A discharged flow deflector is accommodated in a large cavity between the inner and outer shells, in a solution first provided by Walter motors during WW-II (Fig. 8).

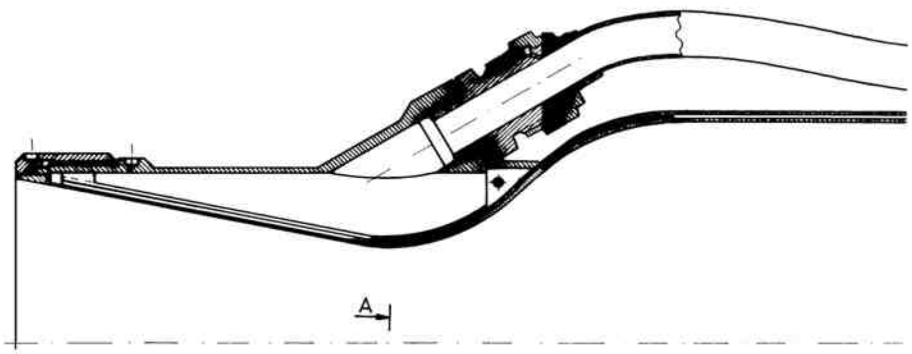

Fig.8. Self supported, large cavity nozzle structure for **ML-26**.

It is observed however that the inner shell is always affected by compressive stresses, due to the dominant pressure of the outer liquid, which circulates in the cooling jacket and its behavior is thus unstable. Consequently such a self supported pressure shell could be used under small pressure levels only and for higher combustion pressures the inner wall must be reinforced by circumferential, helical, longitudinal or punctual connections. Such a solution is present in the original pre-NERVA rocket engine (Fig. 9), which the research team is proposing to improve.

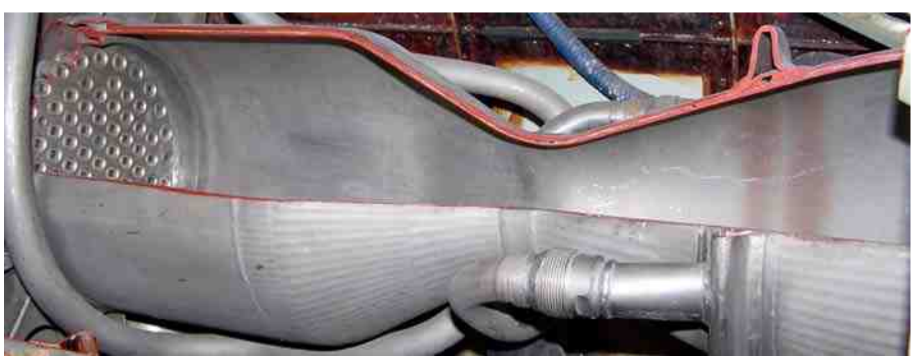

Fig.9. Longitudinal milled-welded connections in the pre-NERVA rocket motor structure.

ADDA reinforced double walls

Well known is also the American alternative of so-called "Spaghetti walls" thin welded tubes technology. In contrast to the welded tubes "spaghetti" solution or to the Russian technology of milled lamellae elevations, welded between the inner and the outer shells of the motor, a technology for impressing rounded hollow channels into the inner wall material, followed by the simple supporting of the outskirt of the engine, was designed for the MRE class of rocket engines at ADDA. Two distinct manufacturing solutions are considered.

Milled inner skirt

The inner shell is provided with the longitudinal separators by milling technology, resulting the structure in Fig. 10 for the cylindrical part of the thrust chamber.

