SUPERSONIC AIR INLET FOR A HIGH VELOCITY PROPULSION SYSTEM

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Abstract: *The paper aims to determine the optimal geometry, the control law and the operational limits of a supersonic air inlet; its operation control consists of its central cone positioning, with respect to the flight Mach number, keeping in sight the necessity of a suitable parameters' distribution into engine's combustion chamber. Inlet's optimal architecture shall be determined (based on an algorithm regarding the maximization of the total pressure recovery), using basic planar geometry principles. Based on the optimal geometry of the inlet, its flow rate characteristic and its control law, as well as inlet's operation limits were calculated and graphically represented.*

Keywords: *air inlet, shock-wave, supersonic, detonation, cone, engine, pressure, limits.*

1. INTRODUCTION

Flying at very high speeds is one of the most important challenges today for aerospace engineers and manufacturers, not only for military purposes, but also for civilian purposes. After supersonic flight became almost commonplace, obviously, hypersonic flight has become the new challenge for specialists, both for atmospheric missions and for suborbital and orbital missions.

The military's interest in hypersonic flight is obvious for several reasons: a very fast and maneuverable weapon is difficult to detect and difficult to counteract, due to its short detection time and speed, leaving little time for defense systems to react. [3] . Requirements that have become more urgent in recent years are shortening response times and rapidly attacking mobile targets. While drones, satellites and the like can easily locate all types of targets, highly mobile enemy units will not be "waiting" for the inevitable counterattack; a very fast weapon platform with the ability to maneuver (given its speed) means that, once found, a target will have little time and fewer opportunities to escape [19].

Regardless of the mission of such a high-speed vehicle, a lot of specific issues and challenges must be overcome before it can be put into operation and perform its tasks. First, the effect of aerodynamic viscous friction and shock waves give the body temperatures so high that no conventional material can withstand them ([7], [12]), so that new heat-resistant materials are designed and resistant and new manufacturing appropriate concepts and techniques to be implemented; ionization of the air around the vehicle body also disrupts the propagation of radio waves and interferes with radio frequency sensors and communications. On the other hand, very high air temperatures reduce the pressure of conventional air-breathing engines ([7], [2]), thus requiring new concepts and means of propulsion (such as scramjets, rockets, detonation engines or those almost fictitious plasma engines).

Last but not least, new body structures, new propulsion systems and new flight techniques require new sensors, new equipment and appropriate command and control architectures.

Among the new high-performance propulsion options for high-speed vehicles are detonation engines, which use a wide range of fuels, from conventional kerosene to cryogenic fuels (hydrogen, methane etc.).

2. ABOUT DETONATION ENGINES AND DETONATION PHENOMENA

From historical point of view, the detonation process was firstly described and studied by Berthelot, Vieille, Mallard and Le Chatelier, around 1880; at the beginning of the 20th century, Chapman and Jouguet have presented independently the zero-dimension detonation theory [19].

From the point of view of reciprocating piston engines, detonation represents a dangerous and undesirable phenomenon (also known as “knock of the engine”); it occurs when the injected fuel ignites before the piston reaches the programmed spark ignition and causes a sudden increase of the pressure inside the cylinder (up to 10 times higher than normal) and extreme temperature rises. The consequences could be serious damages to the engine pistons, rings, rods, gaskets, bearings and even cylinder heads and crankcases. Knock can be caused by incorrect ignition timing (due to incorrect ignition preset), poor air-fuel ratio, inadequate fuel octane, exhaust backpressure, incorrect presetting of the turbocharger and/or of the intercooler, as well as ambient heat. Even the best engine components cannot withstand severe detonation for more than a few seconds at a time, the results being severe engine damage (destroyed pistons, bent connecting rods, cracked housing, engine fire etc).

Early attempts to use the detonation for jet propulsion were reported at the University of Michigan (by J.A. Nicholls, in the sixth decade of the 20th century [19]); since this achievement, many researchers from different countries (USA, Russia, China, Japan etc) have brought their contribution to the domain. Last decades have brought new concepts of detonation jet engines, such as Pulsed Detonation Engine (PDE) ([13], [19], [22]), Rotating Detonation Engine (RDE), Continuous Detonation Engine (CDE) [20] or Continuous Rotation Detonation Engine (CRDE) [21].

These engines use conventional or unconventional fuels, from kerosene to liquid hydrogen; the necessary oxygen for the burning reaction may be obtained from the atmosphere by air breathing (only for atmospheric high speed aerial vehicles) or may be carried on board in high-pressure special tanks (suitable for sub-orbital and orbital vehicles).

All these engines are designed to propel high or very high speed aerospace vehicles; most of them are “air breathing type”, so for them it is mandatory to have adequate air inlets, which must ensure both the engine required air mass flow and the required downstream parameters (temperature, pressure, density etc).

