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FLOW PROCESSES IN OVEREXPANDED CHEMICAL ROCKET NOZZLES
PART 3:
METHODS FOR THE AIMED FLOW SEPARATION AND SIDE LOAD REDUCTION

R. H. Schmucker

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16. Abstract Methods aimed at reduction of overexpansion and side load resulting from asymmetric flow separation for rocket nozzles with a high opening ratio are described. The methods employ additional measures for nozzles with a fixed opening ratio. The flow separation can be controlled by several types of nozzle inserts, the properties of which are discussed. Side loads and overexpansion can be reduced by adapting the shape of the nozzle and taking other additional measures for controlled separation of the boundary layer, such as trip wires.

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List of Symbols

1. Latin letters

A	Cross-section
d	Diameter
F	Force
K	Constant
l	Length
m	Flow rate
M	Mach number
p	Pressure
r	Radius
t	Time
T	Temperature

2. Greek letters

α	Mass flow exponent
β	Longitudinal effects exponent
γ	Isotropic exponent
δ	Boundary layer thickness
ρ	Thickness/density

3. Indices

a	External
abl	Ablation
c	Combustion chamber
foa	Foam insert
i	"Flow separation" (Pressure increase for flow separation)
l	Local nozzle point
ma	Axis deviation
ot	Without trip wire
sl	side load
t	Collar
tw	Trip wire
w	Wall
wt	With trip wire

0. Summary

Strong reductions of overexpansion occur in operation of nozzles with a wide opening ratio on the ground and asymmetric flow separation and side load during the start-up phase. Various measures can be employed to avoid these unwated phenomena. A description is offered of several methods that can be used for a nozzle with a fixed opening ratio. An outline of various theoretical and experimental results is provided to facilitate asessment of the methods.

FLOW PROCESSES IN OVEREXPANDED CHEMICAL ROCKET NOZZLES, PART 3: METHODS FOR THE AIMED FLOW SEPARATION AND SIDE LOAD REDUCTION

R. H. Schmucker

1. Introduction /1*

The performance of a rocket engine operated in the atmosphere depends on the opening ratio of the nozzle and on the difference between the nozzle-end and external pressure. The specific pulse increases in vacuum with increasing expansion ratio and, for that reason, the nozzle for a rocket to be operated in space is determined so as to achieve maximum output through taking into consideration weight and dimensional increases due to an expanded opening ratio. If such a rocket engine is operated on the ground, the specific pulse is diminished due to overexpansion and there can set in a flow separation accompanied by side loads.

Various methods can be used to reduce these unwanted phenomena, i.e., too strong overexpansion, uncontrolled flow separation and side loads. One of these methods involves the use of a nozzle with a variable opening ratio at a constant collar cross-section. At a low pressure ratio (combustion chamber pressure/ambient pressure) the nozzle is operated with a small opening ratio, in vacuum use is made of the maximum opening ratio. The solutions successfully tested in this experiment include an adjustable fixed (cooled) nozzle extension (extensible nozzle |7|)and a flexible nozzle extension (|17|) that can be blown up by exhaust cases from the turbine. However, these methods are very costly.

Thus, a feasible approach involves selection of methods that make it possible to achieve in a relatively simple manner through the use of additional measures a controlled flow separation and a reduced side lcad in a nozzle with a fixed opening ratio. In this case the duration of use depends on the purpose for which it is used.

*Numbers in the margin indicate pagination in the foreign text.

If side loads are to be reduced only during strat-up, the duration of additional measures is limited to several seconds. If, however, /2 it is necessary to reduce overexpansion during certain phases of the flight and to reduced side loads during partially throttled operation, controlled flow separation is required for an extended period.

2. Reduction of the Opening Ratio Through a Nozzle Insert /3

Use of a nozzle insert reduces the opening ratio of the nozzle and increases the wall pressures. This produces a flow separation at a lower pressure ratio and side loads become reduced.

2.1 Central Body Insert

A method often proposed for preventing flow separation at a high expansion ratio is the use of a central body insert in the nozzle. This method can be used only on the ground (start-up phase, test operation), but not in flight.

Various forms of central insert design are shown in Figure 1. /4 They include:

--a simple conical or rounded body with thrust compression,

--a pointed body with isentropic compression,

--gaseous or liquid body with a reverse flow injection into the nozzle.

To increase the wall pressure near the end of the nozzle, the body must reach deep into the nozzle, as the compression lines from the central body (in extreme cases compression impacts) must skirt the wall. Consequently, the dimensions of such a body are very large. After completing its role, the insert is pulled out of the nozzle.

The problems attendant to the use of this method are

--protection of the body against heat

--increase in heat transfer on the nozzle wall due to compression impacts

--centering.

