

Evaluation of F-1 characteristics, based on the analysis of heat transfer and strength of the tubular cooling jacket

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Biographical information about the author

Gennady Ivchenkov graduated from the faculty «Power Engineering» of the Bauman Moscow State Technical University (Bauman MSTU) in 1974, majoring in «Aircraft engines» (Rocket Engines Department) (3rd specialization – solid propellant rocket motors; 1st specialization – LRE (liquid rocket engines)). After graduation he enrolled in graduate school and worked at the Bauman Aircraft Engines Department. Scientific interests – study of heat transfer in rocket nozzles. In 1980 he defended his thesis for the degree of Candidate of Technical Sciences (Ph.D.). The topic of the thesis is a study of combustion in high-speed gas flow. He worked as a researcher at NIIGRAFIT company (research of ablation of carbon-carbon materials), and then as a senior researcher in the department of NIIRP (on the subject of missile defense), then on topics related to the development of HF chemical lasers based on fluorine-hydrogen rocket engines. At the same time he received a second degree at the Faculty of Engineering of the Moscow State Institute of Radio Engineering, Electronics and Automation (MIREA) in specialty «Optical Systems». He was teaching as an Associate Professor in the Khimki branch of Moscow Aviation Institute. Since 1994 he has worked in Canada on the development of fiber-optic devices. He holds seven US patents (fiber-optic sensors and switches).

Introduction

The question of whether the Saturn V rocket engines corresponded to NASA's stated characteristics is directly related to the 'Apollo Moon hoax' – did the Americans really go to the Moon in 1969-1972, or was it an elaborated hoax?

This question was raised initially by the Americans themselves almost immediately after the Apollo missions. Over the years, a considerable amount of direct and indirect evidence has come to light, that at least some of the missions were in fact staged.

It is interesting to ask why known fundamental design flaws of the F-1 engine were not discovered by Soviet rocket engine specialists back in the 1970s. Recently published speeches and letters to the Central Committee of CPSU by Chief Designers of the Soviet space industry Sergei Korolev¹, Valentin Glushko², Vladimir Chelomey³ suggest that

¹ С.П. Королев и его дело. Свет и тени в истории космонавтики. – М.: Наука, 1998. – 716 с., составитель Г.С. Ветров, <http://www.epizodsspace.narod.ru/bibl/vetrov/korolev-delo/obl.html>, p. 363, 449 and others.

² Избранные работы академика В.П.Глушко. Часть 1, Химки 2008, <http://epizodsspace.no-ip.org/bibl/glushko/izbran-rab-glushko/1/01.html>, see. 07.12.1964 Центральному комитету КПСС, 05.04.1965 Секретарю ЦК КПСС тов. Устинову Д.Ф., <http://epizodsspace.no-ip.org/bibl/glushko/izbran-rab-glushko/1/04.html>

even senior management knew about the Saturn V, especially about the F-1 engine mainly from NASA's marketing publications. The Americans didn't disclose any details, at least about the F-1 design⁴. Actually, intelligence data on the articles was classified, often it was very sketchy and largely based on the Russian translations that were publicly available. Thousands of space industry experts did not have any information other than rumors and the US promotional info (and even that was hard to find). To locate an unclassified foreign source in the 1970s in USSR, it was necessary to visit the central library, or to go to the classified library at a place of work.

Furthermore, many documents had to be ordered if they were unavailable. To do this one had to be very persuasive, have patience and a lot of time. It is only relatively recently that a number documents have become available (though having passed through a very thorough American censorship – or so it feels).

A kind of 'critical mass' of evidence has now accumulated, including photographic and film footage, astronauts' accounts, alleged Moon rocks (raising researchers' eyebrows), and inconsistencies (and obvious absurdities) in the design of the Saturn V, its engines, the Apollo craft and the lunar lander (LM). For example, we wonder what genius designed the Apollo Service Module (SM) with pie-shaped sectors and put a large (50 degree circumference) longitudinal reserve compartment in the service partition, which then had to be loaded with a ballast in order to maintain the SM's center-of gravity (!?)⁵ And who decided to include an engine excessive in size and weight – the AJ-10-137 generating 11 tons of thrust, when the Americans themselves state⁶ that it was twice as powerful as necessary, while there was a more suitable engine available (the AJ-10 generating 5 tons of thrust and 200

³ Избранные работы академика В.П.Глушко. Часть 1, Химки 2008, see. 12.12.1966
Центральный комитет КПСС маршалу Брежневу Л.И., <http://epizodsspace.no-ip.org/bibl/glushko/izbran-rab-glushko/1/04.html>

⁴ И.И.Шунейко. Пилотируемые полеты на Луну, конструкция и характеристики Saturn V Apollo, М.,1973, <http://www.testpilot.ru/espace/bibl/raketostr3/1-1.html>, see section «ЖРД F-1 фирмы North American Rockwell, Rocketdyne».

⁵ http://en.wikipedia.org/wiki/Apollo_Command/Service_Module, see Service Module (SM) Construction

⁶ https://en.wikipedia.org/wiki/Apollo_Command/Service_Module, see Service Propulsion System

kg lighter)? But these problematic rocket engines were only one the many questions raised.

Analysis of the available data on the Saturn V and its engines has shown that, with a high probability, it can be argued that the declared characteristics were substantially overstated and did not correspond to actuality. In particular the tubular combustion chamber (C/C) is fundamentally unable to provide the specified pressure and F-1 engine thrust. This is shown in detail in the A.Velyurov study⁷.

Moreover, according to the US Saturn V publicity material, its first stage was the best first stage ever designed. It had five of the most reliable and powerful engines in the world and, in addition, its mass ratio (the ratio of the rocket's wet mass to its dry mass) was the best and unbeatable to date! It was (again, according to the US publicity materials) a sizable 17.5! While this value for the H-1 first stage was equal to 14.4, the Russian Proton booster has a mass ratio of 15, the Soyuz booster 2nd stage – 15.2, the Atlas II – 16, the latest version of the Space Shuttle (if the engines and engine compartment are added to the weight of the external tank) – 17.

This article is an addition to the works published by Professor A.Popov⁸ and A.Velyurov⁹ and is dedicated to assessing the efficiency of the F-1 engine, which, according to US web sites¹⁰, is ‘the pride of the American rocket industry’, ‘the most powerful ever developed’, ‘the most reliable’, etc.

Regarding quotations, all technical information in North America (including technical guidelines, manuals, instructions, etc.) is written by so-called technical writers, who put the available material into literary form accessible to the public. It's just a job, something like a journalist writing on technical subjects. It is quite easy to distort the source material (several inconsistencies in the F-1 records are noted later in this article). Therefore all numbers and technical details in such documents should be treated with caution.

⁷ <http://www.free-inform.com/pepelaz/pepelaz-13.htm>

⁸ <http://www.manonmoon.ru>

⁹ <http://www.free-inform.com/pepelaz/pepelaz-13.htm>, <http://www.free-inform.com/pepelaz/pepelaz-13-2.htm>, <http://www.free-inform.com/pepelaz/pepelaz-14.htm>

¹⁰ <http://history.nasa.gov/SP-4206/ch4.htm>, <http://www.astronautix.com/engines/h1.htm>, http://en.wikipedia.org/wiki/Rocketdyne_F-1

Evaluation of the possibility of cooling and strength of the tubular regenerative cooling of the F-1 engine

This evaluation is based on a comparison of the H-1 and F-1 engines. Having been repeatedly tested and a rather efficient engine (a 'workhorse' of the US space program); the H-1, while an immediate predecessor of the F-1, was superior to it in a number of engineering solutions.

The H-1 Engine

Propellants: LOX & RP-1 (liquid oxygen & kerosene).

The H-1 is an immediate predecessor of the F-1¹¹. H-1 technology was used to develop the F-1. Both engines were developed by Rocketdyne.

H-1: nozzle diameter: 3.6' (1.08 m), C/C diameter: 1.6' (0.48 m), total length of the engine: 8.8' (2.68 m), Area Expansion Ratio: 8 (ratio of nozzle area to throat area), throat diameter: 1.27' (38 cm). These dimensions vary from source to source. Total H-1 length: 218 cm, throat diameter: 33.6 cm¹². Chamber and nozzle are formed by 320 8-mm tubes as a joint body; tube wall thickness is 0.01" (0.254 mm)¹³. C/C pressure is 700 psi (49 kg/cm²)¹⁴.

The kerosene pump pressure is 1020 psi (1 kg/cm² = 14.2 psi) or 71.8 kg/cm².

¹¹ <http://history.nasa.gov/SP-4206/ch4.htm>, <http://www.astronautix.com/engines/h1.htm>, http://en.wikipedia.org/wiki/Rocketdyne_H-1

¹² <http://history.nasa.gov/SP-4206/ch4.htm>

¹³ <http://www.astronautix.com/engines/h1.htm>

¹⁴ <http://www.scribd.com/doc/7244552/Turbopump-Systems-for-Liquid-Rocket-Engines>



Fig.1¹⁵ H-1 cooling tubes

The F-1 Engine

Propellants: LOX & RP-1 (liquid oxygen & kerosene).

Dimensions¹⁶: total length: 19' (5.8 m), the outlet nozzle diameter: 12.3' (3.7 m), throat diameter: 72 cm, chamber pressure: 1000 psi (70 kg/cm²), fuel pump pressure: 1856 psi (131 kg/cm²)¹⁷, or 2000 psi (140 kg/cm²)¹⁸.

The C/C and cooled sections (up to expansion ratio of 3) of the nozzle are made of a set 2x89 tubes, where 89 tubes feed kerosene down and 89 bring it back up. Further (from expansion ratio of 3 to 10) the tubes are bifurcated into 356 tubes – 178 feed up and 178 feed down. The nozzle section from expansion ratio of 10 to 16 is cooled by gas turbine exhaust. The temperature of the turbine inlet gas is 816°C¹⁹ (or 796°C²⁰), output: 650°C²¹. Gas temperature at the exit of the nozzle extension is 1470°K²². Chamber temperature is 3200°C (3500°K). There is no film cooling (it will be explained later). 70% of kerosene is

¹⁵ <http://heroicrelics.org/cosmosphere/engines-h-1-cut-away/index.html>,
<http://heroicrelics.org/cosmosphere/engines-h-1-cut-away/dsc45328.jpg.html>

¹⁶ <http://history.nasa.gov/SP-4206/ch4.htm>,

https://en.wikipedia.org/wiki/Rocketdyne_F-1

¹⁷ <http://www.scribd.com/doc/7244552/Turbopump-Systems-for-Liquid-Rocket-Engines>

¹⁸ <http://www.rocketshoppe.com/forums/attachment.php?attachmentid=8681>

¹⁹ <http://history.nasa.gov/SP-4206/ch4.htm>

²⁰ http://history.msfc.nasa.gov/saturn_apollo/documents/F-1_Engine.pdf

²¹ http://history.msfc.nasa.gov/saturn_apollo/documents/F-1_Engine.pdf

²² Genick Bar-Meir, «Gas Dynamics Tables», Version 1.3, 2007

fed into tubes for cooling, and 30% goes directly to the injectors in C/C²³. Injectors are impinging-stream-type in like-on-like (oxygen-oxygen and kerosene-kerosene) configuration.

The F-1 engine is actually a scaled-up version of the H-1 (H-1 'on steroids'), even more so, remembering that they are both manufactured by the same company, Rocketdyne.