For detonating burning to take place, several thermodynamic and kinematic conditions must be met. Fig. 1 [13] contains the curves which describes the thermodynamic process of detonation burning; as example, cycles for a scramjet (curve 1234) and a standing detonation engine (curve abC-J) are presented; comparing to Hugoniot curves for the detonation wave (with its Chapman-Jouguet C-J point) and for the shock wave, the curve of the detonation engine cycle (continuous line) assure the parameters correlation in order to obtain the detonation, as determined in [2], [11] and [13].

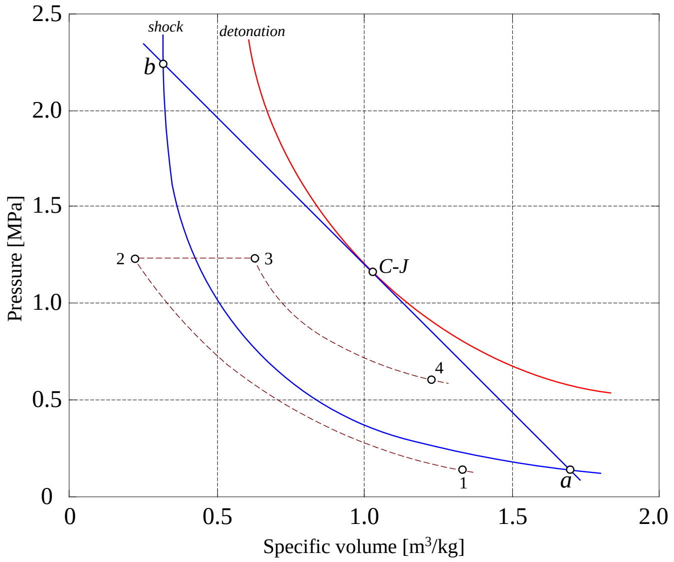


FIG. 1. Thermodynamic general process of detonation, particularly for two different engine cycles

The slope of the detonation line, obviously, is not a constant; from the tangent in the C-J point one obtains, from an isenthropic line passing through C-J point:

$$\left(\frac{p_2}{p_1}-1\right):\left(1-\frac{\rho_1}{\rho_2}\right)=\chi\frac{p_2}{p_1}\frac{\rho_2}{\rho_1}, \tag{1}$$

where χ – isenthropic exponent, assumed as constant (the ratio of the specific heats).

The propagation velocity of the detonation wave U_D is

$$U_D=\left(\frac{p_2-p_1}{\rho_2-\rho_1}\frac{\rho_1}{\rho_2}\right)^{\frac{1}{2}}, \tag{2}$$

proportional to the slope's square root.

As far as the isenthropic transformation gives $\frac{p_2-p_1}{\rho_2-\rho_1}\frac{\rho_1}{\rho_2}=\chi\frac{p_2}{\rho_2}$ and one consider the flow velocity as w, it results that, in the domain between S and D curves in fig. 1, one obtains

$$(U_D-w)^2=\chi\left(\frac{p_2}{\rho_2}\right)=a_2^2, \tag{3}$$

a – sound velocity of the gas.

Thus, for the C-J point on obtains

$$\frac{p_{CJ}}{p_1}=1+\frac{U_D^2}{\frac{p_1}{\rho_1}}\left(1-\frac{\rho_1}{\rho_{CJ}}\right), \tag{4}$$

and having in mind that the detonation Mach number is $M_D^2=\chi p_1 U_D^2/\rho_1$, one obtains

$$\chi M_D^2=\left(\frac{p_{CJ}}{p_1}-1\right):\left(1-\frac{\rho_1}{\rho_{CJ}}\right), \text{ then} \tag{5}$$

$$\frac{\rho_1}{\rho_{CJ}} = \left(\chi M_D^2 + 1\right)/\left(\chi + 1\right)/M_D^2, \tag{6}$$

$$\frac{w}{U_D} = 1 - \left(\chi M_D^2 + 1\right)/\left(\chi + 1\right)/M_D^2, \tag{7}$$

Considering the energy equation applied for the burning gas (assumed as perfect gas)

$$\frac{1}{2}U_D^2 + \frac{p_1}{\rho_1} + \Delta Q = \frac{1}{2}\left(U_D - w\right)^2 + \frac{p_2}{\rho_2}, \tag{8}$$

where ΔQ is the reaction heat per mass unit of mixture gas, one obtains

$$\frac{p_2}{p_1} = -\left(\frac{\chi+1}{\chi-1} - \frac{\rho_1}{\rho_2} - \frac{2\chi\Delta Q}{a^2}\right)/\left(\frac{\chi+1}{\chi-1}\frac{\rho_1}{\rho_2} - 1\right), \tag{9}$$