/3

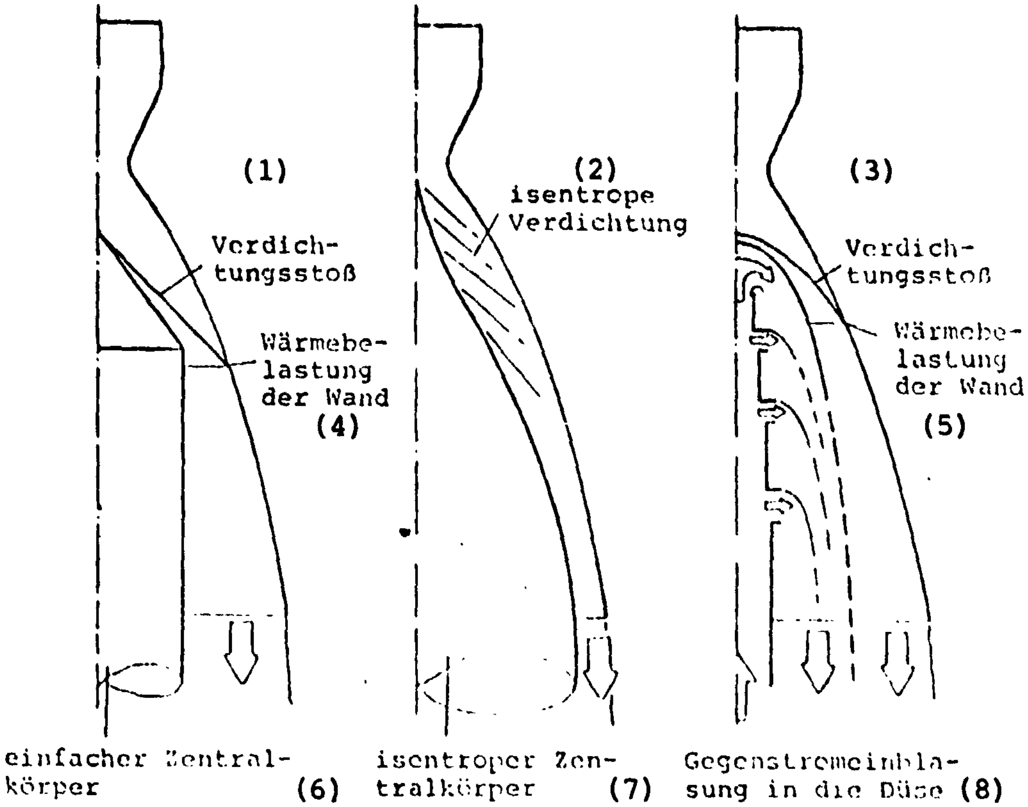

Figure 1. Design Forms for A Central Body Insert

Key: (1) Compression impact; (2) Isentropic compression; (3) = (1); 4) Heat stress on wall; (5) = (4); (6) Simple central body; (7) Isentropic central body; (8) Reverse flow injection into the nozzle.

/4

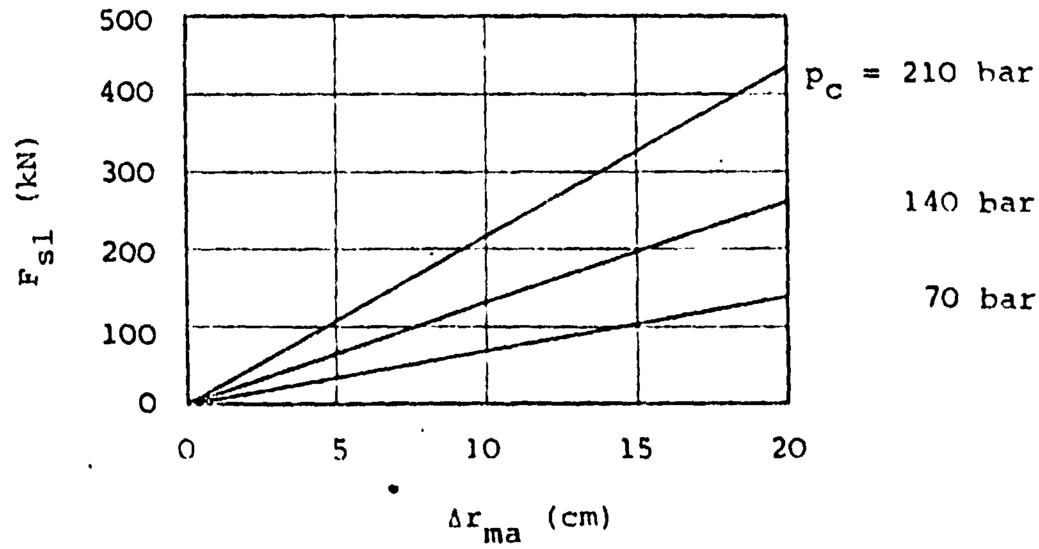

Figure 2. Side Loads Caused by Faulty Centering of the Central Body Insert in a Nozzle (Impact Angle 40°, Nozzle Diameter 1.6 m) |4|

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Figure 2 shows the side loads produced by deviation of the central body axis from the nozzle axis Δr_{ma}. Even slight centering mistakes produces side loads on the order of natural side forces.

Thus, in spite of its essentially great simplicity, this method is used only in experimental stages, because of the detailed technical problems it poses. /5

2.2 Nozzle Lining

In order to get around the problem of centering the central body insert, the nozzle can be lined with a suitable material. Figure 3 shows two potential design forms.

