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INVESTIGATION OF AN ADDITIONAL OXIDIZER CHARGE EFFECT ON SELECTED OPERATIONAL CHARACTERISTICS OF A SOLID-FUEL ROCKET ENGINE

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The paper has been intended to discuss some issues closely related to the elimination of an extremely disadvantageous phenomenon, i.e. the occurrence of a negative oxygen balance in products of combustion emitted by a solid-fuel rocket engine while launching a missile from the aircraft. Findings of experimental examination of such engines furnished with an additional oxidizer charge have also been presented.

Key words: oxygen balance, rocket engine, aircraft, experimental examination, oxidizer charge

1. Introduction

Intensive development of air armament technology enforces integration of newly designed aircraft weapons with those that have already been in service. It always requires a series of tests to be carried out to confirm effectiveness and operational safety of the weapon system in question.

One of numerous characteristics under examination, extremely important to air-launched missiles, is how they affect the carrier. It is required that the air-launched missile do not disturb the carrier's flight and have no adverse impact on the operation of any of on-board systems (Kurow and Dołżański, 1964; Stieczkin *et al.*, 1961). Among the most frequent undesirable effects of launching a missile from the carrier, is the stream of the rocket engine emitted exhaust gas that affects operation of the aircraft turbine engine, which may even lead to the engine stall/surge (Gajewski *et al.*, 1980; Kowalski *et al.*, 2011). This effect was observed with missiles powered by rocket engines supplied with high-energy solid fuels displaying negative oxygen balance. Combustion products are, in this case, rich in incompletely burnt particles of carbon and hydrogen; what is more, their temperature is ca. 926.8°C (Leciejewski, 2010). In consequence, these particles are rapidly burnt out in the air just behind the exhaust nozzle, which causes oxygen deficit behind the moving missile (Torecki, 1989). Releasing the missiles in series from the launch aircraft generates hot oxygen-free air region propagating behind the missiles, which may lead to surge or flameout of the carrier's engine (Nechaev *et al.*, 1980; [14]). To prevent the engine surge, efforts were initiated to modify the missile-dedicated rocket engine. However, first and foremost for economic reasons, only slight design alterations proved permissible, ones that would not change aerodynamic and ballistic characteristics of the missile. Therefore, a decision was made to embark on studies intended to modify the oxygen balance of exhaust gas discharged from the rocket engine, however, with no modifications to the powder-pulp composition.

2. Theoretical analysis

One of the methods of altering the oxygen balance of combustion products from the rocket--propellant decomposition consists in the insertion of an additional portion of the oxidizer

right into the combustion chamber. This method allows for some changes in energetic values/characteristics of combustion products from the rocket engine (temperature) with no modification in design of major structural components (assemblies, parts) of the rocket such as the combustion chamber, the nozzle, or the igniter. This solution also preserves characteristics of the interior ballistics (the operating pressure and the specific pulse). Qualitatively, the improvement in the oxygen balance of the reaction of burning out the rocket combustion products is well illustrated with comparison between:
a) the burn-out reaction that utilizes the air oxygen, with no oxidizer

$$2\mathrm{CO} + 3\mathrm{H_2} + 2\mathrm{OH} + 2\Big(\mathrm{O_2} + \frac{0.79}{0.21}\mathrm{N_2}\Big) \Rightarrow 2\mathrm{CO_2} + 4\mathrm{H_2O} + 2\frac{0.79}{0.21}\mathrm{N_2} \tag{2.1}$$

b) the burn-out reaction that utilizes an additional oxidizer in the form of the K_2SO_4 charge

$$\begin{aligned}&\mathrm{K_2SO_4} + 2\mathrm{CO} + 3\mathrm{H_2} + 2\mathrm{OH} + 2\Big(\mathrm{O_2} + \frac{0.79}{0.21}\mathrm{N}\Big)\\ &\quad\Rightarrow 2\mathrm{CO_2} + 3\mathrm{H_2O} + \mathrm{H_2S} + \mathrm{K_2O} + 2\mathrm{O_2} + 2\frac{0.79}{0.21}\mathrm{N_2}\end{aligned} \tag{2.2}$$

The above written stoichiometric equations do not reflect the complexity of an actual combustion process with the combustion products burn-out. However, they allow one to perform qualitative analyses.