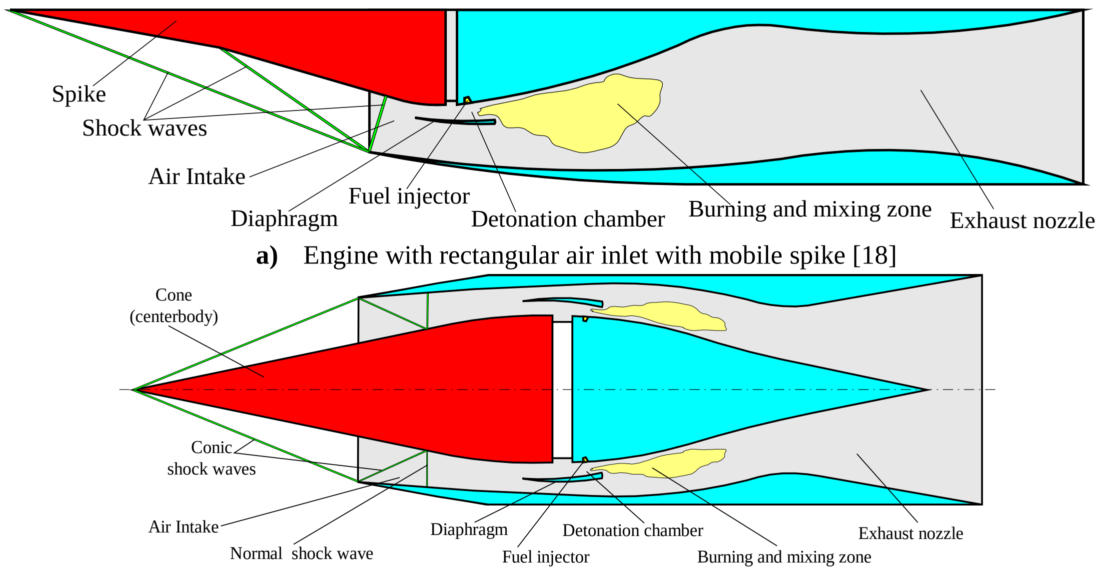
which gives for the detonation Mach number next solutions:

$$M_D^2 = \left[1 + \left(\chi^2 - 1\right)\Delta Q/a^2\right] \pm \left\{\left[\left(\chi^2 - 1\right)\Delta Q/a^2\right]^2 + 2\left(\chi^2 - 1\right)\Delta Q/a^2\right\}^{\frac{1}{2}}. \tag{10}$$

The appropriate solution is those using the + sign ([3], [13]), which associates the propagation velocity to the detonation conditions, while else one obtains the deflagration solution. So, one can observe that the extreme points of the detonation line will give the limits for the free stream velocity, determined with respect to the pressure and density values.

3. PROBLEM DESCRIPTION AND FORMULATION

An air inlet for an aircraft engine must assure the necessary air parameters to keep it in a stable operating mode (air mass flow rate, velocity and pressure) ([5], [7]), whatever the flight regime and the engine speed. The bigger the flight speed is, more important problems are issuing, so the supersonic air inlets are the most important in their class, especially if the engine is of the detonation type.



a) Engine with rectangular air inlet with mobile spike [18]

b) Engine with axis-symmetrical air inlet with translation cone

FIG. 2. Schematics of a high velocity propulsion system based on detonation engine

Some aspects regarding supersonic inlets are presented in [1], [4], [14], [15]; however, the problems solved by those references consider as assisted engine a classical-one, but in this paper one has to solve both the velocity and the pressure distribution problems, in order to keep the detonation process stable.

In [18] a similar problem was solved; as fig. 2.a) shows, the engine was designed to use a subsonic flow of high speed, so the inlet was designed 2D, with a mobile spike. This paper deals with an axis-symmetrical inlet, as in fig. 2.b).

The inlet has very important roles: both connection and correlation, by transforming the air parameters outside the engine into suitable parameters inside the engine, in front of the compressor, especially when it's about the pressure and velocity. Improper pressure condition inside the burning zone can transform the detonation into deflagration and "destroy" the detonating burning, which leads to a significant thrust decrease, without mentioning the thermal overload and the danger of explosion.

Consequently, air parameter configuration inside the detonating burning chamber must be assured at suitable values (pressure and temperature parameters kept within the permissible range ([7], [8] and [12])), no matter the flight regime; that means that the inlet should permanently adapt to the flight regime, which is given by the flight Mach number.