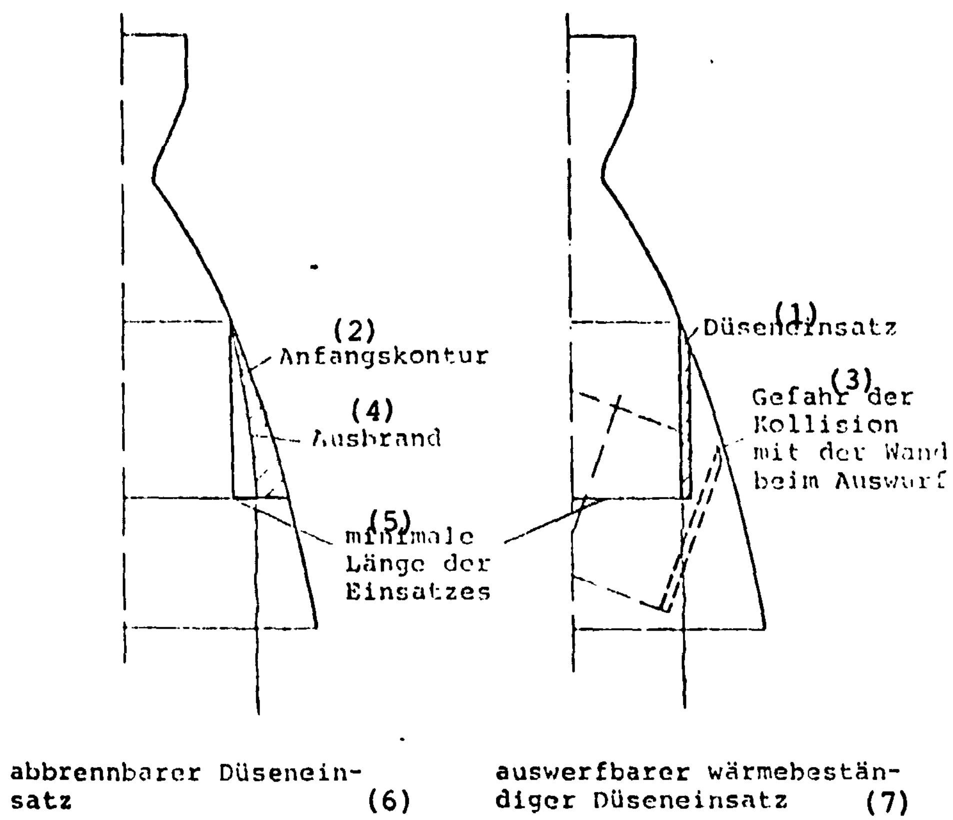


Figure3. Design Forms for a Nozzle Lining

Key: (1) Nozzle insert; (2) Initial contour; (3) Danger of collision with the wall during ejection; (4) Combustion; (5) Minimum length of insert; (6) Combustible nozzle insert; (7) Ejectable heat-resistant nozzle insert.

2.2.1 Heat-resistant Nozzle Insert

A heat-resistant nozzle insert can be used for both reduction of side loads and controlled flow separation during flight |1|. After fulfilling its function the insert is ejected. The length of the lining must be selected so that under unfavorable condi- /6 tions the stream of exhaust gases would not again come into contact with the nozzle wall. The problems attendant to this arrangement are

-material

--method of ejection.

If the separation is uneven, the insert can revolve in the stream of exhaust gases and collide with the wall. NASA-MSFC tests established the fact that even minute, light particles that come into contact with the wall of the nozzle can produce considerable damage (see section 2.2.2).

2.2.2 Combustible Nozzle Insert

The materials that can be used as a combustible nozzle insert are

--a solid fuel |10|

--an ablation material (plastic foam)

Selection of the material depends on the required duration of use and on the dimensions of the nozzle.

To avoid side loads during start-up phase, the combustion velocity must be very high, as the ablation time is on the order of only several seconds. The combustion velocity r depends hereby on the gasification heat of the ablation material, the heat flow against the wall and the density of the material ρ_{foa}. In simplified form, for any given rocket and a given material at any moment applies |11|

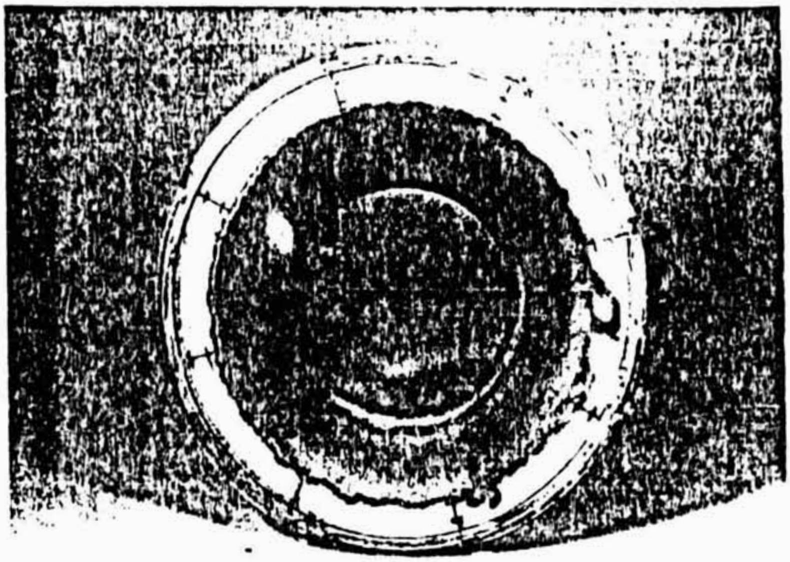
$$\dot{r}\,\rho_{foa} = \text{const.} \qquad (1)$$

The constant depends hereby on the mass flow in the nozzle (flow rate per cross-section area), the flame enthalpy and the gasification point of the insert.

It is recognized that materials of low density must be used for high ablation velocities.

NASA-MSFC rocket tests are conducted for that purpose with /7 nozzle inserts made out of plastic foam. Figure 4 shows a plastic foam insert that was partially combusted in a model rocket. The ablation rate in circumference is very even, ablation in longitudinal direction leads to a bell-shaped combustion. The combustion velocity can be approximatively determined from reduced foam density at the end of the nozzle. Figure 5 shows the measuring points of a film analysis.