With the fact taken into account that rockets use a (nitro) polymeric fuel as the propellant, specific characteristics of that fuel encouraged us to carry out the analysis of applicability of potassium salts as oxidizers since they are inhibitors of natural initiation of the ignition of hot hydrocarbons resulting from combustion products coming into contact with the atmospheric air. Reactions of decomposition of potassium salts are endothermic, which allows for using them as flash reducing agents to dim the burn-out flame. In practice, these salts may be introduced into the solid propellant as an admixture or a separate charge placed externally inside the combustion chamber of a rocket. The literature references report that in the case the oxidizer is an admixture, the burn-out flame is fully suppressed when a 1.14% mass fraction (w/w) of potassium nitrate or 4% of potassium sulphate is added to the propellant. Such admixtures are sufficient to hold back the flames during the entire time of the rocket operation. In the second case, when an external charge is placed inside the combustion chamber, such flame suppression lasts only 20% of the entire time of the rocket engine operation. The much shorter time of complete flame suppression achieved for the second case results from the fact that the oxidizer is placed in the zone of the fuel combustion (flame) front, where temperature of gases reaches as much as 2426.8°C, i.e. it substantially exceeds the temperature of the K_2SO_4 decomposition (1688.8°C, according to Yang *et al.* (2000)). Under such circumstances, the decomposition process may be very violent, the reaction time will depend on the radial thickness of the potassium sulphate charge. Since the ratio of the fuel charge thickness to that of the K_2SO_4 charge is 2.75, the rate of the K_2SO_4 combustion (flame) front will be higher than the rate of fuel combustion (Bagrowski *et al.*, 2011).

The effect of applying the flash reducing agent to dim the burn-out flame from the rocket engine for the very same times from the engine start-up – in our case, the potassium nitrate (KNO_3) – is shown in Fig. 1.

3. Experimental examination

The subject of examination was a charge of the potassium sulphate (VI) salt. The process for manufacturing such charges has been developed as well (Hawley, 2003). The process assumes that a potassium sulphate charge is bonded together from a number of items compacted in the

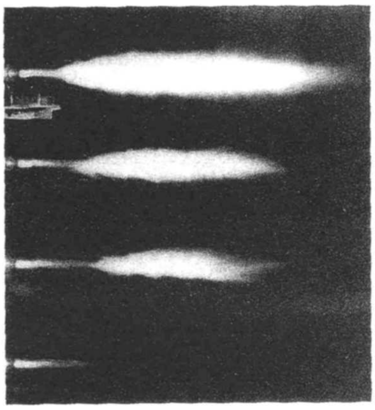

Fig. 1. Potassium nitrate affecting the exhaust gas burn-out – comparison of thermograms recorded at the initial stage of the fuel-charge burning for various mass fractions (w/w) of KNO_3 introduced in the combustion chamber (mass fractions given in %)

form of bushes placed on a stainless steel rod. The compacted items are made of potassium sulphate granulated into grains coated with some adhesive substance (binder). The granular structure was developed on the basis of experiments and tests intended to minimize the content of the binder and to acquire suitable physical properties of the compacted items. An average total weight of the K_2SO_4 salt charge was 120.1 g, whilst the weight of pure potassium salt was 105.1 g. An average length of the salt charge was 489.4 mm. Three rocket engines were subject to examination., one of them was provided with the oxidizer charge. Location of the K_2SO_4 charge inside the rocket engine is shown in Fig. 2.