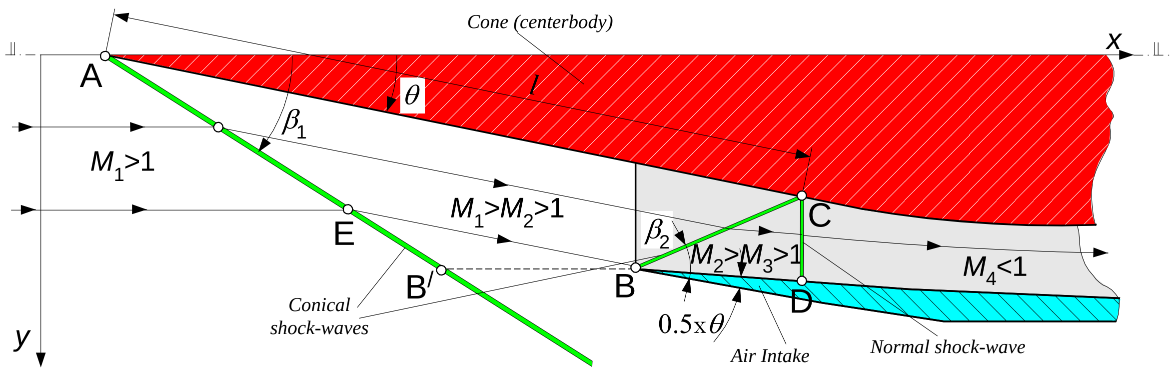


FIG. 3. Supersonic axisymmetric air inlet's geometry

As shown in Fig. 3, the air inlet is an axis-symmetrical -one (with circular cross-section), equipped by a conical centerbody for external compression (which generates a conic shock wave), while the intake's lip generates another shock wave – an internal conic wave, followed by a normal-one, inside the intake. This intake has also a specially profiled sidewall to contain the air flow compression. The inlet adapting to the flight regime should be realized by the central cone translation along the intake's axis, in order to keep the external shock-wave outside the intake ([1], [7], [12], [15]); the nominal flight regime is the most used regime (most used flight Mach number) and corresponds to the situation when the external shock-wave meets the intake's cowl lip D ([8], [12]).

This work aims to determine the optimal architecture (the exact shape of the central cone - section angle and ramp length), as well as the control law of the air inlet (the dependence of this central cone's position on the flight Mach number, in order to assure suitable parameters after the inlet, in the detonation chamber).

4. INLET OPTIMAL ARCHITECTURE ISSUING

Air inlet geometry design is usually based on two categories of methods, which are: a) aerodynamic methods – based on analytical and numerical procedures; b)geometric methods – based on planar geometry elements.

Optimization criteria might be: a) the total pressure recovery maximization (Oswatitsch condition); b) the drag minimization and/or c) the inlet flow rate correlation. Optimization studies for such external compression type inlets usually uses "carpet search method" (described in [8] and [12]), or the "method-of-characteristics" (presented in [2]).

Inlet's optimal configuration determination consists of cone's angle and cowl lip's angle calculus, as well as the dimensionless architecture issuing, based on the determined angle(s). Similar algorithms, but for 2D (planar) inlets, were presented and applied in [8], [9], [12] and [18], while algorithms for 3D inlets' optimal configurations were described in [2], [4] and [8].

As optimization criterion one has chosen the inlet's total pressure recovery σ_i^* maximization. Inlet's total pressure recovery (also known as inlet's perfection coefficient, or inlet's total pressure loss coefficient) σ_i^* is given by

$$\sigma_i^* = \sigma_{csw1}^* \sigma_{csw2}^* \sigma_{nsw}^* \sigma_d^* , \tag{11}$$

where σ_{csw1}^* is the total pressure ratio for the conic shock-wave triggered by the centerbody, σ_{csw2}^* – total pressure ratio for the conic shock-wave triggered by the cowl lip, σ_{nsw}^* – total pressure ratio for the normal shock-wave and σ_d^* – total pressure ratio into intake's duct (assumed as constant, no matter the flight regime or the engine regime would be).

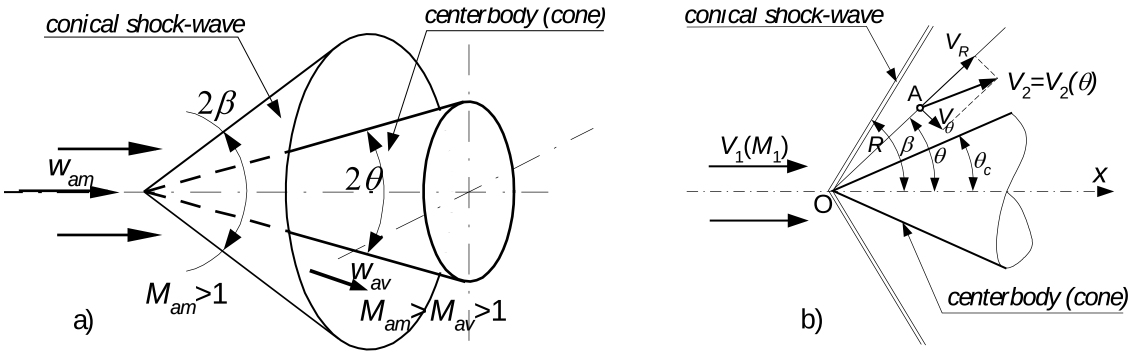


FIG. 4. Conical shock wave's geometry

The first and the most important issue of a conical shock wave (fig. 4) is the calculation of its angle β, with respect to the freestream Mach number M_1 (in front of the wave) and the cone angle θ_c. It might be calculated using an implicit non-linear equation (presented in [2] and in [5]):

$$\sin^2\beta = \frac{1}{M_1^2}\frac{1.2}{\cos\beta}\left(\frac{1}{\cos\beta} - \frac{1}{\cos\theta_c} + \ln\frac{\mathrm{tg}\frac{\beta}{2}}{\mathrm{tg}\frac{\theta_c}{2}}\right)^{-1} , \tag{12}$$

while the other parameters may be calculated very similar to the oblique shock-wave. Thus, the normal Mach number in the front of the wave M_{1n} is