Figure 4. Combusted foam insert in the nozzle of a LOX/H_2 model rocket (NASA-MSFC) →



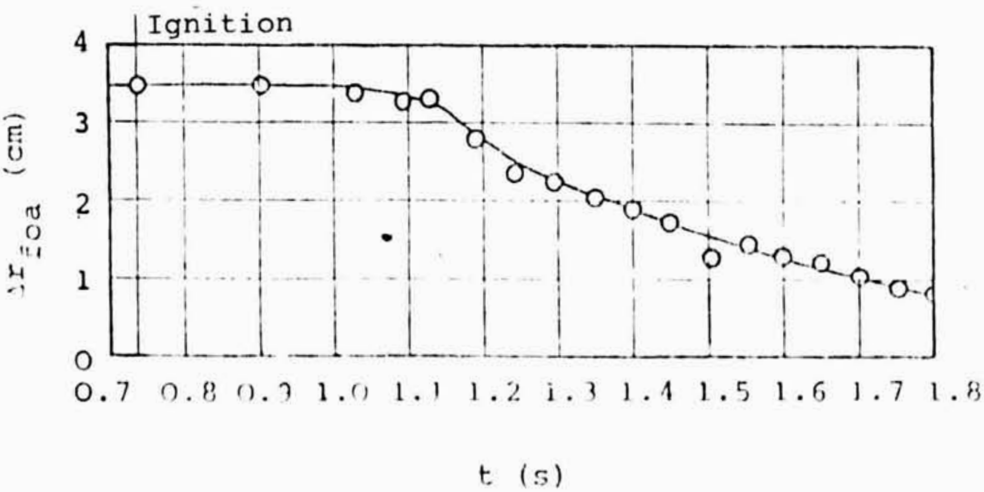


Figure 5. Wall thickness Δr_{foa} of the foam insert at the end of a nozzle as the function of test duration (NASA-MSFC test 268-015)

The combustion burn-off starts at the upper end of the insert, /8 so that no ablation can be determined at the nozzle end cross-section. For this reason this period of time must be considered to be a sort of an initiation process.

Transfer of the experimental model data to a large rocket (J-2) calls for knowledge of regularity of combustibility. If these processes are viewed, for the sake if simplicity, as stationary and if it is assumed that ablation of foam is similar to ablation in a hybrid rocket (the combustion velocity is controlled by heat transfer), then applies that |11|

$$\dot{r} = \frac{K_{abl1}}{\rho_{foa}} \left(\frac{\dot{m}}{A_1}\right)^{\alpha} l^{\beta} \qquad (2)$$

Therein $\dot{m}$ denotes mass flow rate through the nozzle, A_1 the local nozzle cross-section, l the longitudinal coordinate of the insert which is measured from the starting edge downstream, and K_{abl} is the combustibility constant of the foam, α and β describe the mass flow and length effects. Theory yields for β a negative value (-0.2). This tendency has been confirmed in the experiment by stronger burn-off at the beginning of the insert.

If we substitute for mass flow rate in (2) the combustion chamber pressure and introduce the characteristic velocity in K_{abl1}, we then can write, with collar cross-section A_t

$$\dot{r} = \frac{K_{abl2}}{\rho_{foa}} \left(\frac{p_c}{A_1}\right)^{\alpha} A_t^{\alpha} \; l^{\beta} \qquad (3)$$

If we use equation (3) for assessment of the tests we obtain the combustion constants. In figure 6 the combustion velocity of the experiment of Figure 5 is plotted over the expression $p_c/d_1{}^2$, d_1 being in this case the local nozzle diameter.

The experiments showed a very strong effect of mass velocity (Test 268-015:1.6th potency). Nevertheless, the disparity between experiments using approximately same materials was so great that /9 only reference values for ablation velocity could be estimated.

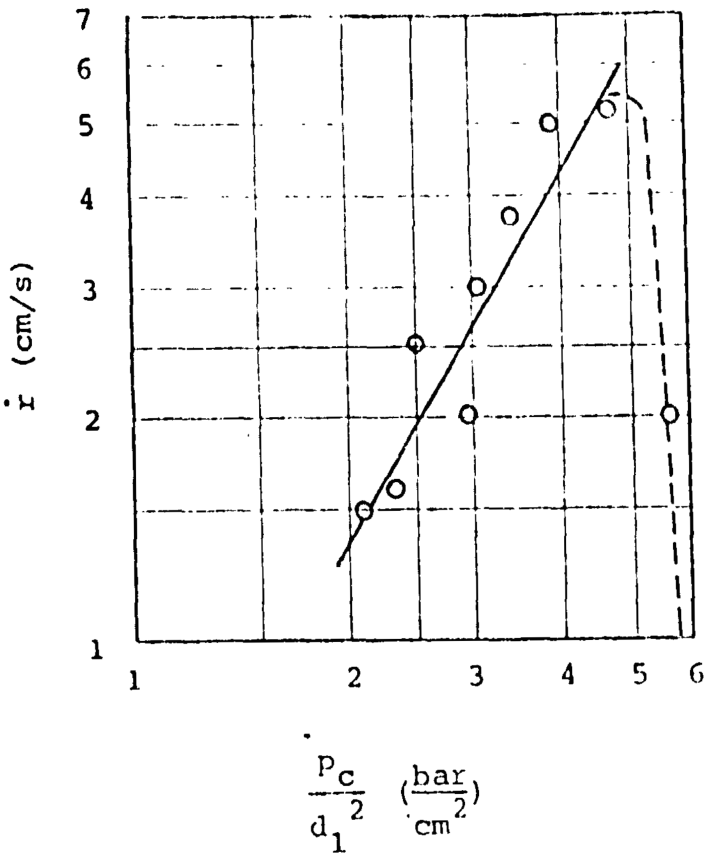


Figure 6. Combustion velocity over mass flow p_c/d_1^2 (Test [illegible]8-015)

The model findings can be transferred to large rockets with the aid of equation (3). If we designate the local opening ratio with ε we obtain

$$\dot{r} = \frac{K_{abl2}}{\rho_{foa}} p_c^{\alpha} \varepsilon^{-\alpha} l^{\beta} \tag{4}$$

If the combustion chamber pressure and opening ratio are identical, then the regression velocity is lower due to greater length. In a large rocket the ablation must consequently last considerably longer than in a model nozzle.