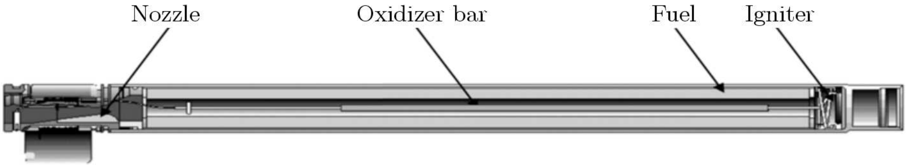

Fig. 2. Cross-section of a rocket with a charge of K_2SO_4

Measurements of changes in both temperature of the exhaust jet and thrust of the rocket engine were carried out on a vertical test stand. The rocket engine was fixed vertically in two yoke clamps, with its exhaust nozzle directed upwards. The engine thrust was measured using a measuring path with the piezoelectric sensor from the PCB Electronic as its key component mounted in a socket of the support plate of the test stand. Measurements were recorded with an oscilloscope. The recording was triggered by a voltage signal from the piezoelectric sensor connected to one of the oscilloscope input channels. The masurements were stored as a text file and then processed with the spreadsheet software [15]. The recorded changes in the engine thrust with time were used to calculate the value of the total impulse of the rocket, according to the formula

$$I_c = \int_{t_p}^{t_k} P \, dt \tag{3.1}$$

where: I_c is the total impulse of the rocket, P – thrust, t_k – time of termination of the engine operation, t_p – time of starting the engine.

Table 1 gives calculated values of total impulses of the rockets under examination.

Computations have been made for each experiment separately applying time intervals that cover totals of measurements recorded (see Fig. 3).

Table 1. Measured values of total impulses of the examined rockets

No.	Engine lot	I_c [Ns]
1	Engine from the 2nd lot as of 2006	6762
2	Engine from the 1st lot as of 2009	6716
3	Engine from the 2nd lot as of 2006, with K_2SO_4 oxidizer	6815

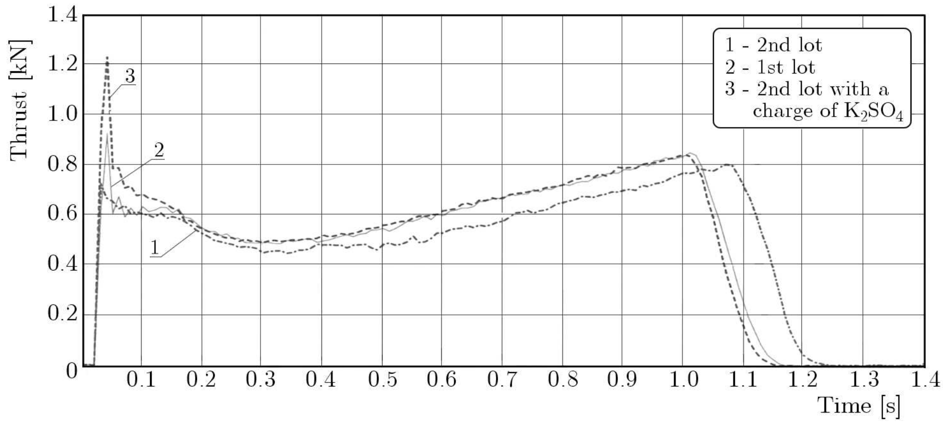

Fig. 3. Thrust measurements

The foregoing results are consistent with the value specified in the Specifications for the total impulse of the rocket ($\geqslant 6.7$ kN); furthermore, they indicate that the oxidizer bar only slightly affects, i.e. changes the engine thrust (Fig. 3), which has no practical effect on ballistic properties of the rocket. The measurement has displayed a much higher engine thrust peak at the initial stage of the engine run (Fig. 4). It may indicate that the rocket engine starts its operation under erosion-attributable unsteady conditions, which is typical of engines, within which the combustion process takes place on the lateral surface of the fuel (Kurow and Dołżański, 1964). It may result from the fact that the oxidizer has been introduced in the already existing engine chamber. The higher the fill factor of the combustion chamber (with the combustion process taking place on the lateral surface of the fuel), the stronger tendency of the rocket engine to change over to the unsteady mode of operation (see, e.g. Safta *et al.*, 2011). Results of some additional tests carried out both on the test bench and under field conditions (not described in this paper) prove that the flow area increased by necking the rocket charge at the nozzle has suppressed the above-described disadvantageous phenomenon, with no effect upon the thrust value.