$$M_{1n} = M_1 \sin\beta , \tag{13}$$

while the normal Mach number behind the wave M_{2n}:

$$M_{2n} = \sqrt{\left(0.4M_{1n}^2 + 2\right) \cdot \left(2.8M_{1n}^2 - 0.4\right)^{-1}} \;; \tag{14}$$

the tangent Mach number value remains the same before and behind the shock-wave , $M_{2t} = M_{1t}$:

$$M_{2t} = M_{1t} = M_1 \cos\beta \,, \tag{15}$$

so the Mach number behind the wave becomes

$$M_2 = \sqrt{M_{2n}^2 + M_{2t}^2} \;; \tag{16}$$

this Mach number will be the front Mach number for the next shock-wave.

Total pressure recovery coefficient becomes

$$\sigma_{csw}^* = \left[\frac{2.4M_{1n}^2}{2 + 0.4M_{1n}^2}\right]^{3.5} \left[\frac{2.4}{2.8M_{1n}^2 - 0.4}\right]^{2.5} , \tag{17}$$

which, obviously, depends on the values β and θ_c as long as $M_{1n} = M_1(\beta, \theta_c)$.

Inlet's optimal configuration is given by the situation when the external conical shock-wave are tangent to the cowl lip (point B in Fig. 3).

Cowl's lip angle is significantly smaller than centerbody's flare angle; it's value is a fraction k of centerbody's angle $(\theta_l = k \times \theta)$, usually equal to $(0.15 \div 0.6)$. The algorithm meant to inlet's optimization must determine the value of θ which assures the maximum value of total pressure recovery coefficient σ_i^*, when the air freestream's velocity is the maximum one, imposed by the aircraft's maximal flight regime.

If the aircraft has the engine(s) inside its body and the inlet(s) on the fuselage behind its nose, as far as aircraft's flight velocity corresponds to a Mach number $M_v = 4.5 \div 5$ aircraft's nose gene-rates a conical shock-wave, so the Mach number in front of engine's inlet is around 3.0÷3.5.

However, if the aircraft has a single engine and its inlet is mounted in the front of the fuselage, the front stream velocity is exactly the flight velocity.

For the present situation one has assumed that the air velocity is constant and it may be considered at an average value of the air free-stream, so the Mach number in front of the supersonic inlet is $M_H = 3.2$.

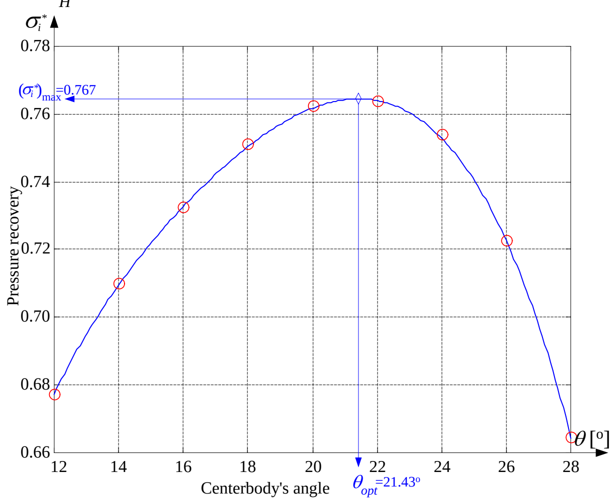


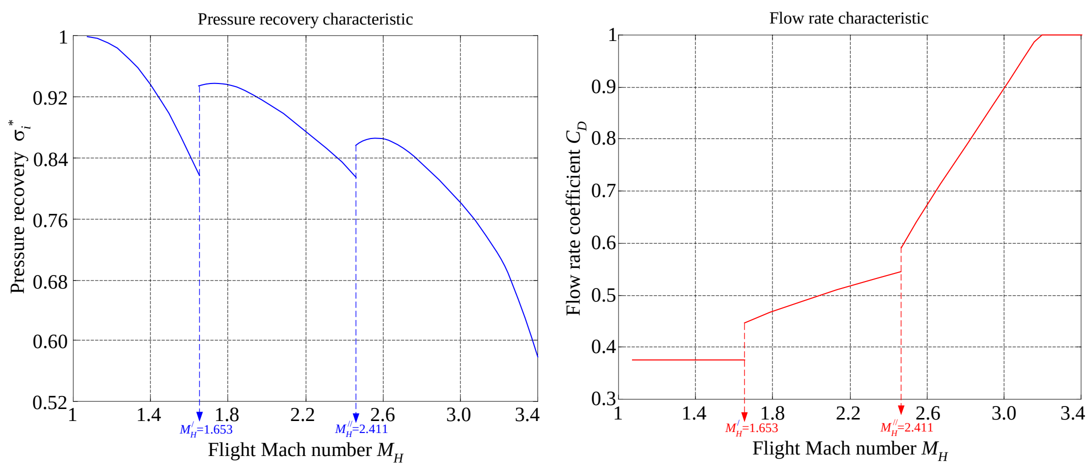
FIG. 5. Total pressure recovery coefficient versus the centerbody's angle

One has chosen $(\theta_l = 0.5 \times \theta)$ and considered a suitable interval for θ. Applying the algorithm in [17], using equations (11) to (17), one has obtained the dependence $\sigma_i^* = \sigma_i^*(\theta)$, as depicted in Fig. 5. The curve in this figure has a peak, a maximum value for σ_i^*, which corresponds to the optimal value of $\theta-$angle; this optimal value is $\theta_{\mathrm{opt}} = 21.43^\circ$.