Figure 7 shows a J-2 rocket with a foam insert. NASA-MSFC experiments have shown that almost all side loads can be eliminated in this manner, but in accordance with equation (4) the duration of ablation is far too hich. In a maximum operation ti e of 12 s (started with an[10] uncooled flame deflector) only a part of the insert became ablated.

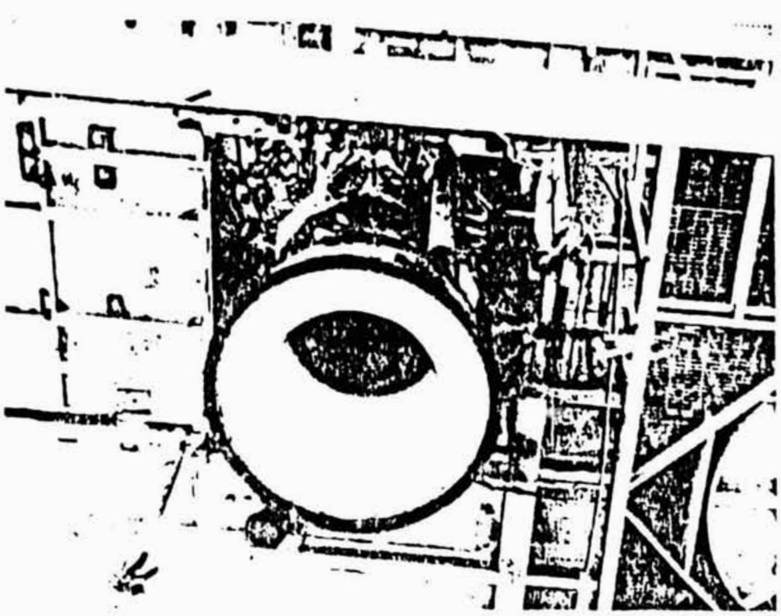


Figure 7. Plastic Foam Insert in the Nozzle of a J-2 Rocket (NASA-MSFC)

Thus, some 10 s would have to be considered for total combustion time. However, in almost all cases the strength of the material was not adequate enough to prevent destruction of the plastic foam. However, this problem will be hard to solve. The insert in the J-2 nozzle is affected by a force which in axial direction amounts to roughly 100 kN. Increased strength of the foam can be achieved only through increased density which, however, would further reduce the combustion velocity.

In order to make these methods suitable for use, additional measures must be resorted to, such as length shortening (see section 3.2) or reinforcement through a fiber matrix.

3. Reduction of Separation Flow Point Symmetries /11

With a suitable shape of the nozzle and with the aid of additional measures for boundary layer separation flow it is possible through reduction of separation flow point symmetries to reduce the side loads and overexpansion.

3.1 Nozzle With a Steep Pressure Gradient

A nozzle with a steep pressure gradient can be used primarily for reducing side loads. Herein at very high opening angles can be observeda decrease in overexpansion |2|. However, the latter effect has little significance, particularly from the viewpoint of rocket performance.

According to |6 |the following applies to side load:

$$F_{sl} \sim \frac{p_a}{p_c} \frac{1}{\dfrac{d(p_w/p_c)}{d(1/r_t)}} \qquad (5)$$

Thus, side load due to asymmetrical flow separation depends on the pressure distribution of combustion chamber pressure p_c in its ratio to ambient pressure p_a and, essentially, on the gradient of wall pressure p_w along the nozzle axis 1. r_t denotes the collar radius. If the pressure gradient is increased, the side loads are reduced. Almost all shunt loads can be practically eliminated in this manner |16|.

The pressure gradient is affected by the opening angle of the nozzle. At an identical pressure slope the pressure gradient increases with increasing divergence angle at reduced nozzle length. Additionally, in the case of a bell-shaped nozzle there is a need for flattening of the wall contour (increased wall curvature radius toward the end of the nozzle)) (J-2S nozzle type), , as otherwise a /12 flatter pressure gradient is achieved at increasing inward curvature (J-2D, Atlas Sustainer nozzle type). For this reason the Space Shuttle 's main rocket uses a contour similar to that of the J-2S.

An increase in the median divergence angle reduces additionally the nozzle mass. This, however, leads to higher divergence losses which diminish output |5|. Thus, the length of the nozzle must be determined so that the advantage offered by mass reduction through lower side loads and a shorter nozzle is not dissipated through a decrease in output.

3.2 Methods for Boundary Layer Flow Separation

Sufficiently strong interference with the boundary layer occurring in ambient pressure can force the flow to separate. In this manner it is possible to determine the point of separation within a narrow range with preceision and, thus, reduce the side loads. Additionally, overexpansion can be reduced as well.