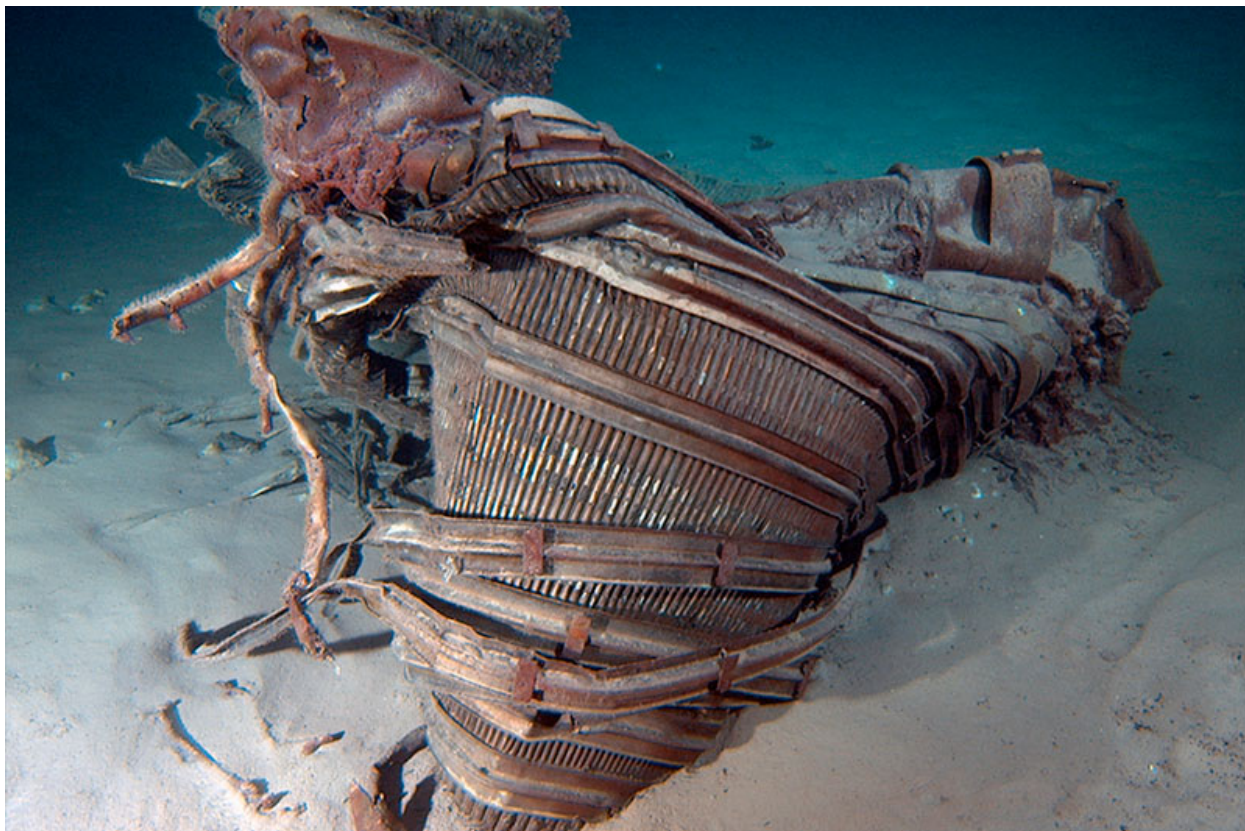


Fig.2²⁴ An F-1 on the ocean floor

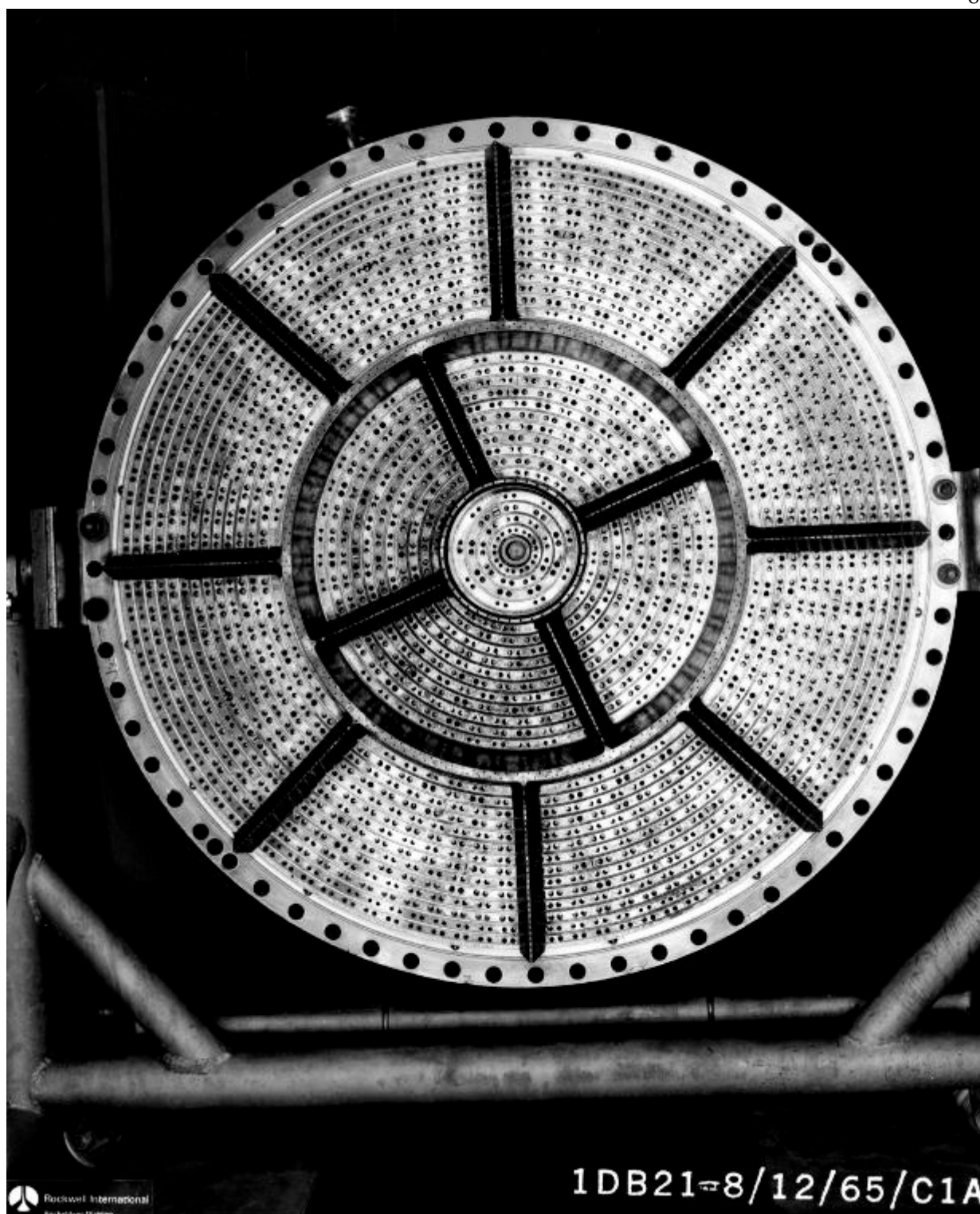
²³ <http://history.nasa.gov/SP-4206/ch4.htm>, https://en.wikipedia.org/wiki/Rocketdyne_F-1

²⁴ <http://www.collectspace.com/news/news-032513b.html>
<http://boingboing.net/2013/03/20/apollo-f-1-engines-recovered-f.html>
<http://www.seattlepi.com/business/boeing/article/Jeff-Bezos-recovers-Apollo-rocket-engines-4370518.php>



Fig.3²⁵ F-1 from the ocean floor

²⁵ <http://www.collectspace.com/news/news-032513b.html>
<http://boingboing.net/2013/03/20/apollo-f-1-engines-recovered-f.html>
<http://www.seattlepi.com/business/boeing/article/Jeff-Bezos-recovers-Apollo-rocket-engines-4370518.php>



F-1 Engine / Injector - 1965

Fig.4²⁶ Injectors are arranged in pairs: oxygen, kerosene. There are no kerosene film injectors.

²⁶ <http://heroicrelics.org/info/f-1/f-1-injector.html>

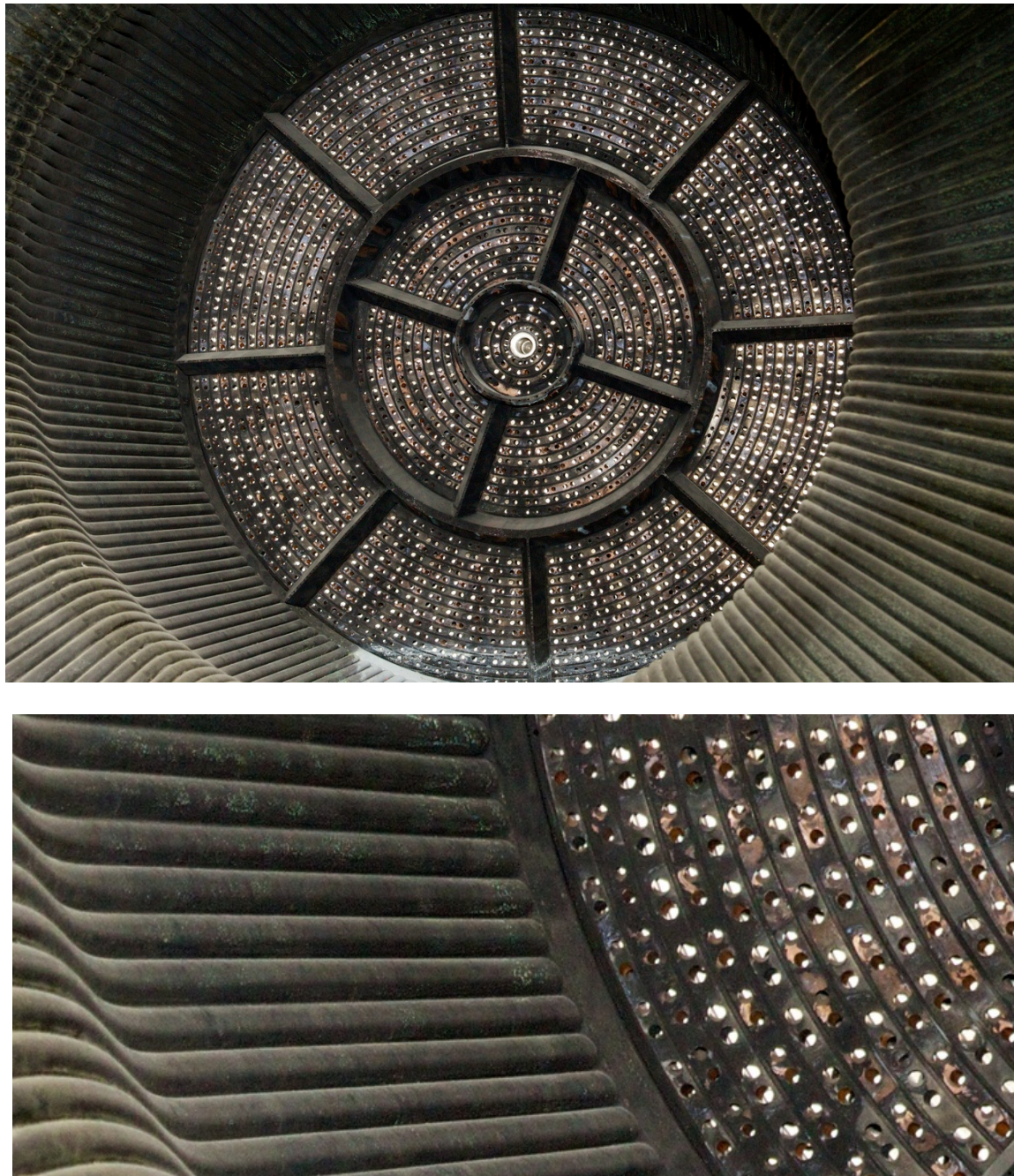


Fig.5²⁷ An F-1 after a test burn. The injector plate is flat. Injector rings are arranged in pairs (oxygen, kerosene, oxygen, kerosene, etc.), wherein injectors are arranged in like-to-like configuration (oxygen to oxygen, kerosene to kerosene). There are no kerosene film injectors. Tubes are covered with patches of green copper oxide evaporated from the surface of the injector plate. An enlarged fragment (lower left) is in the lower photo.

²⁷ <http://cdn.arstechnica.net/wp-content/uploads/2013/03/eande-plate-huge.jpg>



Fig.6²⁸ F-1 injector plate from the ocean floor

The combustion chamber is a 'semi-thermal' type 1 meter in diameter, made of thin-walled tubes of a nickel alloy Inconel X-750 rather than the H-1 stainless steel 347. (in such C/C the chamber diameter is comparable to the throat, like a straight tube without converging nozzle section, and subsonic gas acceleration is performed by gas heating, whereas supersonic accelerating – by gas expansion). This material was selected, according to U.S. sources²⁹, because of the best strength-to-weight ratio, which permitted (again, according to US sources^{30 31}) thinner tube walls and solved the problem of cooling (see quote below).

Inconel X-750 tubing was chosen for the F-1's thrust chamber because it provided the required high strength-to-weight ratios needed to withstand the engine's thrust requirements, which were nearly ten times greater than any previous rocket engine. The high-strength

²⁸ <http://www.seattlepi.com/business/boeing/article/Jeff-Bezos-recovers-Apollo-rocket-engines-4370518.php>

²⁹ <http://history.nasa.gov/SP-4206/ch4.htm>, https://en.wikipedia.org/wiki/Rocketdyne_F-1

³⁰ <http://history.nasa.gov/SP-4206/ch4.htm>, https://en.wikipedia.org/wiki/Rocketdyne_F-1

³¹ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

*property of this alloy permitted design of thinner wall section tubes, resulting in minimum weight. The thinner tubes also design provided adequate thrust chamber cooling with only approximately two-thirds of the total available fuel flow passing through the tubes.*³²

Strength-to-weight ratio is important in aviation and in rocketry, but it is applied to enclosures. In the case of Liquid Propellant Rocket Engines (LPREs) (the internal wall of C/C in rocket and aircraft engines) the ratio of thermal conductivity and strength is most important, as thermal conductivity of the tube material is directly connected with their thickness, and their thickness determines the strength and accordingly maximum pressure in C/C. The weight of the 1st stage engines is of minor importance.

The characteristics of tubular coolant jacket materials of both engines are listed below:

Material of H-1 tubes

Heat-resistant 347 stainless steel³³. Composition: iron 68%, chromium 18%, nickel 11%, the rest is alloy additives.

$T_{\max} = 800-900^{\circ}\text{C}$, yield strength $\sigma = 2480 \text{ kg/cm}^2$ at room temperature, $\sigma = 1725 \text{ kg/cm}^2$ at $T = 650^{\circ}\text{C}$, and $\sigma = 1605 \text{ kg/cm}^2$ at $T = 740^{\circ}\text{C}$ (1350°F), the thermal conductivity $\lambda = 22.5 \text{ W/m}^{\circ}\text{K}$ (thermal conductivity increases with temperature.)

Yield strength³⁴ is the stretching stress at which the material begins to deform plastically, not exceeding 0.2%; not to be confused with tensile strength (ultimate tensile strength or ultimate strength) – the maximum stress before failing or breaking. 347 stainless steel is not hardening by thermal treatment.

Material of F-1 tubes

Heat-resistant nickel alloy Inconel X-750³⁵. Composition: nickel 70%, chromium 14-19%, a bit of iron (5-9%), the rest is alloy additives.

³² <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

³³ <http://www.aerodynealloys.com/products/stainless/specs/ams-5646.php>,
<http://old.upmet.com/media/321-347.pdf>,
<http://www.rolledalloys.ca/alloys/stainless-steels/347/en/>

³⁴ http://docs.google.com/file/d/0B_0eLWFarOI6TTVobExnV3lSZGc/edit?pli=1

³⁵ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

Since this is a material of the tubular cooling jacket of C/C, it is necessary to elaborate on its characteristics and applications.

Inconel is a trademark of the material. It is also known as Nicrofer 7016 TiNb(tm), Pyromet Alloy X-750(tm), Udimet X750(tm), HAYNES(r) X-750 alloy, Nickelvac X-750(tm), HAYNES(r) X-750 alloy(tm), Pyromet X-750(tm). Standards for tubes made from this material are AMS 5582, AMS 5583. It is used mainly in turbines and nuclear power plants (hot water tubes). It is not used in modern rocket engines. Over the years its only application was on the F-1 cooling jacket (H-1 and J-2 cooling tubes were made of stainless steel). Currently, it is replaced by Inconel 718, which is the material of the structural shell of the Space Shuttle Main Engine (SSME) cooling jacket (its C/C and the upper section of the nozzle are formed by double-shell 'Soviet technology', where the inner wall is made of a copper-silver alloy)³⁶.

In contrast to stainless steel 347, Inconel X-750 can be subjected to a heat treating (tempering), which can approximately double its toughness (depending on the tempering method).

Its characteristics are³⁷:

The maximum operating temperature $T_{\max} = 730^{\circ}\text{C}$, yield strength after heat treatment at a constant temperature of 1300°F for 20 hours, is $\sigma = 8600 \text{ kg/cm}^2$ at room temperature, $\sigma = 5400 \text{ kg/cm}^2$ at $T = 730^{\circ}\text{C}$, the thermal conductivity $\lambda = 21.7 \text{ W/m}^{\circ}\text{K}$ at 730°C and $22.4 \text{ W/m}^{\circ}\text{K}$ at $T = 760^{\circ}\text{C}$ (Table 3, page 2³⁸). Thermal conductivity of materials increases with temperature.

At the same time, for the annealed material (Table 17, page 12³⁹) yield strength is about 3270 kg/cm^2 at room temperature, and 2460 kg/cm^2 at 900°F . Further, at the temperature of $1200\text{-}1300^{\circ}\text{F}$ it slightly increases until $3800\text{-}4000 \text{ kg/cm}^2$, and then decreases to 2250 kg/cm^2 at 1500°F and further to 1940 kg/cm^2 at $T = 1600^{\circ}\text{F}$. Increased yield strength at $1200\text{-}1300^{\circ}\text{F}$ is due to commencing crystallization during utilizing the

³⁶ http://en.wikipedia.org/wiki/Space_Shuttle_Main_Engine

³⁷ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

³⁸ http://www.pccforgedproducts.com/web/user_content/files/wyman/Inconel%20alloy%20X-750.pdf

³⁹ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

alloy at this temperature. The process of complete restructuring of the alloy goes slowly and takes hours.

After a sharp rise in temperature and during operation, a heat-treated material has the tendency to form intercrystalline cracks⁴⁰. Problems with alloys like Inconel X-750 have been described professionally and in detail by S.Pokrovsky⁴¹. Under heat and force, changes of the structure in the annealed material were observed, especially under thermal stress. Changes in the Inconel X-750 structure under heat treatment are described in detail in *INCONEL® alloy X-750*⁴² in the Metallography section.