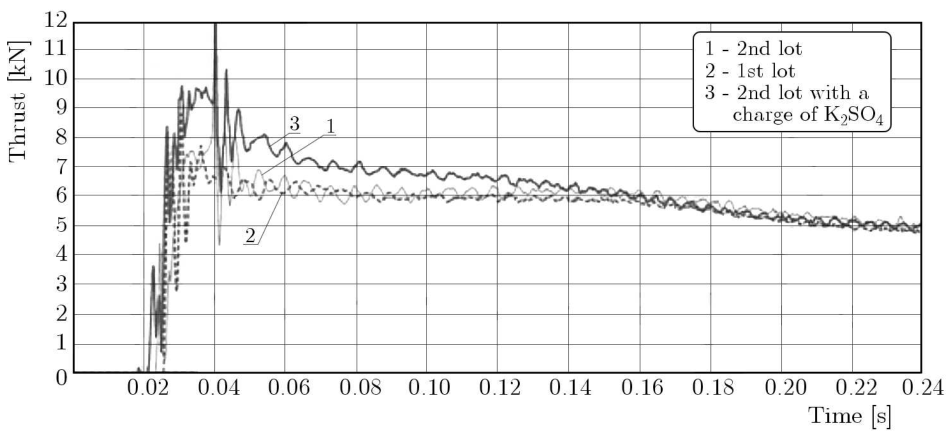

Fig. 4. Thrust measurements – the initial stage of rocket engine operation

Temperature of the exhaust jet was measured using two independent measuring paths. The first one comprised two nickel-chromium, type K, thermocouples from the Czaki Thermo-Product and an oscilloscope to record the measurements taken. The thermocouples were attached to a fixed pole and aligned with the central axis of the tested rocket engine at distances of 1.5 m and 3 m from the nozzle mouth. The thermocouples located in this way could measure temperatures inside the flame of the exhaust gas burning out.

The recording process was triggered by a voltage signal from the control panel, normally used to start rocket engines. Measurements were taken and recorded using two measuring channels of the eCroy WJ334 oscilloscope. They were stored as a text file and then processed with the spreadsheet software to gain a set of discrete values. An example of the set of temperature changes with time is shown in Fig. 5.

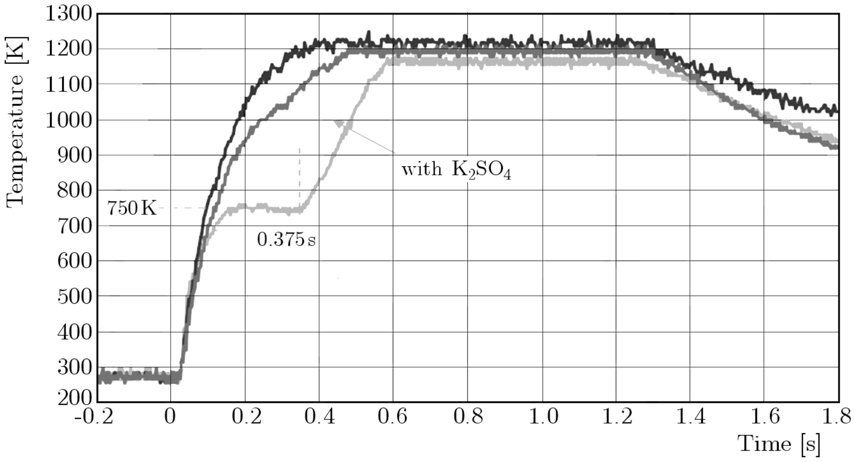

Fig. 5. Changes in temperature recorded by thermocouples at the distance of 3 m from the nozzle mouth