Considering that cowl's lip B coordinate represents the unitary coordinate $(y_B = 1)$, one can determine inlet's dimensionless geometry. With the above determined optimal values of centerbody's and cowl lip's flare angle, one obtains the co-ordinates for the characteristic points in Fig. 3, as follows: A (0,0); B (1.536; 1); C (2.087; 0.819); D (2.087; 1.096). Moreover, based on these coordinates, the length of the center body's ramp results as: $l = 2.242$.

5. INLET CHARATERISTICS

Inlet's optimal architecture was determined for a Mach number when the external shock-wave is attached to the intake's cowl lip and the air flow rate through the inlet is maximum (the flow coefficient C_D is equal to 1); if the flight Mach number decreases, both the oblique shock-waves are depleting, so both β_1 and β_2 angles are growing, which means that air flow rate becomes smaller ($C_D < 1$). If the flight Mach number increases, the oblique shock-wave tends to enter inside the air intake, to interfere with the internal conic wave and to generate reflected shock waves, which will alter the flow and, consequently, the pressure and temperature distribution inside the inlet, which might make impossible the detonation.



a) Inlet pressure recovery characteristic map b) Inlet flow rate characteristic map

FIG. 6. Inlet's characteristic maps (fixed geometry architecture)

Characteristic charts are graphically presented in Fig. 6. The pressure recovery characteristic lays in Fig. 6.a), while flow rate characteristic – in Fig 6.b). It is noteworthy that both curves are not continuous, but they have some discontinuity points, corresponding to some occurred phenomena, such as shock-wave detaching: a) for Mach numbers under $M_H^{//} = 2.411$ the conic shock-wave triggered by the cowl lip detaches and becomes a normal – one, in front of the air intake, so the intake operates in a subsonic flow; b) for Mach numbers under $M_H^{/} = 1.653$ the conic shock-wave triggered by the centerbody detaches and becomes a normal – one, so the whole inlet operates in a subsonic flow.

6. INLET CONTROL LAW

In order to grow the C_D-value, a suitable solution is to keep the conical shock-wave attached to the cowl's lip, progressively displacing longitudinally the centerbody, which means that the inlet should be tuned with respect to the flight regime. As Fig. 3 shows, when the flight regime is less intense than the nominal-one, the conical shock-wave is depleting and moving away from the cowl's lip. Consequently, in order to bring back the conical wave on the cowl's lip, the distance $BB^/$ should be cancelled; it could be achieved only by retracting the centerbody. On the contrary, if the flight regime becomes more intense than the nominal-one, the centerbody should be pulled out of the intake (the distance $BB^/$ has become negative), to keep the conical shock-wave outside the intake.

This is, basically, the ground of the inlet's control law calculus, consisting of centerbody's displacement along its symmetry axis, with respect to the flight Mach number; Fig. 7 presents the shape of the control law.