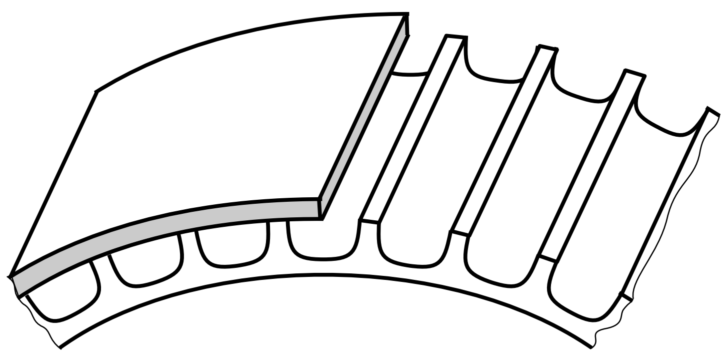

Fig.10. Milled longitudinal cylindrical separator skirt

Profiled mill is used to shape the individual channels as usual, with a converging geometry in the nozzle variable area zone (Fig. 11). For small size thrust chambers, as in the early development ADDA program related to ML-26 construction, a single-piece solution was envisaged for the chamber-nozzle shell.

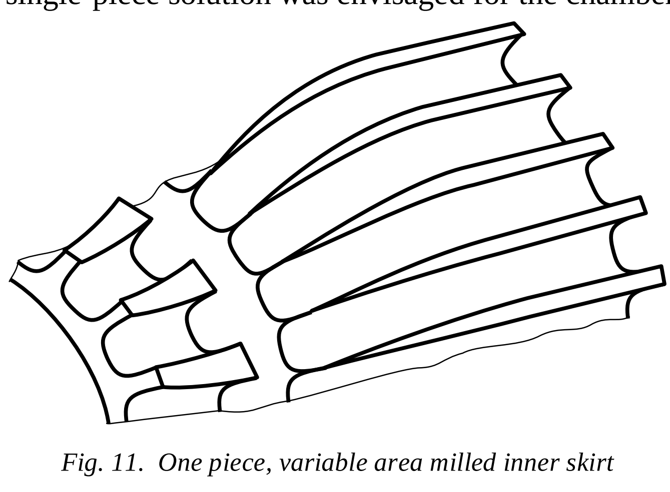

Fig. 11. One piece, variable area milled inner skirt

Pressed inner skirt manufacturing

A much-enhanced structure of the inner metallic shell was envisaged by modeling the inner shirt by underwater explosive forming. The method is currently largely used to manufacture extension of large area ratio nozzles, especially for outer-space propulsion. ESA had extensively used this manufacturing technology. In the present case the forming is produced into the profiled matrix from where the shell with longitudinal reinforcements is extracted. The inner shell is further provided with the longitudinal separators by this technology where no metallic material expelling takes place. The structure of the metallic fibers preserves aligned and the mechanical characteristics are more reliable. The geometry in Fig. 12 results.

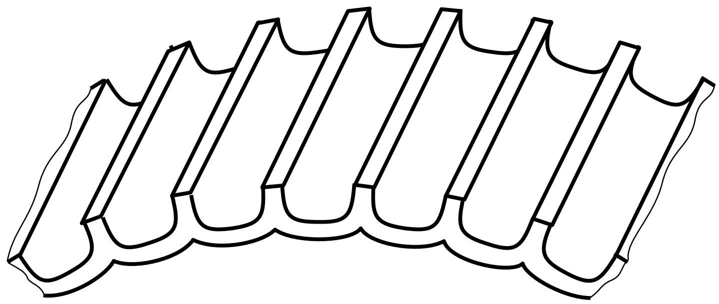

Fig. 12 Result of explosion forming of the inner shell

The main enhancement comes now from the external shell forming and application on the pre-formed inner skirt.

Outer squirt manufacturing

The computational methods developed for the plastic deformation of the double shell under heavy stresses from superposed pressure and temperature gradients [6] show that for low to medium cooling pressures the inner wall tends to enlarge and pushes upon the outer wall from inside. A simple supporting between the walls, meaning without welding between, is assumed as a viable solution and consequently the main technological problem is to properly align the outer skirt upon the already manufactured inner wall.

This process is divided into three steps:

- inner shell channels finishing;
- cooper molding into the channels of the inner skirt (Fig. 13);
- outer surface intermediate finishing of the double metal inner skirt for proper positioning of the outer sell;
- molding/explosive forming of the outer shell upon the massive inner skirt with the cooper filled channels;
- finishing of the outer surface of the outer shell;
- chemical extenuation of the cooper filling from the cooling channels;
- finishing the cooling channels by abrasive spraying and processing.