After proper heat treatment at a temperature of about 1300°F the material hardens, and becomes like a spring, (it's used to make high-temperature springs)⁴³ can't be machined practically, and accordingly, it can't be plastically deformed. Thus, all tube forming, such as tapering (change of tube diameter), bending (C/C and nozzle profiling) must be done prior to thermal treatment. Moreover, even without tempering, Inconel X-750 is very hard for machining, particularly for diameter changing (tapering):

*Because Inconel X-750 was a high-nickel alloy, it possessed a low ductility. This material, coupled with the tubes' large diameter, thin walls, high internal operating pressure, and rounded tube crowns made tapering much more difficult, necessitating the bifurcated design. This arrangement also proved to be lighter than using a single tube to the 10:1 expansion ratio plane.*⁴⁴

As a result, Inconel tubes can form a nozzle with maximum area expansion ratio of 3:1 (or 1.73:1 of THROAT diameter – not to be confused with C/C diameter) without bifurcating tubes (expansion ratio or section ratio is the area of the exit divided by the area of the throat). At the same time, a nozzle made of the more flexible stainless steel 347 may have expansion ratio of 8:1 without bifurcating tubes (such as

⁴⁰ <http://www.science.gov/topicpages/i/inconel+x750.html>

⁴¹ С.Покровский «Почему полеты на Луну не состоялись» (S.Pokrovsky, *Why The Moon Missions Didn't Take Place*) www.manonmoon.ru/addon/22/inkonel.doc

⁴² <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

⁴³ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

⁴⁴ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

H-1 engine). Again, it is practically impossible to form a nozzle after tempering, as tubes buckle and become spring-like.

According to *FURNACE BRAZING THE F-1 THRUST CHAMBER FOR APOLLO*⁴⁵ tube brazing was done in two cycles with special braze-alloy for 18 hours for each cycle at a maximum temperature of 2050°F (1100°C). Thus, simultaneously while brazing, the alloy was undergoing some thermal treatment, similar to annealing (see graph in Fig.33, page 21⁴⁶). Meanwhile the sequence diagram of brazing temperature, contained in the source, completely focuses on the brazing process, without any analysis of changes in the structure of the material⁴⁷. Naturally, this is not hardening, but annealing, which has nothing to do with the standard heat-treatment of Inconel X-750 as described in the document⁴⁸, where Inconel X-750 tubes must be kept at a constant temperature of 1300°F for 20 hours. This source⁴⁹ points out that brazing should be carried out at a temperature of more than 1700°F, and to avoid the temperature zone of 1200-1300°F, where the material has poor ductility. Thus, this ‘thermal treatment’ of tubes did not lead to hardening of the material and its yield strength was $\sigma = 2400 \text{ kg/cm}^2$ order at temperature of 900-1300°F⁵⁰, which is about one and a half times higher than that of stainless steel 347. The fact that the tube material was being annealed, rather than heat-hardened is fully supported by F-1 pictures ‘from the ocean floor’ (Figs.2 and 3), which show that the tubes are bent (i.e. ductile material). If they were heat-hardened, they wouldn’t bend but fracture (try to bend a spring).

Moreover, Inconel X-750 features may cause problems with short heating under pressure (particularly during an F-1 burn). Thus, some crystallization begins on the internal fire wall surface of the tubes with some strengthening and, more importantly, an increase in the hardness and, accordingly, brittleness; whereas inner layers of the fire wall and the other wall are not subjected to this. The pressure in the tubes

⁴⁵ http://agentdc.uah.edu/homepages/dcfiles/UAHDC/Furnbrazflthrucham_082007094528.pdf

⁴⁶ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

⁴⁷ http://agentdc.uah.edu/homepages/dcfiles/UAHDC/Furnbrazflthrucham_082007094528.pdf

⁴⁸ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>, p.16

⁴⁹

http://agentdc.uah.edu/homepages/dcfiles/UAHDC/Furnbrazflthrucham_082007094528.pdf,

p.10

⁵⁰ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>, Table 17

increases and plastic deformation occurs; fractures may appear on the brittle surface. Inconel X-750 features are well described in the source⁵¹ however, with regard to the tubes brazing. It turns out that Inconel X-750 is a problematic material, in particular, because of the possibility of its uncontrolled restructuring during operation. Stainless steel doesn't have such complications. It is not surprising that this material was never again used in rocket engines.

Furthermore, when developing the H-1 engine, additional problems arose related to the interaction of the nickel alloy with kerosene RP-1:

*Not only [102] was this condition a hazardous condition and a hindrance to engine performance, but investigators also suspected that problems of combustion instability could be traced to fuel spraying embrittlement of the nickel-alloy tubes, a shortcoming that did not appear in the 734000-newton (165000-pound) engine because it operated at lower temperatures. In the hotter operating regimes of the 836000-newton (188000-pound) thrust engine, researchers discovered that sulphur in the kerosene-based RP-1 fuel precipitated out to combine with the nickel alloy of the thrust chamber tubes. The result: sulphur embrittlement and failure. The "fix" for this deficiency in the new uprated engine involved changing the tubular thrust chamber walls from nickel alloy to stainless steel (347 alloy), which did not react with sulphur.*⁵²

It turns out that in the early H-1 engines nickel alloy (Inconel X-750 is a nickel alloy) tubes were used. When the pressure increases from 40 to 49 atm, the wall temperature increases accordingly, the tubes become brittle, particularly due to the reaction of sulfur from kerosene RP-1 with nickel, and it resulted in the destruction of tube walls and the motor. **To address this, nickel tubes in the H-1 have been replaced by stainless steel 347 tubes!** The question arises, what about nickel alloy tubes in the F-1? After all, RP-1 kerosene is the same, alloy Inconel X-750 has high content of nickel (70%), and the F-1 temperature and pressure are higher than the H-1 (the ratio of oxygen/kerosene was increased from 2.23 in H-1 to 2.27 in F-1).

⁵¹

http://agentdc.uah.edu/homepages/dcfiles/UAHDC/Furnbrazflthrucham_082007094528.pdf,

pp.5-7

⁵² <http://history.nasa.gov/SP-4206/ch4.htm>, pp.100-102

This factor, together with uncontrolled crystallization of heat-resistant nickel alloys such as Inconel, raises the question of nickel alloys applicability for fire walls of C/Cs burning kerosene. The conclusion is unambiguous: nickel alloys cannot be (and were not) used for cooling tubes with documented F-1 specifications. In this case, it was necessary either to reduce the pressure in the C/C to about 40 atm, or use stainless steel 347 tubes, which also leads to the reduction of the C/C pressure to 40-45 atm since the lower strength of the material; and thrust respectively wouldn't be 680 tons but about 450-500 tons.

Then maybe the F-1 tubes indeed were made of steel 347? A.Velyurov pointed out this possibility in his article⁵³.

The F-1 rocket engine (Continued)

The claimed F-1 chamber pressure is 1000 psi (70 kg/cm²). Gas-generator cycle, also called open cycle. Claimed specific impulse is 263 seconds in atmosphere, that is slightly higher than H-1 (255s).

The gas generator is fueled with reduced gas (fuel-rich mixgas) at a temperature of 816°C (1090°K) with an exhaust temperature of 650°C (923°K), generating a large amount of soot. The turbine exhaust (according to American F-1 sources⁵⁴) is then used to cool the nozzle, extending from the cooled part of the nozzle with an expansion ratio from 10 to 16 (T = 1610-1470°K). This can be seen in photographs of the working engine as a lot of soot at the nozzle exit.

The thrust chamber's tubes were constructed of Inconel X-750, a high-temperature, heat-treatable, nickel base alloy. 178 primary tubes, hydraulically formed from 1-3/32 inch outside diameter Inconel-X tubing, made up the chamber body above the 3:1 expansion ratio plane (approximately 30 inches below the throat centerline plane).⁵⁵

In addition, to form C/C and nozzle shape the tubes must have a variable diameter. The diameter, as well as the wall thickness could not be found in any source and the author had to resort to geometric

⁵³ <http://free-inform.com/pepelaz/pepelaz-13.htm>

⁵⁴ <http://history.nasa.gov/SP-4206/ch4.htm>, https://en.wikipedia.org/wiki/Rocketdyne_F-1

⁵⁵ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

calculations in order to calculate the diameter of the tubes in the defined cross-section (tubes must fit into the cross section perimeter).

The walls of the cooling jacket tubes have a thickness of fractions of a millimeter, for example, the thickness of the H-1 tubes is 0.25 mm. The source⁵⁶ gives the thickness of F-1 tubes to be 0.457 mm, the accuracy of which is highly questionable, since such a wall thickness completely contradicts the cooling requirements (to be shown later). 90 cm below the throat, tubes split into two secondary tubes with diameter 25 mm. Kerosene (70% of total flow) from the inlet manifold is fed into half of the tubes, goes down to the end of the cooled part of the nozzle, returns through the second half of the tubes up to the C/C area, and after that it feeds the fuel injector (Fig.7). 30% of the total kerosene flow is admitted directly to the injectors through the bypass orifice plug, providing 30% of the consumption at the same differential pressure between the pump outlet and the inlet of the injectors⁵⁷.

⁵⁶ http://docs.google.com/file/d/0B_0eLWFarOl6TTVobExnV3lSZGc/edit?pli=1

⁵⁷ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

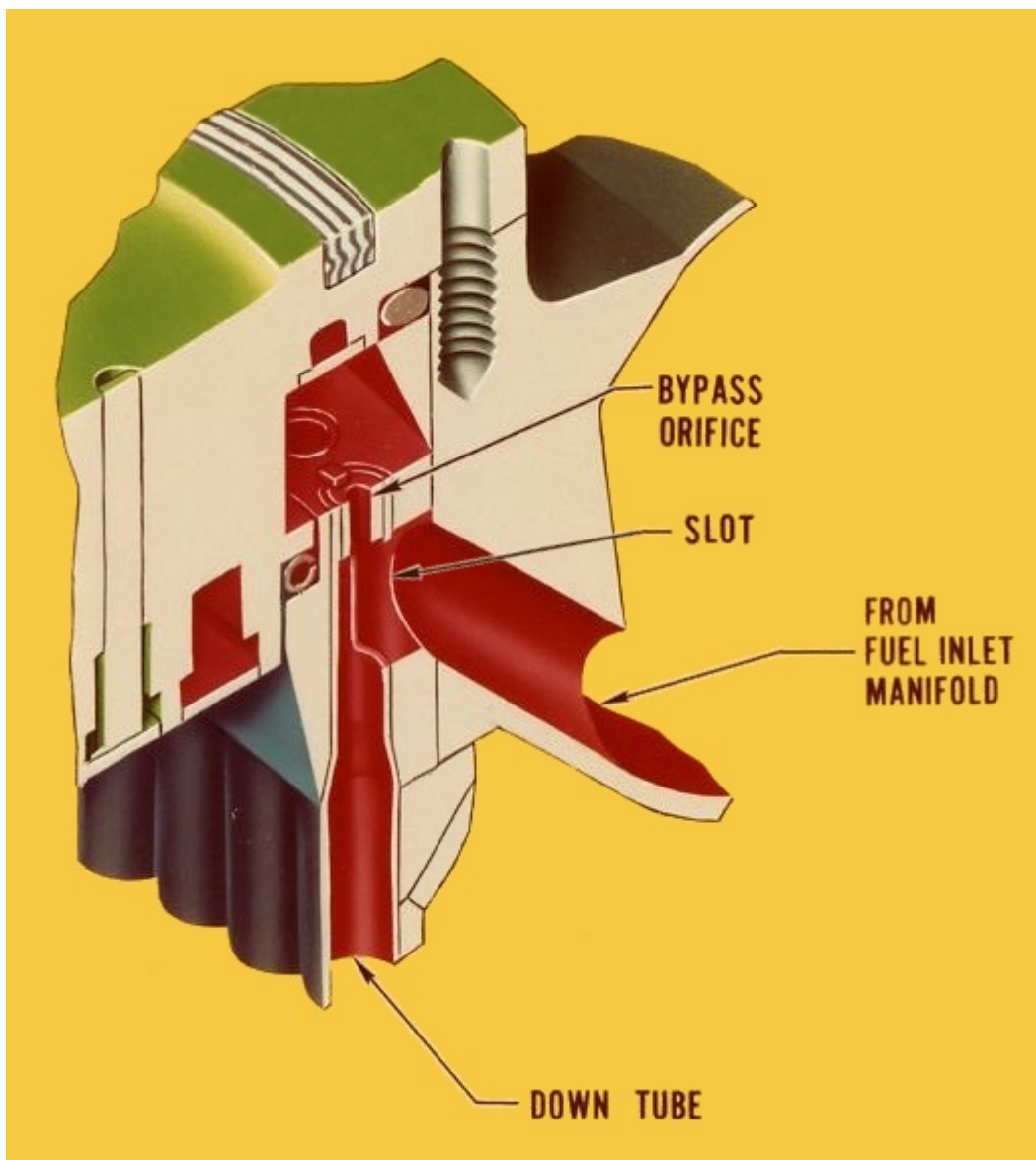


Fig.7⁵⁸ 'The F-1 way' of parting kerosene

⁵⁸ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

Cutaway of Fuel Flow

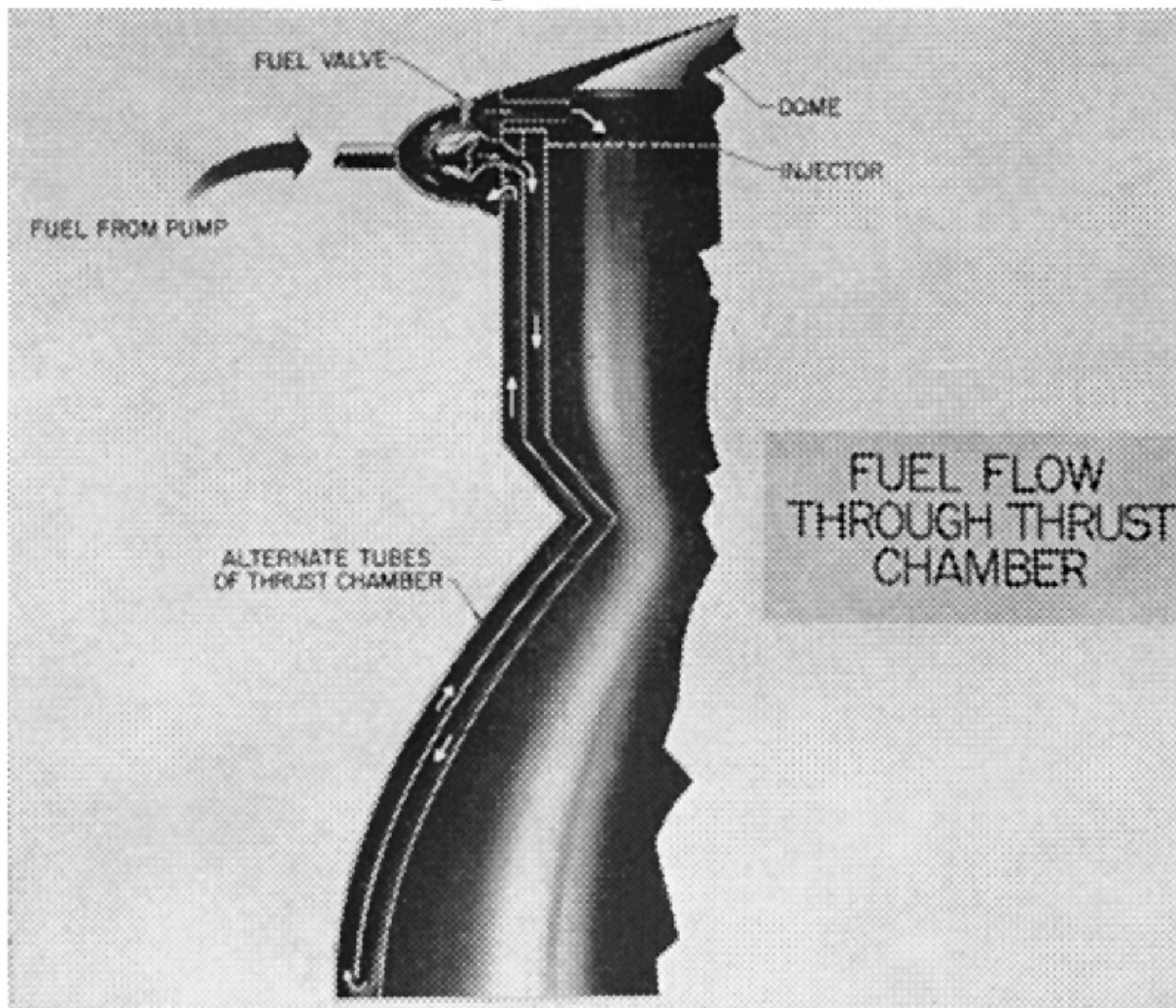


Fig.8⁵⁹ Cutaway of the F-1 cooling jacket

The kerosene temperature increases downstream and the cold kerosene in the tubes, already heated by C/C and by the nozzle, is further heated through the tube walls by now hot kerosene, returning to the C/C.