Figure 8 shows the various methods for achieving flow separation. The following measures are available:

--secondary injection,
--discontinuity of the wall contour angle,
--annular groove,
--trip wire,

/13

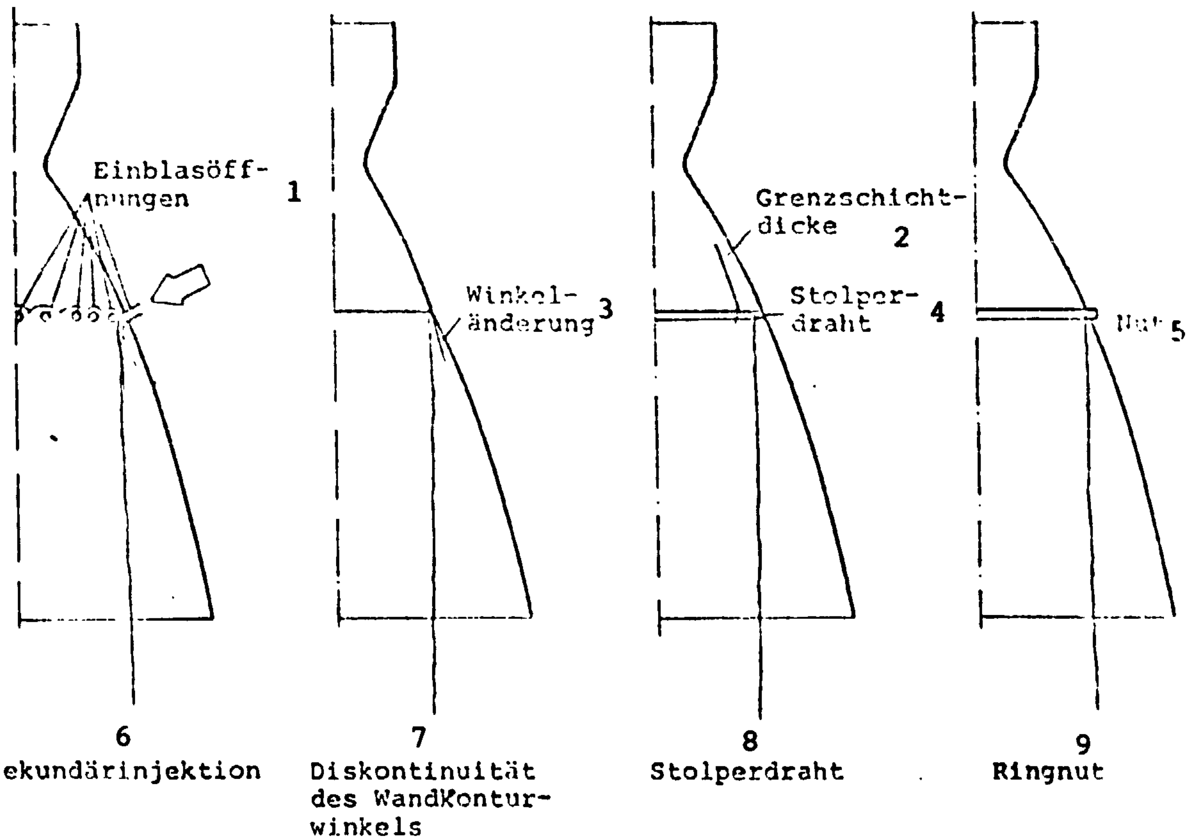


Figure 8. Methods for Intended Boundary Layer Flow Separation (Key provided on next page)

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Key to Figure 8: 1 - Injection ports; 2 - Boundary layer thickness; 3 - Change in angle; 4 - Trip wire; 5 - Groove; 6 - Secondary injection; 7 - Discontinuity of the wall contour angle; 8 - Trip wire; 9 - Annular groove.

3.2.1 Secondary Injection /12

Secondary injection for asymmetric flow separation with shear vector control is one method used in many rocket power units. It makes it possible to achieve through symmetric secondary injection an axially symmetric flow separation line. This can be accomplished in various ways:

--injection of a foreign medium (gas, liquid), /13

--suction of surrounding air |2, 9|.

This method for flow separation offers the advantage of simple switching on and off through control of the secondary current. Its disadvantage is a higher demand on structural design.

3.2.2 Discontinuity of the Wall Contour Angle |2|

Discontinuity of the wall contour angle leads at the point of change in angle to a steep change in the nozzle's wall pressure. An increase in the divergence angle produces a reduction in pressure. Consequently, flow separates at the wall bend at corresponding combustion chamber pressure. Only an increase in the pressure ratio corresponding to the jump in pressure results in jumping over the discontinuity.

This method can probably be used with advantage only for reduc- /14 tion of overexpansion, but not for deacreasing side loads. The reason being that after jumping over the discontinuity the flow separates further downstream naturally with the thereby connected side loads (the conditions are different from those in the case of trip wire that has a certain effective area).

Nothing has been published so far about experimental application of this method.

3.2.3 Annular Groove

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An annular groove can be thought of only as a counterpart to a trip wire. This type of affecting of the boundary layer can be implemented only with difficulty in a nozzle structure consisting of thin-walled tubes.

3.2.4 Trip Wire

Trip wires have been proposed |6, 8, 9| for separation of an overexpanded flow. This method offers the advantage that the requisite flow separation mechanism can be added after development of the rocket without any changes in its configuration.

The trip wire interferes with the development of a boundary layer so strongly that the flow separates at a higher wall pressure that without such an interference point. Thus, the point of separation is upstream from the natural point of flow separation. Figure 9 shows a nozzle with tubular structure and 3 trip wires. At a certain pressure ratio the flow separates at the first wire. If the combustion chamber pressure is increased (or if there occurs a reduction in ambient pressure), there is no immediate change in the point of separation, until at a certain higher pressure ratio the flow crosses the wire and separates at the next wire located downstream. This process is repeated so long till it reaches the end of the nozzle. The nozzle end can hereby be also considered as a wire (of vanishing strength).