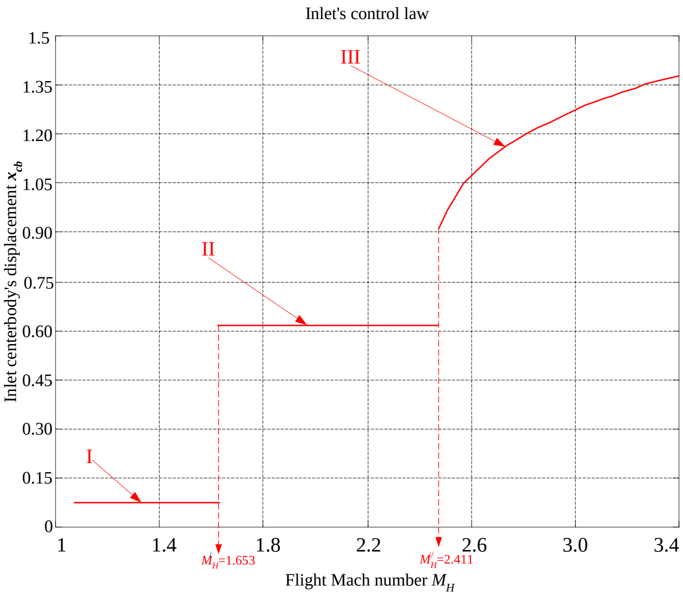


FIG. 7. Supersonic inlet's control law

The control law has three stages: **a) stage I**, corresponding to the low supersonic flight speeds, when the flight Mach number is under $M_H^/ = 1.653$ and the centerbody's tip triggers a detached normal shock-wave. Centerbody's position is fixed, the distance x_{cb} being constant $(x_{cb} = 0.172)$. In fact, this might be the centerbody's position even for subsonic flights; **b) stage II**, corresponding to the medium supersonic flight speeds, when the Mach number is between $M_H^/ = 1.653$ and $M_H^{//} = 2.411$. The centerbody's tip triggers a conical shock-wave, while the cowl lip triggers a detached normal shock-wave. Just as in the first stage, centerbody's position is fixed, the distance x_{cb} being constant $(x_{cb} = 1.364)$; **c) stage III**, corresponding to the high supersonic flight speeds, when the Mach number is bigger than $M_H^{//} = 2.411$ and both the centerbody and the cowl lip trigger conical shock-waves. The control law third slice is obviously non-linear; it might be mathematical described by the polynomial:

$$x_{cb}(M_H) = 0.0729 \times M_H^4 - 0.801 \times M_H^3 + 2.4167 \times M_H^2 - 2.9098 \times M_H + 1.511 . \tag{18}$$

However, the characteristics in figures 6, as well as the inlet's control law in fig. 7, correspond to the entire range of supersonic air velocities; this configuration is suitable for an inlet assisting a classical jet-engine (operating based on a Brayton cycle), with classical combustor and exhaust nozzle. Such combination inlet+engine might be useful for a classic (conventional) aerial vehicle.

Regarding a detonation engine, the condition in the detonation chamber, as well as in the burning and mixing zone, are totally different; the pressure, density and velocity conditions must be strictly met, as fig. 1 shows. According to the detonation limits (points a and b on the tangent line, curve S), pressure and density limits are established. However, the right side domain (a-point zone, point 4 of the ramjet cycle) are corresponding to weak detonation, therefore this zone is to be avoided. Considering the equation (10), which gives the detonation Mach number and the useful range of wave velocities, as well as the fact that, according to this range, its minimum value correspond to a free stream velocity of $M_{H_{\inf}} = 2.503$ (see point b), while the maximum value is $M_{H_{\sup}} = 3.315$ (see point C-J), one may determine the useful flight regime range. Therefore, the range of flight regimes becomes significantly narrower, between $\left[M_{H_{\inf}}; M_{H_{\sup}}\right]$, which narrows the useful adjustment range, restraining the control law to a part of its third stage, as fig. 8 shows. This part is nonlinear and covers a significantly reduced distance of spike movement (around 0.32).

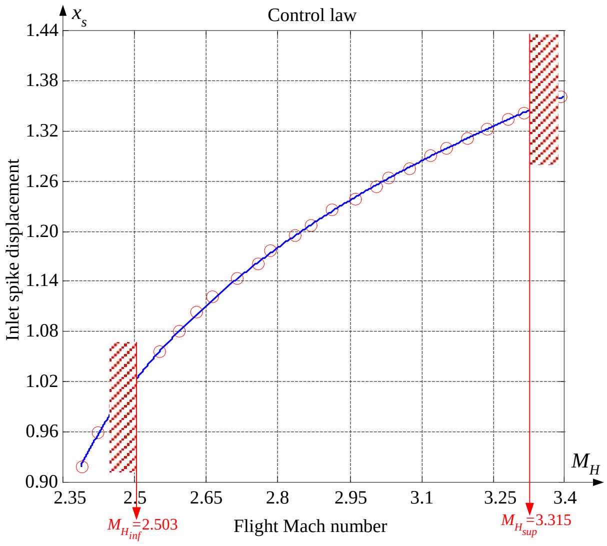


FIG. 8. Inlet's effective control law

7. CONCLUSIONS

High speed flights' nowadays challenge is how to obtain enough thrust with less constructive and operational efforts. Among the current modern solutions to this issue, detonation engines are at the forefront, but still in their infancy. This kind of engine needs special thermo-hydro-dynamic conditions of pressure, air density and velocity inside the detonation chamber, no matter the flight regime, which are very difficult to met, the stabilisation of detonation phenomena being a very challenging task.

Present work has determined one of these conditions: the speed limits of a supersonic propulsion system, based on its inlet optimal design, in order to assure the above-mentioned conditions for the detonation engine (established and given by the Chapman-Jouguet equations and Hugoniot curves); one has determined the domain of existence of the continuous detonation inside engine's combustion room, from air pressure and density points of view, as well as the velocity condition.

Inlet's optimal architecture issuing was performed based on inlet's total pressure recovery coefficient maximization algorithm. Inlet's flow rate characteristics and the extended control law (as if the inlet would operate in the entire supersonic flight speeds interval) were issued. Control law has some discontinuity points, corresponding to the critical regimes, when the shock-waves are to be detached. The last part of the control law is a strong nonlinear and continuously growing with the flight regime.

Overlapping the curve of variation of air parameters behind inlet's shock waves one has obtained the limits of the flight regime which makes possible the detonation.

One can also conclude that such a propulsion system based on detonation burning is possible only for high supersonic flight speeds, but an aerial vehicle equipped with such a system cannot take-off and it should be firstly carried out and launched at a suitable flight regime (altitude and speed).