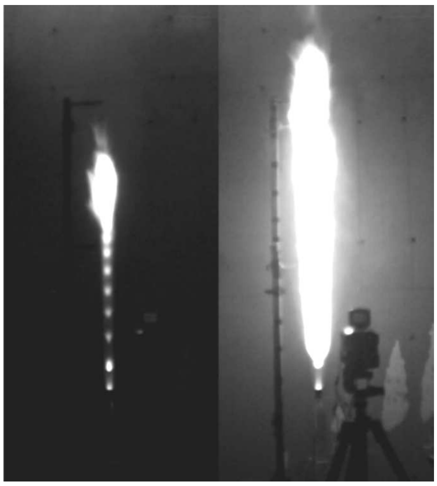

Fig. 6. Thermograms showing corresponding moments of the rocket engine operation: with (left) and without K_2SO_4 oxidizer charge applied (right)

Results of the examination of rocket engines with the K_2SO_4 charges inside were compared with those obtained from tests of engines with no oxidizer bars inserted (Bagrowski *et al.*, 2011). The temperature measurements support the assumed thesis that an oxidizer charge inserted in the combustion chamber of a rocket engine reduces the exhaust gas temperature. Changes in temperature recorded during the examination indicate that the burning time of the K_2SO_4 oxidizer charge is ca. 600 ms, whilst the flame temperature for the combustion with the oxidizer activated was reduced down to 750 K (477°C). Figure 6 shows the process of the exhaust gas burnout as recorded in the course of the testing work, 0.2 s after the rocket engine had been started.

The second independent path to take temperature measurements involved a set of two thermal imaging cameras. Both cameras were located at the distance of 4 m from the central axis of the rocket engine. The 3-5 μm spectral band thermovision images were recorded by means of the FLIR SC6000 HSDR thermal imaging system with the InSb (Indium Antimonide) IR detector with matrix resolution of 640 × 512 pixels. The ND3 IR filter and the optics of focal length of 25 mm made the system complete. The system was calibrated for the temperature range of 100-1200°C using the SR-800 Blackbody Control Master by the CI-Systems Inc. Sequences of images were recorded with the frequency of 126 Hz. The recorded data was then subjected to processing by means of the RTools software from the FLIR Systems Inc. (Lubieniecka and Łukasiewicz, 2011).

The following infrared images areas were selected for the thermographic analysis:

- the exhaust cone region – to estimate the exhaust gas temperature immediately after it leaves the nozzle,
- the region of the exhaust jet, i.e. of the mixture of the exhaust gas and air – it is the so-called burn-out region or the secondary-flame ('backfire') region,
- quasi-point region located at a distance of ca. 1 m from the nozzle.

The switch-over from recording temperature variations at particular points to taking measurements at 'quasi-point' surface regions was intended to get rid of severe temperature fluctuations occurring in the course of some violently proceeding phenomena, e.g.:

- turbulent flow,
- burning of gases.

Both the substitution of points with regions of small surfaces and the read-outs of average temperature made the interpretation of the results gained easier. Average temperature changes with time have been determined.

The changes in temperature for the three above-listed engines (Figs. 7, 8, 9) are presented below:

- region of the engine exhaust cone (Fig. 7) – exhaust gas temperatures are nearly identical for all the engines and range from 496.8°C to 526.8°C; this proves that any admixture/additional charge remains of practically no effect on the exhaust gas temperature,
- region of the secondary flame ('backfire') (Fig. 8) and that at the distance of 1 m from the nozzle (Fig. 9) – for engines #1 and #2 the gas burning temperatures are quite similar and the range from 796.8°C to 846.8°C, whereas behaviour of engine #3 remains different from those two, i.e. nearly immediately after the rocket ignition (in ca. 50 ms) there is a drop in temperature from the initial ca. 796.8°C to ca. 496.8°C; the temperature remains at this level for approx. 0.3 s; for another 0.1 s it rises up to the level of 796.8-846.8°C,
- with analyses of the recorded changes as the basis, one can determine thermal parameters typical for engine performance with reference to characteristic time intervals throughout the entire engine operation:
  - 0-0.1 s – the engine start-up phase – it proceeds in a very similar way for all three engines; characteristic of this phase is temperature rise up to maximum values of 896.8 K; for engines #1 and #2 the run-up phase terminates in the operating conditions getting stabilized at the level just reached; for engine #3, however, a rapid drop in temperature is recorded in some selected regions under control (496.8°C - 596.8°C) owing to the application of an additional oxidizer charge; the delay results from the inertia of physical and chemical processes of the oxidizer material decomposition;