In the H-1 engine, the cooling jacket also consists of a layer of tubes – even tubes feed the kerosene down, from there it is taken by a collector and comes up through the odd tubes back to the C/C. Again, the F-1 is an H-1 'on steroids.' Such a two-pass system increases the hydraulic resistance of the cooling jacket by a factor of two compared with a single-pass system, when the fuel is fed down in a thick duct, and returns up in the cooling jacket⁶⁰.

⁵⁹ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

⁶⁰ <http://kocmocc.ru/firsttr.php>, Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Wiley Interscience Publication, NY, 2001

If the tubes are arranged in one layer (178 tubes), then their diameter is:

- 18 mm in C/C,
- 13 mm in the throat,
- 22 mm in the nozzle section with expansion ratio 3,
- or tubes have elliptical shape.

And, again, there are no tubes with diameter 28 mm (1-3/32" see citation above) in any of these options and there is neither information about tube diameter and shape in different sections, nor information about tube wall thickness. But the source⁶¹ specified diameter of the secondary tubes (after bifurcating) as 1" (25 mm), which clearly does not fit into the perimeter of the nozzle with expansion ratio 3. This once again shows the contradictions and inaccuracies in the data provided by different sources.

Now let's take a look at the picture of the F-1 nozzle on the ocean floor (Fig.2), taken during the Jeff Bezos' Expedition. In the photo one can count 178 tubes placed below the section with expansion ratio 3 (visible up to the section of expansion ratio 8, and then the tubes are crumpled and bent). And how many tubes are actually there and what is their real diameter?

At the same time in the NASA (Rocketdyne) pictures we can clearly count 178 tubes arranged in a single layer and bifurcating into 356 tubes below the expansion ratio plane of 3:1⁶².

Next, returning to the F-1 design, according to the source⁶³, and other F-1 documents, at the pump outlet 30% of the kerosene is admitted directly to the injector manifold through a bypass orifice plug and the remaining 70% of the fuel is directed down the tubes. If kerosene is fed simultaneously into the tubes and to the injectors, the pressure at the tubes inlet and outlet will be the same and nothing will flow down the tubes. In this case (30% directly and 70% into the tubes) it's necessary to equalize the pressure in the injector plate – that is to throttle the

⁶¹ http://agentdc.uah.edu/homepages/dcfiles/USSRC/F1EngiFamiTraiManu%20Section%201_072308152849.pdf

⁶² <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

⁶³ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

kerosene flow going directly to the injector manifold (the source⁶⁴ indicates something similar – bypass orifice plug). But this is a direct loss of the kerosene turbopump power. Wasn't it easier to direct the entire flow into the tubes? But in this case kerosene pump outlet pressure should increase 1.5 times (with the same tube hydraulic resistance), which, clearly, the tubes of the cooling jacket won't withstand.

Anyway, the F-1 designers have said that they'd successfully solved the problem of F-1 cooling.

F-1 cooling capability and tubing strength

Distinctive features of a combustion chamber of rocket engines

The design of combustion chambers of the US pre-Saturn and Saturn engines H-1, F-1, J-2, RL-10 is based on a large number (from 178 to 356) of brazed thin-walled tubes, made of stainless steel or nickel alloy, which form a cooling jacket, and using jet injectors (impinging-stream-type) in the injector plate.

This is a purely American invention, which was used only in the United States and never ever used again anywhere else. All modern rocket engines, WITHOUT EXCEPTION, including the post-Saturn American ones use 'Soviet technology.'⁶⁵ 'Tubular' American engines are either dumped into landfill, or have ended up in museums, or are occasionally flying on old rockets (H-1 engine modification), such as the Delta II or Japanese H-1.

But engines built on the basis of 'Soviet technology' use the cooling jacket of the two brazed shells, of which the inner skin is made of a bronze alloy and the outer load-bearing skin from a high-strength material. For example, the Vulcain engine for the Ariane 5 has an inner skin made of bronze, and a load bearing skin of nickel alloy. The same

⁶⁴ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

⁶⁵ http://en.wikipedia.org/wiki/Space_Shuttle_Main_Engine
<http://en.wikipedia.org/wiki/RS-68>
<http://en.wikipedia.org/wiki/J-2X>
http://en.wikipedia.org/wiki/Merlin_%28rocket_engine_family%29
<http://kocmocc.ru/firstr.php>
http://vivovoco.rsl.ru/VV/JOURNAL/VRAN/2004/04_06/ROCKET.HTM

with the shuttle motor SSME: its inner skin is made of copper-silver alloy, and the load-bearing of nickel alloy Inconel 718.

The main injector and dome assembly is welded to the hot-gas manifold, and the MCC is also bolted to the hot-gas manifold.^[3] The MCC comprises a structural shell made of Inconel 718 which is lined with a copper-silver-zirconium alloy called NARloy-Z, developed specifically for the RS-25 in the 1970s. Around 390 channels are machined into the liner wall to carry liquid hydrogen through the liner to provide MCC cooling, as the temperature in the combustion chamber reaches 3,315 °C (5,999°F) during flight – higher than the boiling point of iron ... The inner part of the flow is at much lower pressure, around 2 psi (14 kPa) or less.^[9] The inner surface of each nozzle is cooled by liquid hydrogen flowing through brazed stainless steel tube wall coolant passages.⁶⁶

Some engines, such as SSME and LE-7 have C/C, the throat and the upper part of the nozzle formed by the ‘Soviet’ double-wall technology, and the lower part of the nozzle – the nozzle extension, where the temperature and pressure are low – is made of stainless steel tubes, because, as already mentioned, this design is somewhat lighter compared to double-wall.

The Japanese, in the development of liquid hydrogen engines, the LE-5 and LE-7, initially tried to use tubular technology (previously they purchased a license for manufacturing the H-1 engine for their H-1 rocket), but, realizing its lack of prospects, switched to the double-wall technology:

In the LE-5B engine the design was revised from a brazed-tube combustion chamber to an electroformed combustion chamber with copper-alloy cooling grooves... The brazed-tube combustion chamber was abolished, and simplification of the nozzle structure reduced the engine cost.⁶⁷

A curious description and justification of 'American technology' is given in the source⁶⁸:

⁶⁶ http://en.wikipedia.org/wiki/Space_Shuttle_Main_Engine

⁶⁷ <http://en.wikipedia.org/wiki/LE-7>

<http://www.mhi.co.jp/technology/review/pdf/e484/e484036.pdf>

⁶⁸ <http://history.nasa.gov/SP-4206/ch4.htm>

Many early liquid-propellant engines featured a conical nozzle. Engineering improvements in thrust chambers were aimed at more efficient shapes for increased performance and decrease in weight. Designers sought higher performance through higher area-ratio shapes with higher chamber pressures to minimize the size and weight of the thrust chamber. In the drive to produce large, high-pressure engines, a major hurdle was a satisfactory means to cool the thrust chamber. An early solution used double-wall construction; cold fuel passed through this space en route to the combustion chamber, thereby reducing the temperature of the inner chamber wall. But design limitations restricted coolant velocity in the critically hot throat area of the engine. Thin-walled tubes promised an ideal solution for the problem of the thrust chamber walls. Tubes reduced wall thickness and thermal resistance and, more importantly, increased the coolant velocity in the throat section to carry off the increased heat flux there. As chamber pressures continued to go up along with higher temperatures, designers introduced a variable cross section within the tube. This configuration allowed the tube bundle to be fabricated to the desired thrust chamber contour, but variations in the tube's cross section (and coolant velocity) matched the heat transfer at various points along the tube. The bell-shaped nozzle permitted additional [93] advantages in reducing size and weight when compared with what engineers called the "standard 15-degree half-angle conical nozzle." Without any reduction in performance, the bell shape also permitted a 20 percent reduction in length.

It is obvious that the authors (and technical writers) of this quoted NASA report, released in 1974, were totally unfamiliar with the technology used in the Soviet engines, otherwise they would not have written this. Here, however, we must not forget that NASA is a purchaser of the complete rocket, and not an engine developer, and it only has information that was provided by the Rocketdyne Company, and even that was 'processed' by technical writers.

Double-wall construction, as described in this report was used in the V-2, and since the walls were not actually bonded, the shell was losing its firmness at pressures above 20-30 atm. In the Soviet developments in the early 1950s a double-wall stainless steel construction was used, which was coupled by spot welding, thus raising pressure up to 40 atm.

A conical nozzle was used in the earliest engines (V-2, Redstone, Thor) and since then it is not used anywhere.

Tubes with a variable diameter (tapered) were used to form C/C, the throat and a lower part of the nozzle. In the F-1 C/C and nozzle tube diameter varied from 18 mm in C/C to 13 mm in the throat, and up to 22 mm in the nozzle plane with expansion ratio 3 with the transition to 11-22 mm after bifurcating tubes⁶⁹.

Variable thickness of the F-1 tube walls is not mentioned in any source. Again, 'tubular technology' allows only a 'semi-thermal' chamber to form and a supersonic nozzle with area expansion ratio of 8 (without bifurcated tubes) using stainless steel 347, or a nozzle with expansion ratio 3 using Inconel X-750, as the extent of diameter change and tube shaping is technologically limited by ductility of the material (as noted above, Inconel X-750 has very poor ductility).

The first engines using the modern ('Soviet') technology and materials were RD-107/108 (developed in 1954-56). This allowed raising C/C pressure up to 60 atm. The pressure in modern engines, such as the RD-170 and RD-180, made with 'Soviet technology', reaches 250 atm.

All of this suggests that the development of engines in the USSR and in the United States have followed different paths. At the same time, practice has shown that 'American technology' is flawed, deadlocked, and doesn't provide satisfactory engine characteristics, such as the chamber pressure (not more than 50 atm) and, accordingly, the specific impulse. Moreover, the 'American technology' doesn't permit designing modern staged combustion cycle engines, which also reduces specific impulse. One of the very few advantages of 'tubular construction' compared with 'double-wall' is its somewhat lighter weight.

Furthermore, in 'Soviet technology' engines an injector plate doesn't contain jet 'liquid-liquid' injectors, but mono- or bipropellant centrifugal injectors. The 'American technology' injector plate has jet (impinging-stream-type) injectors and resembles a 'washboard with holes' (or a flat plate with holes formed at an angle, see Figs.4, 5). Practice has shown that this is a flawed technology, which does not

⁶⁹ <http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

provide a satisfactory spraying and mixing of the propellants. Actually, the fact that the US used jet injectors is strange, since their well known V-2 engine has all kinds of injectors, and they chose the worst, the only advantages of which, are lower hydraulic resistance and relatively higher flow rate per C/C cross section.

Heat exchange in the F-1 engine

The intensity of the heat exchange in C/C and in the nozzle, in this case, directly affects the strength of the tubes, as it is directly related to the thickness of the tube walls, and the wall thickness defines their durability and thus the allowable C/C pressure.

Here it should be noted that the initial specified F-1 thrust was 1,000,000 lbs (454 tons), which roughly corresponds to C/C pressure of 46 atm. This value was obtained by calculation and F-1 C/C computer simulations. Below is a quote from that same NASA report:

The original Air Force prospectus in 1955 called for an engine with a capability of 4 450 000 newtons (1 000 000 pounds) of thrust or more. Various studies went into comparisons of single engines and clustered engines in terms of their availability and reliability. Parallel studies included detailed consideration of engine subsystems to operate at thrust levels of 4 450 000 newtons (1 000 000 pounds) and up. By 1957, Rocketdyne had produced full, detailed analyses of a 4 500 000-newton (1 000 000-pound) thrust engine, and had also produced some models of components for the big engine, as well as a full-scale thrust chamber. In fact, work progressed so well that Rocketdyne began the first attempts to demonstrate main-stage ignition during the same year. The company's work on the F-1 received a big boost from a new Air Force contract awarded in mid-1958. This document called for Rocketdyne to proceed with the design of a 4 500 000-newton (1 000 000-pound) thrust engine, paralleled by the development of appropriate new fabrication techniques, and capped by running initial tests for a thrust chamber and injector components. Including the prior effort, Rocketdyne had attempted several firing tests of the full-sized thrust chamber between 1957 and 1958. In January 1959, Rocketdyne's NASA contract included requirements for a series of feasibility firings of the new F-1 booster; two months later the engine hinted at its future success with a brief main-stage ignition. The trial

run demonstrated stable combustion for 200 milliseconds and achieved a thrust level of 4 500 000 newtons (1 000 000 pounds). In conducting these tests, Rocketdyne used a solid-wall "boiler-plate" thrust chamber and injector-a far cry from flight hardware-but the unheard of mark of 4 500 000 newtons (1 000 000 pounds) of thrust had been reached by a single engine.

As can be seen from the above quotation, the F-1 design with a thrust of 450 tons, was based on detailed calculation and testing. As shown below, this value is quite realistic and, most likely, was the actual thrust of the real F-1.

It should be noted that the F-1 design, and in particular the injector plate, changed in the process of optimization⁷⁰ (and possibly operation). Also, it is unknown whether Inconel X-750 was initially considered as a material for F-1 tubing.

We will estimate the increase of the thermal flow to the chamber wall in an F-1 while scaling in comparison with the H-1. This takes into account only the convective heat transfer. Radiant flux is neglected, since it is relatively low in the throat plane⁷¹, although in the C/C it can make up to 30% of the total flow, which is an additional factor that increases the total heat flow in the C/C and, accordingly, further limiting the tube thickness.

The F-1 C/C doesn't have film cooling, as can be seen in the photographs of the injector plate (Figs.4, 5) in which oxygen and kerosene injectors in the peripheral compartments are separated into pairs⁷² and there is no a special belt of kerosene injectors along C/C walls.

The photo in Fig.5 (the engine after test firing⁷³) reveals that the outer injectors are apparently overheated, blackened and C/C cooling tubes

⁷⁰ Joseph C. Oefelein, Vigor Yang, «Comprehensive Review of Liquid-Propellant Combustion Instabilities in F-1 Engines», Journal of Propulsion and Power, Vol. 9, No 5, Sept. - Oct. 1993

⁷¹ <http://www.free-inform.com/pepelaz/pepelaz-13.htm>, В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969

⁷² <http://heroicrelics.org/info/f-1/f-1-injector.html>,
http://agentdc.uah.edu/homepages/dcfiles/USSRC/F1EngiFamiTraiManu%20Section%201_072308152849.pdf

<http://cdn.arstechnica.net/wp-content/uploads/2013/03/eande-plate-huge.jpg>

⁷³ <http://cdn.arstechnica.net/wp-content/uploads/2013/03/eande-plate-huge.jpg>

are blotched with copper oxide, evaporated from the surface of the outer sectors of the injector plate, which fully confirms the conclusion about the absence of the fuel film (and even the existence of the inverse flow in the wall zone). Fig.5 shows that the F-1 injector plate neither provided a film, nor did it even provide its own cooling (the surface temperature of the peripheral injectors managed to reach the evaporation temperature of copper!). Moreover, as seen in the photograph, apparently there were problems with feeding the propellants and their combustion. The photo shows that in the course of this test firing, the injector plate did not provide equal propellants mixture ratio that caused fuel-rich mixture in one half of the C/C and oxidizer-rich in the other.