REFERENCES

[1] I. Aron and A. N. Tudosie, Hydro-mechanical system for the supersonic air inlet's channel's section control, in *Proceedings of the International Conference on Applied and Theoretical Electricity*, ISBN-973-657-005-3, Craiova, 2000, pp 266-269.

[2] C. Berbente and N. V. Constantinescu, *Gases Dynamics*, vol. I, II. Politehnica University in Bucharest Inprint, 1985.

[3] F. Falempin, Continuous detonation wave engine, in *Advances on Propulsion Technology for High-Speed Aircraft*, 2008, pp. 8-1 – 8-16.

[4] Y. P. Goonko and E. Alexandrov, Aerodynamic design of a supersonic three-dimensional inlet, in *Thermophysics and Aeromechanics*, Vol. 1, Nr. 17, 2010, pp. 57-68.

[5] P. G. Hill, C. Peterson, *Mechanics and Thermodinamics of Propulsion*. Addison - Wesley Publications, New York, 1993.

[6] M. A. Libermann, *Introduction to Physics and Chemistry of Combustion. Explosion, Flame, Detonation.* Springer Verlag, Berlin Heidelberg, 2008, pp. 360.

[7] V. Pimsner, *Air-breathing Jet Engines. Processes and Characteristics*, Bucharest, Didactic and Pedagogic Publishing, 1983.

[8] V. Pimsner, C. Berbente, *Flows in turbomachinery*, Tehnica Publishing Bucharest, 1996, ISBN 973-165-4-17, pp 384.

[9] H. Ran and D. Mavris, Preliminary Design of a 2D Supersonic Inlet to Maximize Total Pressure Recovery, in *Proceedings of AIAA 5th Aviation, Technology, Integration and Operations Conference* (ATIO), 26-28 Sept., Arlington, Virginia, 2005, pp. 1-11.

[10]C. Rotaru, I. R. Edu, M. Andres-Mihaila and P. Matei, Applications of multivariable control techniques to aircraft gas turbine engines, in *Review of Air Force Academy*, no.2 (26), 2014, pp. 45-50.

[11]C. Rotaru and G. Raducanu, Analytical approaches of detonation waves, in *Review of the Air Force Academy*, No.3 (35), DOI: 10.19062/1842-9238.2017.15.3.6, Brasov, 2017, pp. 57-64.

[12]J. J. Seddon and E. L. Goldsmith, *Intake Aerodynamics*, 2nd edition, AIAA Education Series, 1999.

[13]K. Terao, *Irreversible Phenomena. Ignitions, Combustion and Detonation Waves*, Springer Verlag, 2007, pp. 409.

[14]A. N. Tudosie, Supersonic Air Inlet's Control System Based On The Inner Normal Shock Wave's Position Stabilisation in *Proceedings of the International Conference on Military Technologies* ICMT 07 (ISBN: 978-80-7231-238-2), University of Defense, Brno, 2-4 May 2007, pp 342-349.

[15]A. N. Tudosie, Supersonic Air Inlet Control System Based On The Inner Channel Minimum Cross-Section Position Control in *Proceedings of the International Conference on Military Technologies* ICMT 09 (ISBN: 978-80-7231-648-9), University of Defense, Brno, 5-7 May 2009, pp 350-359.

[16]A. N. Tudosie, Control laws for an aircraft supersonic inlet with mobile panel, in *Proceedings of International Conference Scientific Research And Education In The Air Force* –AFASES 2017, Brasov, 22-24 May 2017, pp. 231-242.

[17]A. N. Tudosie, Pneumo-hydro-mechanical control system for an aircraft supersonic inlet with mobile ramp, in *Review of Air Force Academy*, no.1 (33), 2017, pp. 123-130.

[18] A. N. Tudosie, Control Law for the Air Inlet of a High Supersonic Velocity Aerial Vehicle, in *Proceedings of International Conference on Control, Artificial Intelligence, Robotics and Optimization,* DOI: 10.1109/ICCAIRO46080.2018, Prague, 19-21 May, 2018, pp. 6-13.

[19]P. Wolanski, Detonation Engines in *Journal of KONES Powertrain and Transport*, Vol. 18, No. 3, 2011, pp. 515-521.

[20]S. Yetao, L. Meng and W. Jianping, Continuous detonation engine and effects of different types of nozzle on its propulsion performance, in *Chinese Journal of Aeronautics,* No. 23, 2010, pp. 647-652.

[21]L. Yang, W. Yuhui, W. Jianping, L. Yongsheng, Detonation instability of continuously rotating detonation engines for H2-air mixture in *International Invention Journal of Engineering Science and Technology* Vol. 1(1), June, 2014, pp. 1-7.

[22] V. Prisacariu, C. Rotaru, M.L. Niculescu, *Considerations and simulations about Pulse Detonation Engine*, MATEC Web of Conferences 2019, vol. 290, p.4009, EDP Sciences - Web of Conferences, eISSN 261-236X, https://doi.org/10.1051 /matecconf/201929004009.