- 0.5 to ca. 1.1 s – all three engines produce similar changes in temperature – at a distance of approx. 0.6 m from the nozzle the temperature rises from 496.8°C up to 796.8°C, and then to 896.8°C at the distance of approx. 1 m;
- after approx. 1.1 s – all three engines produce similar changes in temperature; the temperature gradually decreases.

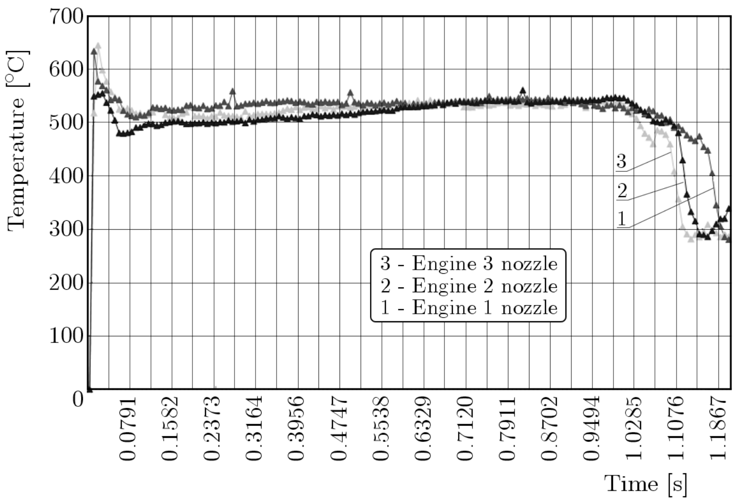

Fig. 7. Changes in exhaust gas temperature within the exhaust cone region

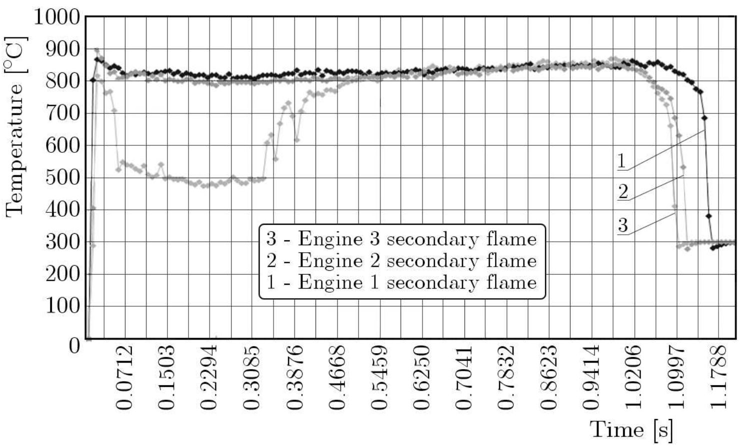

Fig. 8. Changes in exhaust gas temperature within the region of the secondary flame ('backfire')