So the claims of 3% kerosene for the C/C film cooling of the 'regular' F-1 in a 1993 article⁷⁴ on preventing F-1 combustion instability (a relatively new article written by authors who graduated from their universities long after 'the Moon race') are highly questionable, especially as these claims, refer to a modification of the injector plate, which was tested throughout the engine design. Some film cooling by gas generator exhaust was used only for cooling the nozzle extension (expansion from 10 to 16).

In addition, one can see fundamental differences in the F-1 C/C, nozzle, and the injector plate design, described in the given sources, in comparison with the actual design of the recovered 'from the ocean floor' engines. This issue will be discussed later.

Since the H-1 and F-1 C/Cs are a 'semi-thermal' type, the ratio of chamber length to diameter L/D is the same, and the F-1 flow velocity distribution along the length of the C/C is the same as that of the H-1. The maximum operating temperature of the tube outer wall of both materials is in the order of 720-730°C (1000°K) – i.e., the maximum allowable for both materials. This applies not only to the F-1, but also to all LRE, as it allows the full use of the capabilities of the material in the most stressed areas, especially as the lower temperature of the C/C and the throat internal wall doesn't satisfy the thermal exchange,

⁷⁴ Joseph C. Oefelein, Vigor Yang, «Comprehensive Review of Liquid-Propellant Combustion Instabilities in F-1 Engines», Journal of Propulsion and Power, Vol. 9, No 5, Sept. – Oct. 1993, p.661

especially for F-1 tubes, in which the thermal flow is substantially higher than that of H-1 (see below).

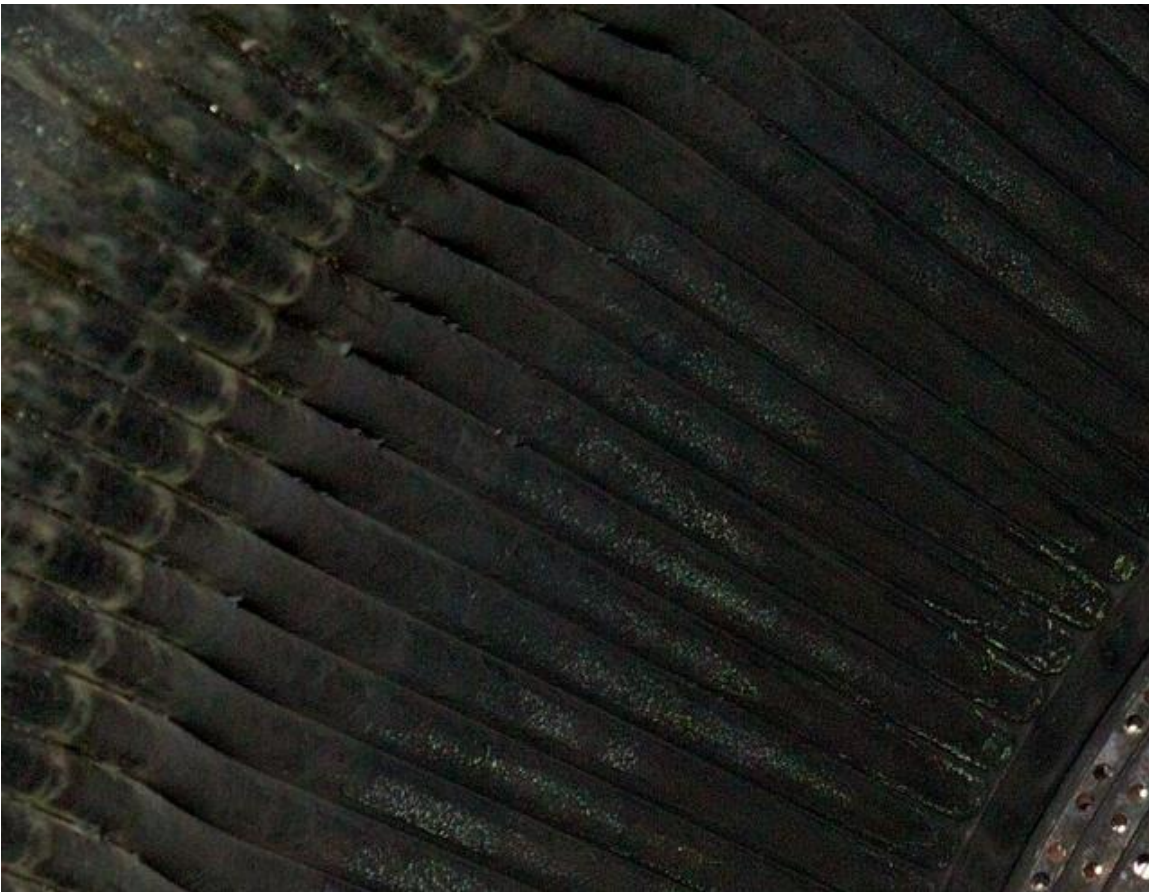


Fig.9⁷⁵ Fragment of Fig.5 (top left part). C/C tubes after F-1 testing, before the throat and in the throat.

In convective thermal exchange the heat flow (Q) goes from the combustion products to the wall. Further, through conduction heat transfer, the heat flow over the tube walls is transmitted to the cooler, which 'takes' this heat flow, due to convection heat transfer at high Reynolds numbers Re , and 'absorbs' it due to the cooler thermal capacity (and, sometimes, also due to a phase transition).

In general, the above specified a classical problem of the conjugate heat transfer in which **specified heat fluxes in steady state must be identical** (as with an electric current in a series circuit).

⁷⁵ <http://cdn.arstechnica.net/wp-content/uploads/2013/03/eande-plate-huge.jpg>

In addition, for further analysis, the most important is the heat transfer through the tube wall which, with the known heat flux, determines the wall thickness.

It should be noted that the heat flow in the C/C and in the nozzle varies significantly along the C/C and nozzle length, reaching maximum values in the throat⁷⁶.

We use the criterion equation of the heat transfer for a tube, which is convenient to use when scaling⁷⁷.

The criterion equation for convective heat transfer in a tube with turbulent flow ($Re > 10,000$) has the form:

$$Nu = N \times Re^{0.85} Pr^{0.4} \quad (1),$$

where N is an empirical coefficient (for turbulent flow in a tube N is approximately equal to 0.023).

Since $Pr \approx 1$ then

$$\frac{\alpha D}{\lambda} = \left(\rho \frac{VD}{\eta} \right)^{0.85} \quad (2),$$

where D is a C/C characteristic size (for a tube – its diameter), then the form for the heat transfer coefficient α is:

$$\alpha = N \frac{\lambda}{D^{0.15}} \left(\frac{V}{\eta} \right)^{0.85} \rho^{0.85} \quad (3)$$

Then when scaled, relative change of the heat transfer coefficient is approximately equal to

$$\frac{\alpha_F}{\alpha_H} \approx \left(\frac{D_H}{D_F} \right)^{0.15} \left(\frac{P_F}{P_H} \right)^{0.85} \quad (4)$$

A similar conversion formula of flows 'for two geometrically similar chambers', that is with a known reference C/C (in this case, this is H-1 engine C/C), is shown in V.Alemasov⁷⁸ (Chapter 14.2 'Conversion of specific convective heat flows', formula 14.55):

⁷⁶ В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969.
Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Wiley Interscience Publication, NY, 2001

⁷⁷ В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969.

⁷⁸ В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969.

$$\frac{Q_k}{Q_{k0}} = \left(\frac{P_k}{P_{k0}} \right)^{0.85} \left(\frac{d_{kp}}{d_{kp0}} \right)^{0.15} \frac{S}{S_0} \quad (5), \text{ where } S = f(T, \eta).$$

Since the combustion products in H-1 and F-1 are the same, then $\frac{S}{S_0} \approx 1$ and the formula is the same as the previous one.

Then the conversion formula is

$$\alpha \propto \frac{1}{D^{0.15}} P^{0.85} \quad (6)$$

For the transitional mode ($2300 < Re < 10000$), the formula is

$$Nu = N \times Re^{0.9} Pr^{0.43} \quad (7)$$

Then the conversion formula for this mode will be

$$\alpha \propto \frac{1}{D^{0.1}} P^{0.9} \quad (8)$$

Consequently, due to the increased pressure from 49 atm (in H-1) to 70 atm (in F-1) the heat transfer coefficient α increases by 1.22–1.29 times (for further evaluations we will use a smaller number). The similar result is obtained in the calculations performed in A.Velyurov's work⁷⁹.

The heat flow in the convective heat transfer is determined by the formula

$$Q = \alpha \Delta T_l, \text{ где } \Delta T_l = (T_{ch} - T_{wall}) \quad (9)$$

Thus, with the same as in H-1 temperature gradient ΔT_l (i.e., with the same temperature difference between the combustion products T_{ch} and H-1 outer wall temperature T_{wall}), the heat flow Q in the F-1 increases about 1.22 times compared to H-1, or ΔT_l should decrease in the F-1 by 1.22 times while maintaining the same heat flux as that of H-1 (see formula (9)). In this case T_{wall} increases up to 1220°K. But, according to the characteristics of both tube materials of the cooling jacket, they will not withstand such temperatures. That is, in any case, the heat flow increases in F-1 by 1.22 times compared with H-1 (at stated pressure 70 atm). **This means that the tube wall must transfer this heat flow to the cooler (kerosene) by the process of heat conduction.**

⁷⁹ <http://www.free-inform.com/pepelaz/pepelaz-13.htm>

The formula for conduction heat transfer (heat transfer through the wall) is:

$$Q = \lambda \frac{dT}{d\delta} \quad (10),$$

where δ is wall thickness.

Or for a monolithic wall⁸⁰ (equation 20.3):

$$Q = \lambda \frac{\Delta T_2}{\delta} \quad (11),$$

where $\Delta T_2 = (T_{wallin} - T_{wallout})$ is the temperature gradient between the inner and the outer walls.

Thus, in order to transfer through the tube wall increased heat flow, it is necessary either to reduce proportionally the wall thickness, or also proportionally increase ΔT_2 (p.377⁸¹). But, with the heat flux increased by 1.22 times and the same wall thickness as that of the H-1 (0.25 mm), ΔT_2 inevitably increases by 1.22 times, i.e. the outer wall temperature rises to 1160°K, which Inconel X-750 will not stand (see formula (11) above).

In maintaining the same wall temperature (more precisely, a temperature difference of 'fire' and 'cold' wall surfaces), its thickness should be reduced by 1.22 times from 0.254 mm (as in H-1) to about 0.2 mm.

Mechanical stress applied to F-1 cooling tubes

In the following analysis, we assume that the tubes are arranged in one layer, and the tube wall thickness is constant.

From the F-1 description it is known that the tubes are brazed with a certain silver-gold alloy and they form the cooling jacket of the C/C and the nozzle. This grouping of tubes, in addition to the thermal stress, also experiences a tensile mechanical stress, which is significantly higher in the F-1 than in the H-1 due to increased diameter and higher C/C pressure.

⁸⁰ В.Е.Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969.

⁸¹ В.Е.Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969.

Tensile stress formula for thin tubes is:

$$\sigma = \frac{PD}{\delta} \quad (12),$$

where P – pressure, D – tube diameter, δ – wall thickness.

In this case it is possible to evaluate the variation of tensile stress of the chamber walls only (assuming that the pressure in the tubes and in the chamber is the same), since the chamber wall has two layers: the tubular shell and the outside jacketing.

Excluding jacketing, it is:

$$\sigma = \frac{PD}{\delta} = \frac{70 \times 100}{4 \times 0.02} = 8.7 \times 10^4 \text{ kg/cm}^2 \quad (13),$$

(4 walls are involved in single layer tubing, see Fig.10a).

This far exceeds $\sigma = 2400 \text{ kg/cm}^2$ for the annealed Inconel X-750 at 720°C and its 'cold' value of 3250 kg/cm^2 . Obviously, the tube material won't withstand such a stress and it turns out that the outer jacketing – the second layer (see Fig.10a) – takes the main tensile force, caused by the C/C pressure. Its exact characteristics were not found, although the source⁸² indicates the jacketing thickness is 5 mm (again, given by technical writers). In this case, the tensile stress of the jacketing will be about $7 \times 10^3 \text{ kg/cm}^2$, which is also greater than its yield strength (of course, if its thickness is 5 mm and it is made of the same material as the tubes).

Here it is crucial not to forget the kerosene pressure of the tube inlet, cooling C/C (see Fig.8) For F-1 it equals to $131\text{-}132 \text{ kg/cm}^2$ (1856 psi ⁸³ or 1870 psi ⁸⁴). Calculation shows that the tensile strength of the tube with diameter of 18 mm (its 'cold' half in contact with the atmosphere) again exceeds the maximum (at overpressure of 131 atm):

$$\sigma = \frac{PD}{\delta} = \frac{131 \times 1.8}{2 \times 0.02} = 5.9 \times 10^3 \text{ kg/cm}^2 \quad (14),$$

⁸² http://agentdc.uah.edu/homepages/dcfles/UAHDC/Furnbrazflthrucham_082007094528.pdf

⁸³ <http://www.scribd.com/doc/7244552/Turbopump-Systems-for-Liquid-Rocket-Engines>

⁸⁴ http://agentdc.uah.edu/homepages/dcfles/USSRC/F1EngiFamiTraiManu%20Section%201_072308152849.pdf

while, as stated, at room temperature the tensile strength is 3270 kg/cm² (see the section of this article about Inconel X-750). This can be compensated for by the jacketing surrounding the C/C, the throat and the nozzle upper part. The F-1 picture shows that the jacketing spreads to almost the section of the nozzle where nozzle area ratio is equal to 3:1 (its end is just above this plane).

Next, half of the tube wall facing the C/C is also under the pressure of

$$P_t - P_{ch} = 61 \text{ kg/cm}^2 \quad (15)$$

at the temperature of 1000°K (since the kerosene pressure in the tubes are higher than in the C/C, see Fig.10b), where P_t is the tube pressure, P_{ch} is the C/C pressure.

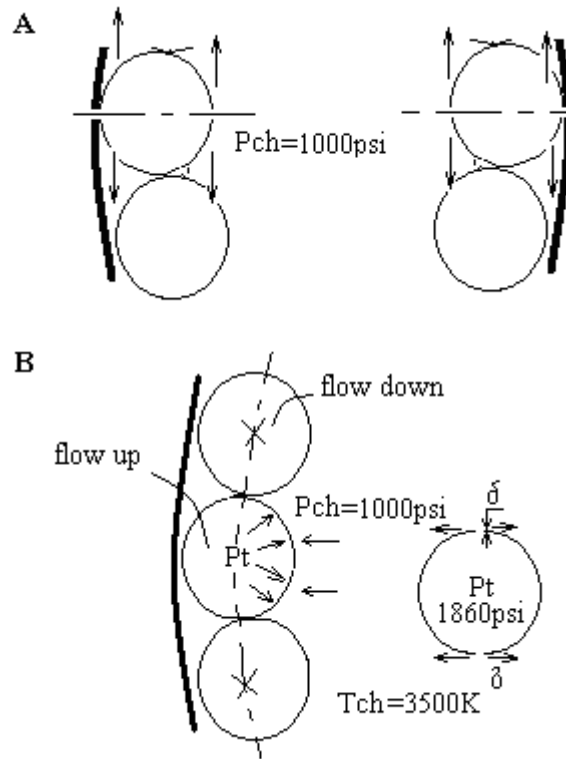


Fig.10. Diagram of the forces applied to the tubes in the C/C

Then, the tensile stress is:

$$\sigma = \frac{(P_t - P_{ch})D}{\delta} = \frac{61 \times 1.8}{2 \times 0.02} = 2.74 \times 10^3 \text{ kg/cm}^2 \quad (16),$$

which is close to the yield strength, but yet again slightly higher. This can't be fixed by any jacketing.