Removing of the welding process from the technological chain leads to the possibility of dual metal manufacturing of the double wall structure of the engine to better fulfil the different requirements for the inner and outer skirts of the engine, mainly related to the very different temperature regimes encountered. In case of corrosive chemicals as propellant compounds (N_2O_4 or HNO_3 oxidizers), simultaneously used as cooling agents, stainless steel for the inner wall and aluminum/magnesium alloys for the outer may be coupled in an efficient dual metal shell of the motor.

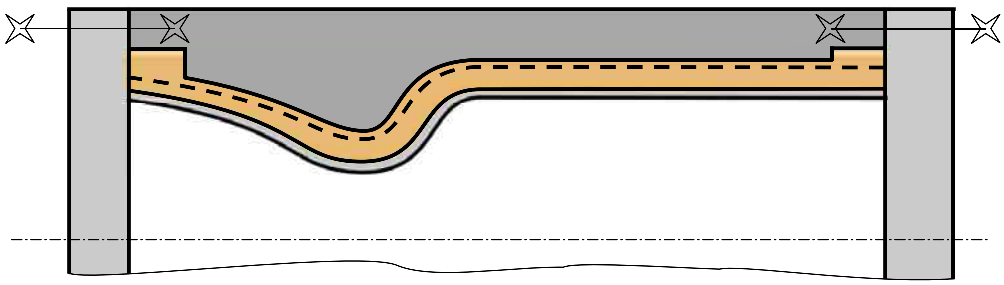

Fig. 13 Cooper molding upon the inner shell.

Extra material is added to assure a complete coverage of the inner shell and terminal masses for the stiffening of the cooper grid.

Finishing of cooling channels.

The finishing operation related to cooling channels is important in regard with the surface microgeometry and with the requirement of removing all possible stress concentrators. A multiple step procedure is followed (Fig. 14).

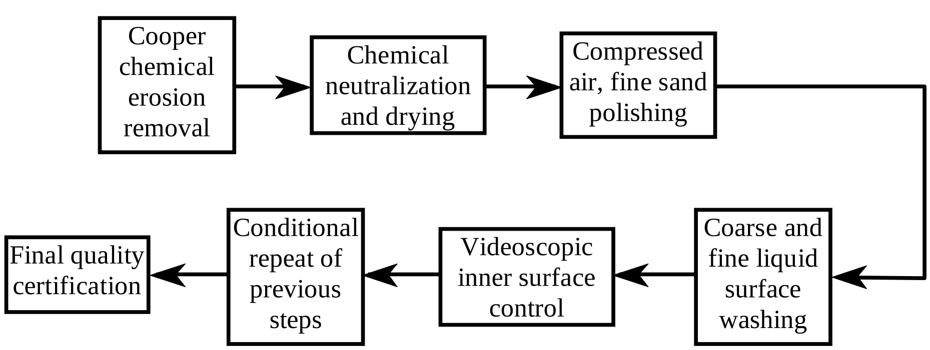

Fig. 14. Finishing plan for the interior of the double wall structure.

Concluding remarks

The double wall, external or regenerating cooling technology for liquid propellant rocket engines is developed into the frames of the ADDA program and follows two directions of investigation. The self-supported, non-bound twin shell technology is mainly accommodated for the low-pressure motors, essentially in regard to the pressure fed systems. The mechanical pressure forming and chemical shaping is adopted for higher-pressure double shells upon a technology recently developed. Its immediate application is related to the improvement of an existing rocket engine system under the NERVA project. This project is intended to create an assessment vehicle for high-speed atmospheric reentry experiments [12] and sub-orbital flights. Enhanced vehicle speed capacity is gained through the development of both more efficient vehicle structure and enhanced capabilities of the rocket motor.

The encouragement of late Prof. Hermann Oberth were of considerable support for the ADDA program, begun in 1962 while the researchers in the team were still students. The feasibility of rocket propulsion for a country like Romania, where all industrial aerospace activities were stopped after the war, was fortunately demonstrated through the construction and successful test firing of MRE-1B. The newly developed technology involves a series of finishing techniques supported on a research project in current promotion. It proves to serve as a reliable test bed for further improvements in the area of higher-pressure space propulsion systems.

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