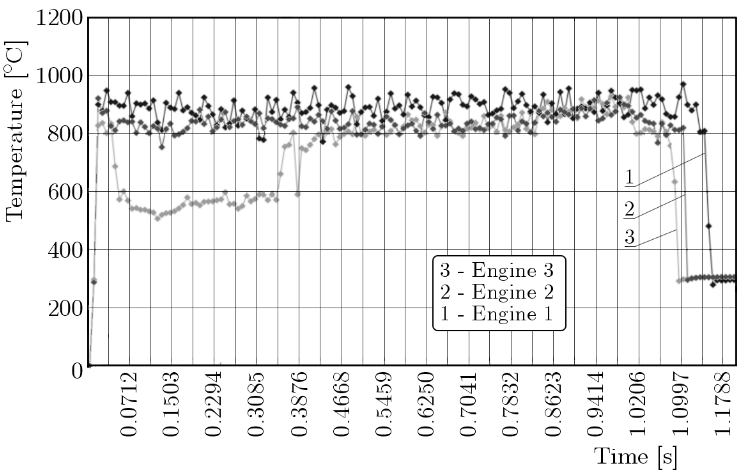

Fig. 9. Changes in exhaust gas temperature at the distance of 1 m from the nozzle

Another phase of the work consisted in the examination of spectral characteristics of rocket engines recorded with the SR-5000 spectral radiometer (Lubieniecka and Łukasiewicz, 2011). Four rocket engines were subject to examination, two of them, No. 5 and No. 6, were provided with oxidizer charges.

Determination of spectral characteristics proved to be extremely difficult since radiation of the exhaust gas is a sophisticated phenomenon. The spectrum of any particular exhaust-gas component is very complex. It reflects the complex and complicated nature of interactions of ions of a gas molecule; the radiation itself can be related to particular vibrational modes of ion oscillations – lateral and transverse vibrations. When spectra of particular components overlap, the resulting spectral resolution proves to be relatively complicated. It should be emphasized that the parameters of the spectral resolution are also affected by temperature. Another difficulty arises from the fact that certain substances, e.g. CO and CO_2 radiate in a very similar way, and in practice, their spectral bands overlap. The next problem is the presence of water vapour that generates hundreds of radiation bands over the entire infrared range. The infrared spectrum analysis enabled us to identify major components of the exhaust gas mixtures from the engines under examination. Molecules that have greatest shares in the IR radiation are those of H2O, CO_2 and CO. Figure 10 shows radiation wavelengths for components that substantially contribute to the IR radiation of rocket engines. For two engines with the oxidizer inside, one can easily see changes in the recorded radiation, which resulted from the reduction in temperature of the burning flame, and hence, the reduced amount of emitted radiation. The bandwidth of 4.3 to 4.9 μm is the area of particular interest since it covers radiation of both the CO and CO_2 within the above mentioned proportion of ca. 80% of CO_2 and 20% of CO. The substantial reduction in the radiant power for that bandwidth for engines #3 and #4 as compared to the adjacent bandwidth for CO (4.1-4.3 μm) suggests that with the potassium sulphate applied less amount of CO_2 is produced in favour of CO. It means that the exhaust gases from the engines, where the K_2SO_4 oxidizer was added, do not react with the ambient air.

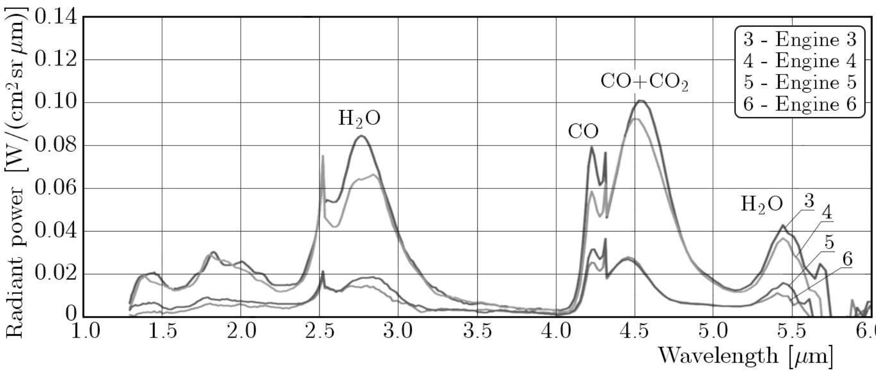

Fig. 10. A spectrogram of the third scan (after ca. 300 ms of combustion) for the wavelength range of 1-6 μm for engines #3, #4, #5 and #6

The above is also confirmed in Fig. 11 that shows variations of the radiation power for individual engines and functions of time. For the modified engines (with the oxidizer inside), the radiation power is much less during the first moments after the missile engines kick off. After having the mixture of salts combusted, the results are quite similar for all four engines. The effect of K_2SO_4 is observed during the initial 400 ms after ignition of the engines.