The situation is similar in the throat, despite the fact that the diameter of the tubes (13 mm) is less. In addition, not only is the heat flux greater there (see Fig.9), but the internal pressure (static pressure) is 0.56 of the total C/C pressure (see 'Gas Dynamics Tables'⁸⁵ for $k=1.2$ and $\lambda=1$), that is 39 atm (besides, the throat temperature is somewhat lower: 0.91 of the C/C temperature, i.e. 3190°K).

In addition, the pressure in the kerosene tubes, due to the tube hydraulic resistance, drops roughly proportional to the ratio of the tube length (from the entrance to the throat) and its whole length (two passes – down and up). Given the hydraulic resistance of the injectors, it is roughly 4/5 of the inlet kerosene pressure, which is about 105 atm (at backpressure – the static gas pressure – in the throat of 39 atm).

Then, the tensile stress of the hot wall of the tube 13 mm in diameter is equal to:

$$\sigma = \frac{(P_t - P_{ch})D}{\delta} = \frac{66 \times 1.3}{2 \times 0.02} = 2.15 \times 10^3 \text{ kg/cm}^2,$$

even slightly less than the yield strength of the tubes material, but very close to it.

In the section of the nozzle where nozzle area ratio (expansion ratio) is equal to 3:1 the kerosene pressure in the tubes, taking into account hydraulic losses, is about 63 atm, whereas the static gas pressure in this nozzle plane is the order of 3.9 atm; so it is the extra pressure of 59 atm applied to tube that is also affected by gas temperature of 2200°K. Therefore, the tensile stress of the hot wall of the tube with diameter of 22 mm is equal to:

$$\sigma = \frac{(P_t - P_{ch})D}{\delta} = \frac{59 \times 2.2}{2 \times 0.02} = 3.24 \times 10^3 \text{ kg/cm}^2$$

which again exceeds the yield strength.

All of the above values do not take into account the factor of safety (it is assumed to be 1). If we take it 1.3 (yield safety, that is at the limit) or 1.3 – 1.8 (ultimate safety, sufficient) according to the source⁸⁶, then the above results must be multiplied by this factor and they furthermore exceed the yield strength.

⁸⁵ Genick Bar-Meir, «Gas Dynamics Tables», Version 1.3, 2007

⁸⁶ http://docs.google.com/file/d/0B_0eLWFarOl6TTVobExnV3lSZGc/edit?pli=1

There is a way out of this situation, however, it's rather risky – to allow some plastic deformation, especially for the reason of engine non-reusability. Table 17⁸⁷ contains ultimate tensile strength (UTS) values, which are:

- at room temperature: 7700 kg/cm²,
- at 900F: 7000 kg/cm²,
- at 1200F: 5800 kg/cm²,
- at 1350F: 5400 kg/cm².

Obviously, the above values of tensile stresses of tube material for different nozzle sections are less than ultimate tensile strength of annealed Inconel X-750, i.e. they are likely to work. However, under such strength the material is elongating to the maximum values (before rupture) of 51% at room temperature and 55%, 23%, and 6% at 900°F, 1200°F, and 1350°F respectively (again, at yield strength the elongation should not exceed 0.2%). The value of 6% was obtained for the partially crystallized alloy operating at 1350°F. After a brief exposure to temperature of 1350°F, it will be about 40-50% (the same as at lower temperatures). Since the tube wall temperatures are very different, subject to thickness and place (hot or cold wall), the tube walls will inflate unevenly – the hot more than cold. Together with the effect of increasing the hot wall hardness (its embrittlement) it can cause cracks on the hot wall surface and their spread deeper.

Conclusion

Thus, the tube material of the cooling jacket works at the yield strength, and even beyond it, and this is without taking into account the safety factor. Permitting plastic deformation (under ultimate tensile strength) is extremely dangerous, as the appearance of a crack on the outer surface of the tube 'fire wall' is highly probable and therefore tube rupture is likely (which apparently occurred in the Apollo 6 mission, see below). As noted above, Inconel X-750 is a problematic material, and it was never again used in construction of internal walls of rocket engines (and in rocket engines in general), which further suggests that F-1 developers were to make changes in design 'in the

⁸⁷ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

process' of testing and even engine operation (enormous expenditure and the reputation of the country were at stake).

In this case the main objective was that, under any circumstances, engines wouldn't explode or catch fire 'before highly respectable people's eyes'. Perhaps, the only way to avoid this was to decrease C/C pressure by approximately 30% (see the ultimate safety) to about 50 atm. This decrease wouldn't enable Inconel to pass the yield strength and begin to deform plastically (very likely tubes break).

Decrease of the C/C pressure could be achieved, for example, by throttling of a kerosene flow at the tube inlet (and accordingly oxygen), or by controlling the propellant flow rate in the gas generator, which of course is not optimal for the performance of the engine.

By the way, materials such as Inconel X-750 can perform quite treacherously – in some cases there are no tube breaks, and in others, under the same conditions, tubes can burn through. That means that perhaps in some tests carried out in 'hothouse conditions', for example, at the test facility, the F-1 could have even been able to work at 70 atm for some time (which is, however, doubtful), and in flight, even with minor pressure beats, tubes would break and engine burst in flames.

Taking into account the importance of Saturn V launches, there is a very high probability that Rocketdyne and NASA in some way resorted to a necessitated F-1 throttling to about 50 atm and 500 tons of thrust.

At the same time, for the H-1 everything is satisfactory

Tensile stress of the cold tube is

$$\sigma = \frac{PD}{\delta} = \frac{71 \times 0.8}{2 \times 0.025} = 1.14 \times 10^3 \text{ kg/cm}^2 \quad (20)$$

The hot tube wall tensile stress is

$$\sigma = \frac{(P_t - P_{ch})D}{\delta} = \frac{22 \times 0.8}{2 \times 0.025} = 0.35 \times 10^3 \text{ kg/cm}^2 \quad (21),$$

that with safety margin satisfies the claimed tensile strength of the 347 stainless steel. (1630 kg/cm^2 at 1000°K).

By the way, a rough preliminary estimate of another Saturn J-2 engine (although this is not the subject of this article) shows that tube strength

didn't raise further questions (C/C pressure is 53 atm, 360 stainless steel tubes, tube wall thickness is 0.3 mm). That data is more realistic, though this does not mean that everything was all right – according to NASA, they caused major problems.

Why the Americans had used 22-18 mm tubes instead of 8 mm

The question arises, why had the F-1 designers chosen tube diameters of 22-18 mm, since reducing the tube diameters would have improved the situation, for example, to 8 mm as in the H-1? Then the tensile stress of the tube material would have been about 3 times less. But it turns out, with decreasing diameter, the tubes hydraulic resistance would be increasing and it would require more pressure gradient to maintain the same kerosene mass flow rate, which again would cause strength problems. Thus, turbine power must increase accordingly.

We can estimate the increase in hydraulic loss along the length of the flow after transition from 18-mm tube to the group of 8-mm tubes, having the same total cross-sectional area as one 18-mm tube. Then, with the same mass flow rate, the fluid velocity in the 18-mm tube and in 8-mm tube is the same, but the hydraulic resistance and the pressure loss are different.

Pressure loss per unit mass of the fluid is determined by the formula⁸⁸:

$$h_{1-2} = h_t + \Sigma h_m \quad (22),$$

where h_t –friction loss along the tube, h_m – local hydraulic resistance (loss at the tube entrance, loss in the transition from one to two tubes, etc.).

The pressure loss due to friction along the length of the fluid flow h_t is calculated by Darcy–Weisbach equation⁸⁹:

$$h_t = \lambda \frac{LV^2}{d2g} \quad (23),$$

where λ – hydraulic resistance factor, L – tubes length, d – tubes diameter, V – fluid velocity.

⁸⁸ <http://ars.gubkin.ru/rasthet.htm>

⁸⁹ <http://ars.gubkin.ru/rasthet.htm>

Hydraulic resistance factor (in the equations in this fragment this is λ , is not to be confused with the thermal conductivity) is dependent on the Re number, and is defined by the equations (24, 25, 26):

$$\lambda = \frac{2.7}{\text{Re}^{0.53}} \quad (24)$$

for $2000 < \text{Re} < 3000$, and

$$\lambda = \frac{0.3164}{\text{Re}^{0.25}} \quad (25)$$

for turbulent flow in smooth pipes at $\text{Re} > \text{Re}_{cr}$ (Blasius equation).

Then

$$\lambda \propto \frac{1}{\text{Re}^{0.25}}, \quad \frac{\lambda_1}{\lambda_0} = \left(\frac{V_0 d_0}{V_1 d_1} \right)^{0.25} \quad (26)$$

and, assuming $V_1 = V_0$, we get

$$\frac{\lambda_1}{\lambda_0} = \left(\frac{d_0}{d_1} \right)^{0.25} \quad (27)$$

When $d_1 = 8 \text{ mm}$ and $d_0 = 18 \text{ mm}$ the hydraulic resistance (λ) increases by 1.23 times. Then, putting (27) into the equation for calculating the pressure losses

$$\frac{h_{t1}}{h_{t0}} = \frac{\lambda_1}{\lambda_0} \frac{d_0}{d_1} = \left(\frac{d_0}{d_1} \right)^{1.25} \quad (28),$$

obtained from the equation (23) for $L_1 = L_0$ (the same tubes length) and $V_1 = V_0$ (the same fluid velocity); we obtain almost 3 times increase of the friction pressure loss during the transition from 18 mm to 8 mm tubes. The same calculation is obtained from the equations given in Alemasov⁹⁰ (equations 20.24 and 20.25, page 385).

It is quite obvious that this option does not work and it is clear why the F-1 designers decided to increase tube diameters. But at the same time increasing their diameter leads to increase of stress in the tubes material ($\sigma \propto d$). It is a vicious circle, and apparently a trade-off was selected, which, despite 1.5 times greater Inconel X-750 yield strength, didn't save the situation and, again, C/C pressure could not match the documented for F-1 70 atm.

⁹⁰ В.Е.Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969;

Two-pass tubes cooling system, inherited from the H-1, increases twice the cooling jacket hydraulic resistance compared with supplying kerosene back to the C/C face through a pair of fuel ducts. In principle, on the contrary, one could supply kerosene down through fuel ducts, and it would go up through 178 parallel tubes, the length of which would be two times less (see formula (23)). Such cooler feed is widely used in rocket engines⁹¹, for example, in the RD-107 engine⁹². However, such a design 'spoils the beauty of the engine'.

Throughout the design, similar calculations (of course, more detailed, including computer modelling) for certain were conducted by the F-1 designers, and they got quite a real operating pressure of 46–50 atm and engine thrust of about 450 tons. How they tried to further force F-1 to 70 atm and 690 tons, and what came of it, is a great Rocketdyne secret. One can assume that the proposal to increase the C/C pressure to 70 atm could be received due to the fact that the strength of Inconel X-750 is half times greater than 347 steel and the 'Edisons of Rocketdyne' had decided that it was possible to also increase half times the pressure – from 49 atm (as in H-1) to 70 atm.

The author is not going to incriminate the F-1 designers in the ignorance of the process of thermal exchange or the strength of materials (we assume it was studied in full at MIT), and he has no intention of questioning the real achievements of NASA, but it is obvious that in the case of the F-1 stated characteristics and design, the tube material (Inconel X-750) was exposed to temperatures and pressures which exceed its capability.

Technically, the use of the Inconel X-750 alloy could enable reduction of the tube wall thickness due to its half times greater strength, compared with 374 stainless steel, which was absolutely necessary for the transmission of increased, compared to H-1, heat flux; but such reduction (to 0.2 mm) in reality doesn't work because of both cold and hot material strength (see above). The main cause is an excess pressure in the cooling tubes, which 1.8 times exceeds C/C pressure. The given

⁹¹ http://docs.google.com/file/d/0B_0eLWFarOl6TTVobExnV3lSZGc/edit?pli=1; Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Willey Interscience Publication, NY, 2001

⁹² <http://kocmocc.ru/firstr.php>

(by NASA in Table 1⁹³ F-1) tubes material thickness of 0.457 mm completely fails to cool in the C/C and the throat areas, as well as mentioned in [95] reduced temperature of the hot wall (530°C)⁹⁴. Both of these values don't satisfy the requirements of the tube cooling rate of the C/C and the throat and 'are on the technical writers' conscience'.

Let us assume that somehow we have lowered the outer wall temperature of Inconel X-750 to 800°K (530°C as mentioned in [95]), with the same wall thickness and constant heat flux. This inevitably reduces the difference between the wall temperatures to 300 degrees and also reduces the thermal conductivity of the alloy by 1.23 times (at 530°C)⁹⁵. Then, the heat flux through the tube wall would be reduced by 2 times, see formula (11) (and thus, the heat flux from the gas to the wall would increase by about 8%, since the temperature difference is 200 degrees higher with the same heat transfer coefficient α), which unavoidably leads to an increase in wall temperature to the same 1000°K, and the balance of heat fluxes again establishes as the electric current in the series circuit where the current is the same for all elements. With the heat transfer gas-solid wall-liquid, as already mentioned, **the heat flux must be the same (in steady state)**. That is, the flux from gas to wall is equal to flux through the wall, and equal to the flux from wall to kerosene.

Furthermore, since the heat flux from gas varies along the length of the chamber and the nozzle, then the wall temperature also varies for different sections. For example, in the exit of the cooled nozzle the heat flux from gas is about 10 times less than in the throat⁹⁶, respectively the heat flux through the wall is 10 times less as well. This allows for increasing the wall thickness of the lower bifurcated tubes to 0.45 mm and lowering the temperature difference to about 120 degrees, but only from this point of the nozzle and below. Then the temperature of the outer wall is 620°K, and this is enough to take the heat flux in this plane. Accordingly, in the tubes bifurcation and down the nozzle the wall temperature is around 800°K (530°C). So a

⁹³ http://docs.google.com/file/d/0B_0eLWFarOl6TTVobExnV3lSZGc/edit?pli=1

⁹⁴ F-1 rocket engine data manual, pp.3-7

http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19750070161_1975070161.pdf quote from <http://vif2ne.ru/nvz/forum/arhprint/287676>

⁹⁵ <http://www.specialmetals.com/documents/Inconel%20alloy%20X-750.pdf>

⁹⁶ В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969

technical writer can safely write about the wall thickness of 0.45 mm and about the temperature of 530°C (800°K), but this data is useless without specifying to what area of the tubular cooling jacket it relates.

Especially because for the throat and the C/C given numbers certainly do not work; there the temperature of the wall is at maximum and the wall thickness is at minimum. In the C/C and in the throat, with the wall thickness of 0.45 mm, the heat flux, transmitted by the wall, must be about two times smaller (because the temperature gradient is twice lower)⁹⁷ than in the H-1, and to achieve at least the same heat flux as in H-1, the wall temperature should increase up to 1500°K. It is obvious that no material will stand it. But in the divergent section of the nozzle the temperature drops to 1600°K and walls with thickness of 0.45 mm can be quite acceptable.

In addition, getting acquainted with documents submitted by NASA shows that NASA does not own the detailed engine design documents from Rocketdyne, (again, NASA is a customer, not an engine designer). For example, in the above mentioned Table 1⁹⁸ there is no data on the RL-10 engine combustion chamber; it's 'data not available'(?). It turns out that NASA only knows about engines to the extent as provided by Rocketdyne.