For this arrangement to be effective, the wires in their dimen- /15 sions and their positions in the nozzle must be mutually balanced. Even though this method of multiple wires or of a "continuous wire along the nozzle wall" appears to offer great advantage, use of only one or only a few wires is in order, because each trip wire leads to reduced output and interferes with the field of flow.

This method was examined in closer detail in NASA-MSFC. The 4-k LOX/H_2 model rocket and a J-2 rocket were used as test engines.

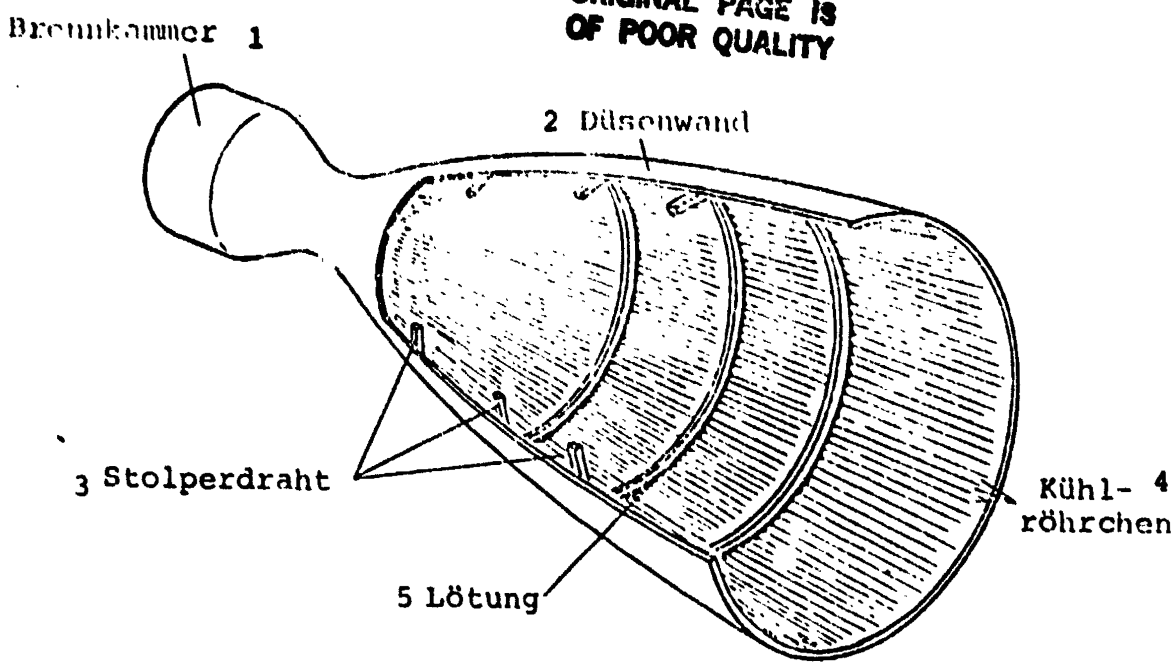

Figure 9. Trip Wires in a Nozzle With Tubular Structure (as in J-2 Experiments)

Key: 1 - Combustion chamber; 2 - Nozzle wall; 3 - Trip wire; 4 - Cooling tubes; 5 - Brazing.

A simplified equation for the separation pressure in the use of a trip wire is |13|

$$\frac{p_{i_{wt}}}{p_a} = \frac{p_{i_{ot}}}{p_a} \frac{1}{1 - 0.16\, M_i^2 \frac{d_{tw}}{\delta_i}} \qquad (6)$$

Therein the index wt denotes the trip wire, ot stands for separation without wire, p_i represents pressure at which separation occurs, M_i /16 is the Mach number at wire points, d_{tw} denotes the wire diameter and δ_i is the local boundary layer strength. The constant 0.16 has a limited dependence on the gas composition, the temperature distribution in the boundary layer and the Mach number. The pressure $p_{i_{ot}}$ can be derived from |15|.

Figure 10 shows flow separation pressures for a model rocket with trip wires of varying strength. Separation occurs earlier with increasing wire strength.