In addition, extremely small wall thickness could also cause strength problems (thermal expansion ratio, thinning of the wall, thermal shock, etc.) Problems with Inconel X-750 were described above (see section Inconel X-750) and by S.Pokrovsky⁹⁹, who personally worked with such materials.

It turns out that the pressure (and probably temperature of the combustion products) had to be reduced which, apparently, was done.

Therefore the F-1 thrust was only 450-500 tons instead of the declared 690 tons. Naturally, these engines can lift off the pad only a smaller version of the Saturn V – 2000 tons instead of the stated 2,800 tons.

⁹⁷ В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969

⁹⁸ http://docs.google.com/file/d/0B_0eLWFarOl6TTVobExnV3lSZGc/edit?pli=1

⁹⁹ С.Покровский «Почему полеты на Луну не состоялись» (S.Pokrovsky, *Why The Moon Missions Didn't Take Place*) www.manonmoon.ru/addon/22/inkonel.doc

This estimate corresponds well with the fact that the first F-1 versions had a designed thrust of 1,000,000 lbf (454 tons).

We can also assume that the 'real' Saturn V (S-V) rockets which were launched after Apollo 6 (such as Apollo 8 and Apollo 11) had their S-V 2nd and 3rd stages replaced by Saturn I (S-1) rockets set on the top of the first stage of S-V. A two-stage S-1 rocket is very similar in dimensions and appearance to the 2nd and 3rd stages of 'real' S-V and has a comparable weight. (Its first stage H-1 engines use oxygen-kerosine; the second stage RL-10 engines use oxygen-hydrogen). Therefore, for such a substitution to look right, the first stage of an S-I (replacing the 2nd stage of this 'modified' S-V) would simply require a little 'make-up' to disguise the bunch of tanks and then such a combination would look absolutely identical to the 'real' Saturn V.

Then the payload to LEO would have been about 30-60 tons, depending on what comprised the second and third stages. This assumption is supported by two independent sets of evidence, from those who witnessed unchanged brightness of the jet flame after the separation of the Saturn V first stage¹⁰⁰ (the brightness of a hydrogen flame is far less). Perhaps the Skylab space station had the same weight¹⁰¹. It must again be noted that analysis of technical solutions used in the Saturn I and Saturn V, as well as in the Apollo craft indicates that the project was being developed in a hurry. For example, the design of the Saturn I first stage is very revealing and consists of a set of tanks of different rockets from Redstone (modified V-2) to the Jupiter rocket:

The propellant tanks consisted of a central Jupiter rocket tank containing LOX, surrounded by a cluster of eight Redstone rocket

¹⁰⁰ An amateur film of Apollo 11 launch shot by Phil Pollacia

http://aulis.com/apollo11saturn_v.htm Fig.3. Rockets and People, Volume IV. By Boris Chertok, (NASA History Series), 2012,

http://www.nasa.gov/pdf/621513main_RocketsPeopleVolume4-ebook.pdf Chapter 9, «When the first and second stages separated, everything was shrouded in billows of smoke and flame. It created the impression that an explosion had taken place, but seconds later the bright, pure plume rushed onward.», observation of the Apollo 8 flight.

¹⁰¹ С.Покровский «Почему полеты на Луну не состоялись» (S.Pokrovsky, *Why The Moon Missions Didn't Take Place*) www.manonmoon.ru/addon/22/inkonel.doc

*tanks: four painted white, containing LOX; and four painted black, containing the RP-1 fuel.*¹⁰²

This is a real Frankenstein. It should also be noted that in the early 1960s there was a ‘megalthic era’ in the American rocket industry – megalomania – at that time that these monster engines were being developed, of which the largest was the M-1 engine¹⁰³. It looked very impressive – as if it was for real, the size of a three-story building and in a picture people next to it looked like ants (see Fig.11). It was surely the ‘King-engine’, but it was never tested and it didn’t ever fly.

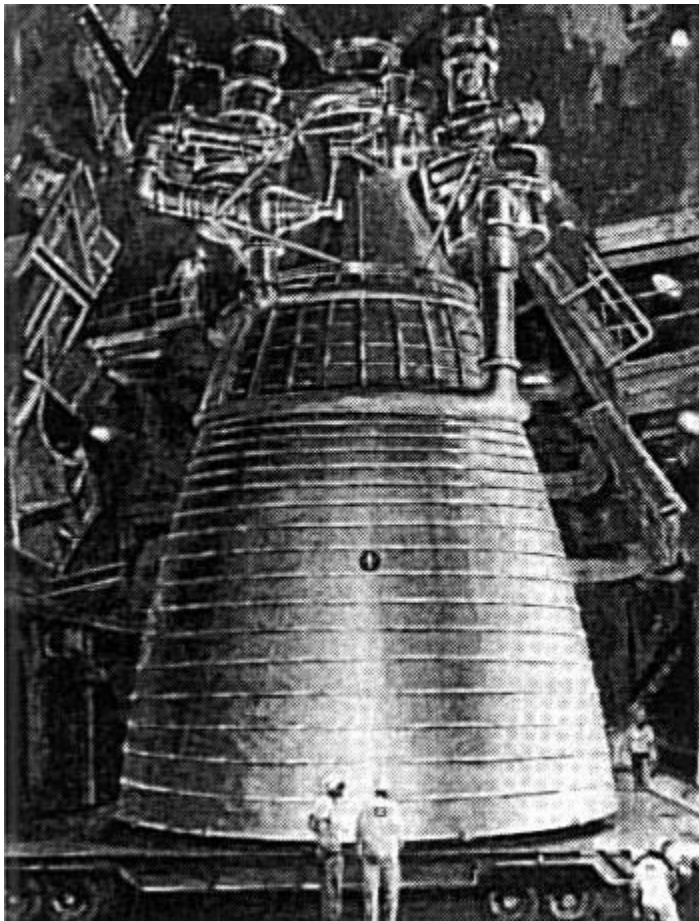


Fig.11¹⁰⁴ The M-1 engine – looks impressive

The differences in the design of the engines (pulled from the ocean floor by Jeff Bezos) from those presented in NASA sources are listed in this article.

¹⁰² http://en.wikipedia.org/wiki/Saturn_I

¹⁰³ [http://en.wikipedia.org/wiki/M-1_\(rocket_engine\)](http://en.wikipedia.org/wiki/M-1_(rocket_engine))

¹⁰⁴ <http://www.astronautix.com/engines/m1.htm>

Let's come back to the picture of the real F-1 at the ocean floor (Fig.2). In the supersonic section of the nozzle 178 tubes can be counted in the part with the expansion greater than 3:1. 178 tubes can be traced to the expansion of about 8:1 with no hint of any bifurcation (tubes can't be counted farther since the nozzle is wrinkled); although after the 3:1 section there should be 356 tubes.

In Fig.6 (the right side) it's noticeable that the face of the injector plate differs from that specified in NASA's documentation.

The question arises therefore, what was the real design of the engine, the one that isn't in any museum and/or in any black-and-white NASA pictures, or the one that did actually fly?

Also, why are 178 tubes located below the expansion ratio plane 3:1? It is known that Inconel X-750 tubes cannot go beyond expansion ratio 3:1. Is this tube material not Inconel X-750? In the A.Velyurov's article¹⁰⁵ the possibility was discussed when, unable to deal with Inconel X-750, Rocketdyne was forced to replace it with the proven stainless steel 347, which can be expanded up to 8:1.

Ejecting turbine discharge gas into the F-1 nozzle exit section

In the F-1 engine, tangential ejection of gas was used into the nozzle exit section with the expansion ratio of 10:1, as it can be seen by the large amount of soot emitted from the nozzle. Here it should be noted that the uncooled nozzle exit was the result of inferiority of the 'tube technology' in which it is impossible to make a cooled nozzle exit with expansion ratio more than 8:1 (for stainless steel tubes without bifurcation as in H-1) or up to 10:1 (for Inconel bifurcated tubes as in F-1)¹⁰⁶. For further expansion an uncooled nozzle has to be used or expansion ratio is limited by 10, whereas using 'Soviet technology' there is no such limitation (the RD-107 engine has expansion ratio of 16 to 1 without any nozzle exit, and the RD-170 has expansion ratio 37:1). Ejecting turbine discharge gas into an uncooled nozzle exit was explained by cooling it by turbine discharge having a temperature of

¹⁰⁵ <http://www.free-inform.com/pepelaz/pepelaz-13.htm>

¹⁰⁶ <http://history.nasa.gov/SP-4206/ch4.htm>
<http://heroicrelics.org/info/f-1/f-1-thrust-chamber.html>

650°C¹⁰⁷ (temperature of the combustion gases at this point reaches 1300°C).

A quote from an MIT textbook:

*Turbine discharge gas (700 – 1100 °C) has also been used as a film coolant for uncooled nozzle exit sections of large liquid propellant engines. Of course, the ejection of an annular gas layer at the periphery of the nozzle exit, at a temperature lower than the maximal possible value, causes a decrease in a specific impulse. Therefore, it is desirable to reduce both the thickness of this cooler layer and the mass flow of the cooler gas, relative to the total flow, to a practical minimum value.*¹⁰⁸

In addition, it is known that gas injection (or liquid injection) into the supersonic section of the nozzle in any case (radial or tangential) causes a shock wave in the injection area (it is not related to film cooling in the subsonic C/C). In solid rockets it is used for thrust vector control¹⁰⁹ as a local shock increases pressure in this place, and, accordingly, produces a control force, perpendicular to the thrust vector.

When building the LE-7 engine the Japanese tried to install such a nozzle. As a result, a shock wave was formed in the nozzle, a significant lateral force appeared and the nozzle burned through. The Japanese gave up on the installation in the LE-7 (as well as in the SSME):

For the new engine model, a nozzle extension was designed that could be added to the base of the new standard “short” nozzle when extra performance was required. But when the engine was fitted with the nozzle extension, the 7A encountered a new problem with unprecedented side-loads and irregular heating on the nozzle strong

¹⁰⁷ <http://history.nasa.gov/SP-4206/ch4.htm>
https://en.wikipedia.org/wiki/Rocketdyne_F-1

¹⁰⁸ Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Willey Interscience Publication, NY, 2001

¹⁰⁹ <http://www.findpatent.ru/patent/204/2046202.html>
<http://pochit.ru/fizika/33692/index.html?page=18>

В.Е. Алемасов и др., «Теория ракетных двигателей», Машиностроение, Москва, 1969

*enough to damage the gimbal actuators and regenerative cooling tubes during startup.*¹¹⁰

Furthermore, in the F-1 a conic system of shocks, caused by injecting a large volume of gas into the divergent section of the nozzle and distributed along the perimeter of the nozzle, undoubtedly narrows the supersonic flow cross section downstream, distorts the expansion ratio of the nozzle, reduces specific impulse (see citation above), makes profiling of the nozzle below the injection point pointless and, as a result, it is unclear what actual expansion ratio and specific impulse the F-1 had. It is possible that such a decision had been empirically suggested by some 'Edison in Rocketdyne'. During tests the nozzle extension (unlike with the Japanese) didn't burn through and that solution was applied for the F-1. By the way, a similar scheme of injecting turbine discharge gas at the nozzle exit was used in the H-1, but it has nothing to do with cooling and serves just for discharging turbine exhaust.

At the same time, some Russian specialists are now trying to introduce 'American technology' into the Russian rocket science¹¹¹, in particular, injection of gas into the supersonic section of the nozzle and a gas generator cycle engines (thankfully at least not tubes), justifying this with simplicity, a lower weight, more reliability and economic benefits. Naturally, the results of this thesis¹¹² were presented at AIAA conferences in the United States. Here, it would be appropriate to remind the candidate for a degree, that the gas generator cycle has a specific impulse lower than the staged combustion cycle engines (closed cycle) of about 20 sec and more (for hydrogen), which is very important for upper stages. And the weight and size of the staged combustion cycle engines will be less. For example, the Soviet RD-170¹¹³, with the same weight as the F-1, has 50 tons more thrust, is 1.5 times smaller in dimensions, has 40 seconds(!) higher specific impulse, and is designed for reuse. The RD-180 (half RD-170), with a thrust of

¹¹⁰ <http://en.wikipedia.org/wiki/LE-7>

¹¹¹ <http://www.dslib.net/mechanika-sostojanij/metod-rascheta-techenija-v-soplah-s-gazovoj-zavesoj-v-sverhzhukovoj-chasti-i.html>

¹¹² <http://www.dslib.net/mechanika-sostojanij/metod-rascheta-techenija-v-soplah-s-gazovoj-zavesoj-v-sverhzhukovoj-chasti-i.html>

¹¹³ <http://en.wikipedia.org/wiki/RD-170>

390 tons, is lighter and smaller than engine RS-68 of the Delta IV rocket, designed as a gas generator cycle¹¹⁴.

By the way, this engine (the RS-68) uses a nozzle extension without gas injection in divergent section (such a possibility wasn't even mentioned in its description), and with the ablative cooling, but a gas-generator power cycle allowed a record low for the liquid hydrogen-oxygen specific impulse of 410 seconds (lost roughly 40 sec), and, in particular, due to the high C/C pressure (105 atm) exceeding the optimum for the gas-generator cycle (about 70-80 atm). Furthermore, all references to its economy¹¹⁵ don't look credible, since high characteristics of staged combustion cycle engines reduce weight and, therefore, the cost of the entire rocket, and cover the costs for their production. Modern staged combustion cycle engines (again the RS-25 or all Soviet staged combustion cycle engines, including the NK-33, made 40 years ago) also have high reliability.

So that the conclusions of a candidate for a degree, A.L.Voinov¹¹⁶ and his advertising of the 'American technology' look highly questionable.

By the way, here's how American sources describe the RS-68 engine design solutions:

*The engine itself is a gas generator cycle engine with two independent turbopumps. The combustion chamber uses a channel-wall design to reduce cost. This design, pioneered in the former Soviet Union, features inner and outer skins brazed to middle separators, forming cooling channels. This method is heavier, but much simpler and cheaper than the tube-wall design (using hundreds of tubes, bent into the shape of the combustion chamber and brazed together) used in other engines. The lower nozzle has an expansion ratio of 21.5 and is lined with an ablative material. The nozzle's lining is designed to burn away as the engine runs, dissipating heat. This is heavier than the tube-wall nozzles used in other engines, but is also much easier and less expensive to manufacture.*¹¹⁷

¹¹⁴ <http://en.wikipedia.org/wiki/RS-68>

¹¹⁵ <http://www.dslib.net/mechanika-sostojanij/metod-rascheta-techenija-v-soplah-s-gazovoj-zavesoj-v-sverhzhvukovoj-chasti-i.html>

¹¹⁶ <http://www.dslib.net/mechanika-sostojanij/metod-rascheta-techenija-v-soplah-s-gazovoj-zavesoj-v-sverhzhvukovoj-chasti-i.html>

¹¹⁷ <http://en.wikipedia.org/wiki/RS-68>

Pay attention to this quote. Benefits of the 'Soviet technology', used to build all modern engines, are outlined briefly and very clearly.

It should also be noted that the American designers use the gas generator cycle because of bare necessity, since they don't have enough experience in the sophisticated technology of staged combustion cycle, especially for kerosene/oxygen engines, in which a gas generator turbine is to run with very explosive oxygen rich gas to avoid falling soot. By the way, the only American staged combustion cycle engine is the Space Shuttle engine RS-25 (SSME), in which a gas generator turbine runs on hydrogen rich gas that does not produce soot. The most viable American post Shuttle SLS program is supposed to use the same RS-25 engines and the rocket itself is strongly reminiscent of the Soviet Energia rocket, just smaller.