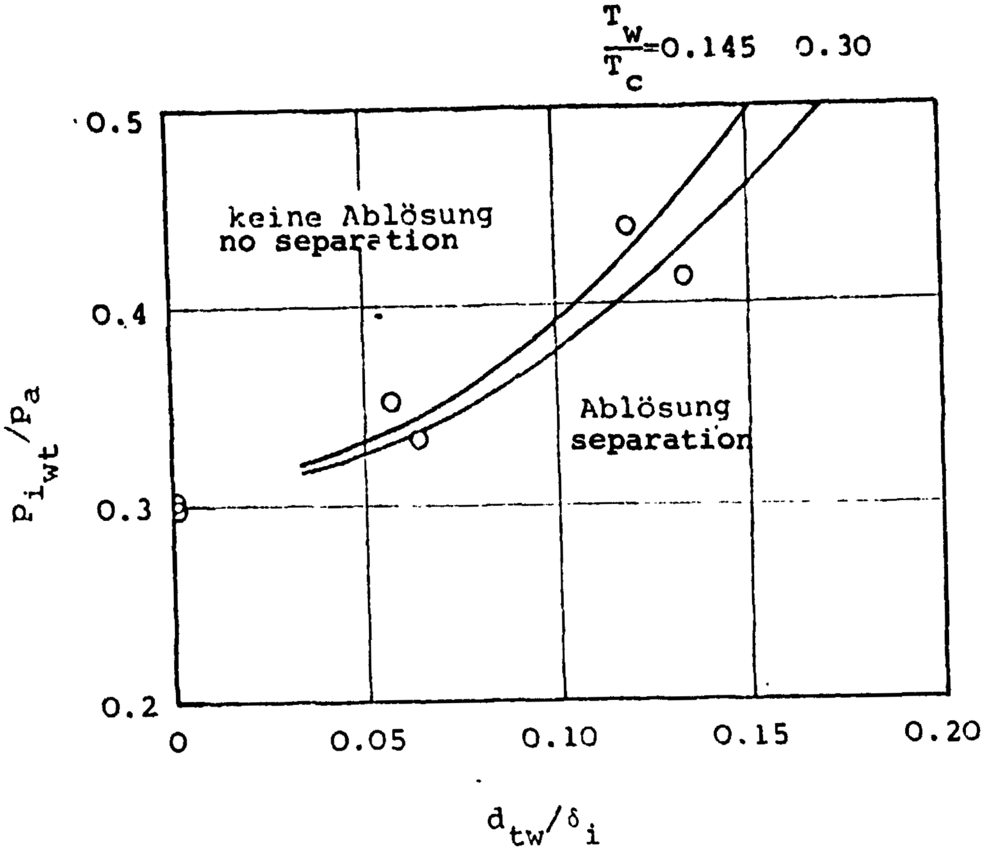

Figure 10. Measured and Computed Separation Pressure Conditions of the NASA-MSFC 4-k LOX/H_2 Model Rocket (M_i = 3.6, γ = 1.26)

o Test data

A comparison of separation measurements in various rockets with the theory according to |13| is shown in Figure 11. It shows a relatively good correspondence between theory and experiment. Deviations are probably due to gross simplifications in theory. /17

Experiments show that trip wires of approximately 1/10 of the local boundary layer strength are sufficient to induce separation of flow.

The efficiency loss of a wire is less than 0.2% for the specific pulse |3|. Of course, this value depends on wire strength. Loss of efficiency is affected by the wire's drag resistance (approximately 50%) and the wall pressure reduction downstream of the wire.

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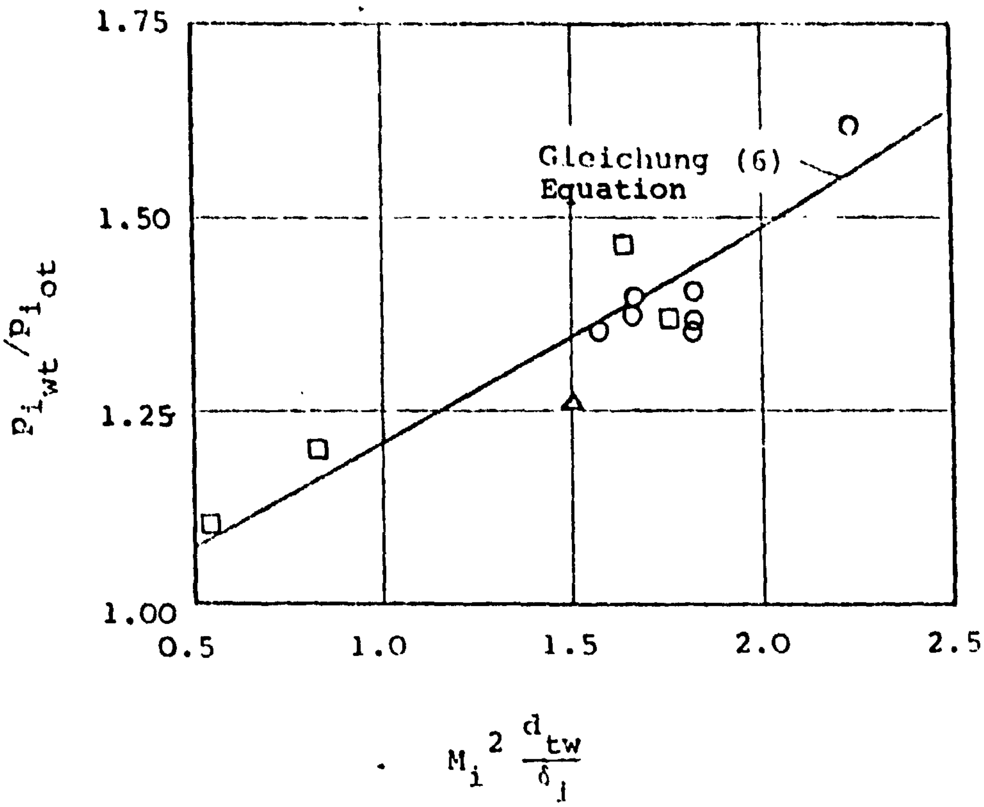

Figure 11. Comparison Between Experimental and Computed Separation Data 14

o 4-k LOX/H_2 Model Rocket (NASA-MSFC)
□ J-2D Rocket (NASA-MSFC)
Δ J-2S Rocket |3|

The measurements showed a differing behavior in regards to reduction of side load. While in NASA-MSFC an almost continuous reduction could be achieved, tests with rocketdyne had a less positive progress. Precise positioning of the wires has a favorable effect /18 on side load reduction. However, problems are encountered during wire crossing. Flow asymmetries and fluctuations produce an asymmetric crossing of the trip wire, so that high side loads can be produced temporarily. Fim analyses showed this asymmetric behavior. Avoiding these effects called for shortening the distances between the wires and for a rapid build up of combustion chamber pressure so that the flow could very expediently reach in leaps the nozzle end cross-section.

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