Regarding Rocketdyne's struggles with combustion instability

According to the US sources¹¹⁸, both engines H-1 and F-1 during their development exhibited pressure oscillations leading to explosion.

They solved this problem as follows¹¹⁹: a 50-grain (3.5 g) bomb was attached to the face of the injector, enclosed in a nylon case. During engine start the bomb wouldn't explode, as it was protected by the engine fuel from the injector and by the case. When run up the bomb is heated and ignited, creating an acoustic wave in the C/C.

Thereby, the pressure recovery time was analyzed, and next baffle is added to the face of the injector. This way, again, according to the Americans, they had coped with the combustion instability. Below is a quote from the NASA report:

Late in 1963, a research group evolved a technique to induce combustion instability. Workers fixed a special boss to the face of the injector, and attached a small, 50-grain bomb to it. Enclosed in a cylindrical nylon case designed for initial cooling by engine fuel, the bomb was protected during engine start and run up but soon heated

¹¹⁸ <http://en.wikipedia.org/wiki/LE-7> <http://history.nasa.gov/SP-4206/ch4.htm>
http://en.wikipedia.org/wiki/Rocketdyne_F-1

¹¹⁹ <http://en.wikipedia.org/wiki/LE-7> <http://history.nasa.gov/SP-4206/ch4.htm>
http://en.wikipedia.org/wiki/Rocketdyne_F-1

Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Wiley Interscience Publication, NY, 2001

*up, and after a time, it ignited. The explosion disturbed the combustion flame front sufficiently to create an unstable operating condition.*¹²⁰

One must say that this is, of course, a fulsome idea, but it would be possible to initiate a disturbance by feeding the fuel (using a valve with a vibrator or something like that, and in this case the frequency could even be changed), moreover, when working off the combustion chamber on a test stand, a pressure-fed system is used, that is, it wasn't necessary to blow up the chamber.

It is also unclear how one could find out where to add the extra baffle?

By the way, the image of the injector plate (Fig.5) shows that the radial baffle has a number of injectors (apparently kerosene). It has nothing to do with the (cooling) film, but due to injection, increases the effective height of the radial baffle, preventing formation of acoustic oscillations – not a bad idea, by the way (there is no remark regarding this in scientific texts).

Also, it is known that combustion instability depends on C/C pressure disappearing at lower pressure. So, it could be possible that F-1 designers achieved combustion stability not at the stated 70 bar pressure, but at a pressure of just 40-50 bar (as in the H-1).

Reliability of the F-1

In documents submitted by NASA and on various US websites, there are endless comments referring to the high reliability of the F-1, which have logged so many accident free hours, stating that the combustion instability problems were very successfully resolved. Now look at the photo of Apollo 6 flight (Fig.12).

¹²⁰ <http://history.nasa.gov/SP-4206/ch4.htm>



Fig.12¹²¹ The F-1 engine, burning well

Various documents¹²² and others state that Apollo 6 and Apollo 13 had problems with the J-2 engines on the 2nd and 3rd stages. The J-2 papers mention several cases of cooling tube breakthroughs, but in the available F-1 documents there is no such information.

At the same time, in the picture of the Apollo 6 flight, it is obvious that one or more F-1 motors of the first stage are burning. Kerosene is leaking, catching fire and forming a huge tail of flame with soot. It is all clear, but then the question arises, why didn't they explode? The case here is the tubular construction of the cooling jacket. The tubes are not hydraulically connected, and each forms its own independent circuit, and at the breakthrough of one or more nozzle tubes (there are 178 or 356 of them downstream) kerosene pressure in the injectors (and in C/C) drops in proportion to the drop of the total hydraulic

¹²¹ <http://www.npr.org/blogs/thetwo-way/2013/03/20/174834978/we-have-liftoff-apollo-rocket-engines-reportedly-pulled-from-ocean-floor>

¹²² <http://www.universetoday.com/62672/13-things-that-saved-apollo-13-part-5-unexplained-shutdown-of-the-saturn-v-center-engine/>
<http://www.dslib.net/mechanika-sostojanij/metod-rascheta-techenija-v-soplah-s-gazovoj-zavesoj-v-sverhzhukovoj-chasti-i.html>

resistance (breakthrough reduces the hydraulic resistance of a given tube and the flow is redistributed, decreasing in the intact tubes), and kerosene jet leaked from broken tubes burns in the atmosphere at the nozzle exit (with a lot of smoke from the thermal decomposition of kerosene), which, incidentally, may not always lead to an explosion – it depends on where there was a breakthrough. It's like the war song:

'Though there's one motor gone. We can still carry on, comin' in on a wing and a prayer...'

However, you won't fly far away 'on a wing'. But still, basically, it's one of the few pros of the tubular cooling system.

Moreover, as already noted, the engine in Fig.5 had an obvious problem with the uneven mixture ratio along its perimeter (more fuel-rich in one half and less rich in the other), which significantly impairs characteristics of a given engine. Interestingly, do all F-1s exhibit this, or is it just selectively?

Thus, it turns out that NASA, for whatever reason, has not provided complete information about these F-1 problems. Taking into account that the Apollo 6 damage, essentially related to the design of the F-1 engine, occurred seven months before the Apollo 8 mission, it can be concluded that there was no time for fine-tuning of the engine to the declared characteristics.

In addition, it is clear that the flame of burning kerosene 'climbs up' to the 1st stage kerosene tank. Incidentally, to a lesser extent this was noted for all Saturn V launches. Such a phenomenon is unique and is not observed on other rockets, including American ones. For 'normal' Saturn V launches it can be assumed that this is happening at the subsonic phase of the flight and the subsonic part of the plume is sucked from the nozzle – the very same 'injection' into the nozzle extension. The supersonic plume from the nozzle forms 'Mach barrels' – the American textbooks call it 'diamonds'¹²³. These 'diamonds' dissolve far downstream and their gas can't be 'sucked up'. The subsonic combustion products are sucked in due to poor Saturn V aerodynamics – the 'thick' boundary layer with backward flow (they managed to design it in such a way!). In addition, a part of this 'flow' is

¹²³ Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Wiley Interscience Publication, NY, 2001

sucked through the bottom low pressure into the engine compartment that could well cause a fire. NASA's explanation that it is 'quite harmless' does not hold water for obvious reasons. Rocket exhaust plumes are described in detail in the MIT textbook¹²⁴. There, a separate chapter is devoted to them.

And in the case of Apollo 6, the flame of the burning F-1 'is sucked in' (of course, subsonic) into the engine compartment and above, like into the engine bay of a shot down aircraft.

Here, by the way, we can assume that during the Apollo 6 launch it had been attempted to use the F-1 and J-2 engines 'at full extent' with the stated thrust and C/C pressure. The result was a burn-through of the F-1 cooling jacket and J-2 problems (vibration, emergency cutoff, etc.) on the 2nd and 3rd stages. It may have been the real Saturn V rocket with the stated characteristics (probably for the first and last times).

Here a question arises: how did NASA dare to test a rocket with engines with a design that didn't provide the claimed strength characteristics? The fact is, as already mentioned, the material has the second tensile strength – ultimate tensile strength – it is the tensile limit wherein the plastic deformation takes place (i.e. a product changes its shape). This limit is usually 1.5 times higher than yield tensile strength, but in this mode the material can only work for a very short time, only once, until, for example, a tube bursts. This possibility was discussed in this article in the section 'Mechanical stress applied to...'. Taking into account the adventurous nature of the United States Moon program and the enormous amount of money already spent on the program, NASA would 'try' to launch such a rocket: 'what if it happens to work out?'

Moreover, note the flexibility of the Saturn V control system, which was controlling the Saturn flight in near real time without any self-destruction and cut-off commands (apart from the obvious danger of the engine explosion), in contrast to the Soviet N1 rocket, which was practically killed by an automatic control system (in particular, absence of the forced staging command).

¹²⁴ Oscar Biblarz, George P. Sutton (consultant), «Rocket Propulsion Elements», Seventh Edition, A Willey Interscience Publication, NY, 2001

Comments on the F-1 engine design

As already mentioned above:

- The design of the cooling jacket for such a powerful engine with return tubes (this is not the H-1) appears to be quite strange – return tubes further heat kerosene and, in addition, approximately double the hydraulic resistance of the cooling jacket, compared with the 'regular' single-pass systems where the fuel is fed into the nozzle exit and goes up to the C/C injector plate. The only advantage is that the engine looks 'prettier' without 'ugly', fat ducts.
- Tube material (Inconel X-750) is not able to withstand the load applied to the tubes of the cooling jacket as tensile stress is close to (no reserve) or greater than the material yield strength. Furthermore, Inconel X-750 has problems with uncontrolled restructuring during the brief thermal impact (temperature of 1200-1300°F) and with chemical reaction of sulfur from kerosene RP-1 with nickel. Because of this, Inconel X-750 has not been used anywhere in high-temperature parts of the rocket engines, although it could have been used in the upgraded H-1 (the same company – Rocketdyne). Then the C/C pressure could have been raised up to 70 atm (the thrust would have risen to 140 tons), secondary tubes formed to expansion ratio 10:1 and quite a decent engine would have been put together, especially considering that the H-1 was in production up to the early 1980s, and the H-1 license was sold to the Japanese.
- Bifurcation of the flow by 70% into tubes and 30% directly to the injectors causes power loss of the pump and turbine and such solution also looks strange.
- The F-1 nozzle has a significant overexpansion ratio (1:16) that corresponds to the pressure at the exit of the nozzle extension (for $k=1.2$) $0.006 P_{c/c} = 0.42 \text{ kg/cm}^2$ (with the specified C/C pressure of 70 kg/cm^2)¹²⁵ or $0.013 P_{c/c} = 0.91 \text{ kg/cm}^2$ at the exit of the

¹²⁵ Genick Bar-Meir, «Gas Dynamics Tables», Version 1.3, 2007

cooled part of the nozzle (expansion ratio of 10). At the same time, H-1 (ratio of 8) has the pressure at the nozzle exit $0.024P_{c/c} = 1.1 \text{ kg/cm}^2$. In theory, such overexpansion should improve the average specific impulse on the flight path of the 1st stage, but the gas injection into the nozzle extension (see below) completely distorts the nozzle expansion ratio and it is unknown what was the real F-1 expansion ratio and specific impulse.

- Gas injection for ‘cooling’ into the supersonic section of the nozzle unambiguously causes shock waves in the injection area (it has nothing to do with cooling). This narrows the cross section of the supersonic flow and makes pointless profiling of the nozzle extension beyond the place of injection. In this case, again, it is not clear what the actual expansion ratio and exit pressure are. A similar scheme of the H-1 gas generator exhaust at the nozzle exit has nothing to do with cooling and serves to discharge the gas mixture.
- The designs of the F-1 nozzle and injector plate, cited in numerous NASA sources and in other documents, differ from the design of those elements of the Apollo 11 F-1 recovered by Jeff Bezos’ Expedition in 2013.

This list of F-1 design oddities raises the suspicion that the actual F-1 engine was of a different design to the one specified.

Conclusions

The pressure in the F-1 combustion chamber is likely to have been significantly lower than that stated, due to the fundamental shortcomings of the tubular cooling system of American rocket engines. This is further confirmed by the fact that all rocket engines now being developed and currently in use, including those in the United States, use the 'Soviet-style' cooling system.

Consequently, the launch weight of the Saturn V was lower and, accordingly, could not have ensured the accomplishment of the Apollo Moon landing program.

Nevertheless, although the F-1 engines produced smoke – they worked; they were not exploding in front of the public and were

delivering 'something to somewhere'. In general, we can express admiration for the expert Rocketdyne engineers, who managed to get at least some characteristics out of this 'miracle of an American mastermind' (the F-1), as even 500 tons of thrust per chamber is quite a lot.

The true F-1 design and its characteristics, apparently, were significantly different from those stated.

In addition

Let's compare the F-1 with a cluster of 4 NK-33 engines, developed in the same period and for a similar program.

The thrust of 4 NK-33 engines is $154 \times 4 = 616$ tons at sea level, and with use of high-revving NK-33-1 engine (C/C pressure is increased from 150 to 175 atm) – 740 tons¹²⁶. The declared F-1 thrust is 681 tons at sea level.

The weight of 4 NK-33 engines is about 5.6 tons ($1,393 \times 4 = 5572$), and NK-33-1 – 6.96 tons. The weight of one F-1 is about 9 tons¹²⁷.

NK-33 dimensions: height – 3.7m; diameter of the nozzle exit – 1.5m¹²⁸. F-1 dimensions: height – 5.8m; diameter of the nozzle exit – 3.7m.

NK-33 specific impulse – 297 sec (at sea level) and 331 sec in a vacuum¹²⁹, NK-33-1 – 304 and 334 sec, respectively.

The F-1 specific impulse is 263 sec (at sea level) and 301 sec in a vacuum¹³⁰.

Fig.13 shows the F-1 and the NK-33 in comparable scale.

Therefore, it turns out that with almost the same thrust (and NK-33-1 even has 50 tons more) the weight and size of a cluster of 4 NK-33 is more than one and a half times less than the 'highly praised' F-1

¹²⁶ <http://ru.wikipedia.org/wiki/%CD%CA-33>

¹²⁷ http://en.wikipedia.org/wiki/Rocketdyne_F-1

¹²⁸ <http://ru.wikipedia.org/wiki/%CD%CA-33>

¹²⁹ <http://ru.wikipedia.org/wiki/%CD%CA-33>

<http://www.federalspace.ru/main.php/www.mai.ru/conf/video/main.php?id=187&did=873>

¹³⁰ http://en.wikipedia.org/wiki/Rocketdyne_F-1

(according to its very questionable 'stated' characteristics). Moreover, the NK-33's specific impulse was 30-40 UNITS MORE than that of the 'miracle of American technology'. It is easy to estimate how it will affect the performance of the rocket. Comparison of these engines is like the comparison of a steam engine with a modern internal combustion engine. The efficiency of the NK-33 is independently verified by the Americans (the Antares rocket already made a few flights with the NK-33. The NK-33 is certified in the United States and Aerojet plans to sign a contract for the restoration of their production at Kuznetsov's plant), while there is no news on the F-1 in circulation (and there won't be – it's the same as putting a 19th century steam engine on a modern vessel), that is, according to the American saying, the F-1 is an 'urban legend'.

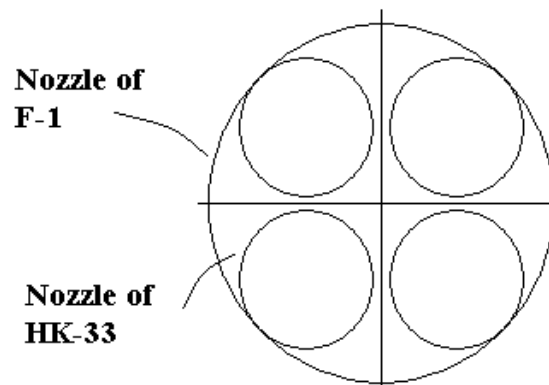
**F-1****2 HK-33**

Fig.13 Comparison of the F-1 and the cluster of 4 NK-33s

Thus, in the 1960s, the United States built an engine that could not possibly have been created on the basis of 'tubular technology'. There is a very high probability that, unable to comply with the claimed F-1 design characteristics, and failing to ensure its reliability, the Americans used a hit-or-miss approach (as with Apollo 6), and then moved on to pure deception, backed up by 'Hollywooding' the record.

They lacked the technology and the knowledge, but could and did splurge to perfection.

Gennady Ivchenkov

November 2013

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