The modified rocket engines (#5 and #6) demonstrated about fourfold reduction in the infrared radiant power (hence, reduction in the exhaust gas temperature) over approx. 400 ms and a substantial reduction in the carbon dioxide emission. The probable reason for this reduction

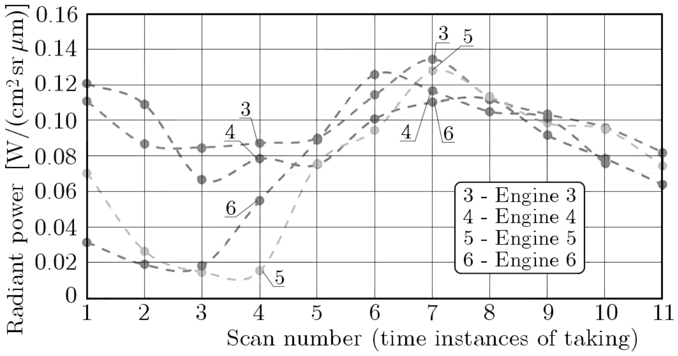

Fig. 11. Changes in the radiant power against time while operating particular engines (time is reflected by scan numbers, scans being spaced every 100 ms)

in CO_2 emission was that the temperature of the exhaust jet was lowered down to the level at which the reaction

$$CO + \frac{1}{2}O_2 = CO_2 \tag{3.2}$$

runs at a very low rate. That hypothesis is confirmed by the following observations:

- comparison between CO radiation intensities within the bandwidth of 4.1-4.3 μm for modified and non-modified engines (over first 400 ms of engine operation) gives the ratio of ca. 0.45;
- the ratio of intensities of carbon dioxide emissions (the bandwidth of 4.3-4.9 μm) by a modified and a non-modified engine amounts to ca. 0.25.

Therefore, attenuation of the CO_2 radiation bandwidth is much stronger than that of the CO radiation bandwidth, which confirms the thesis that the oxidation of the carbon monoxide is interrupted due to reduction in the exhaust gas temperature effected by the application of potassium sulphate.

4. Conclusion

The experience gained while using the rocket missiles of the same type but different batches (i.e. missiles manufactured on the basis of the same technical documentation) proves variations in the level of risk that the engines of the aircraft and the missile launch platform system would surge. This results from some differences in parameters of particular components used to manufacture the rocket propellant – even in the case when these parameters are kept within a certain acceptable limits. The nowadays used acceptance tests and equipment do not allow any hazard of this kind to be detected at the manufacture level. This is the main reason why the authors have decided to take a more global approach to the problem and solved it by applying the K_2SO_4 charge. The introduction of an additional component in form of an oxidizer bar to the combustion chamber of a solid-fuel rocket engine gives a substantial reduction in temperature of the exhaust gases emitted by the engine and mitigation of the rocket engine exhaust gas burnout, in particular of carbon dioxide. Measurements taken up to the present prove that this method of reducing the burn-out flame remains irrelevant to mechanical, dynamic and ballistic parameters of the rocket engine. It means that improvements in the aircraft safety at the moment the missiles are launched can be achieved by improvement in the oxygen balance. Hence, it seems reasonable to furnish the missile power plant with an oxidizer bar, as described above. Obviously, such a modification must be preceded with thorough in-flight examination/tests on the interactions between the weapon and the carrier. Moreover, lower temperature of the rocket engine exhaust gases results in the reduced wear-and-tear of structural components of the missile launcher and prolongation of its lifetime. Another advantage and argument for applying oxidizer bars is the fourfold reduction in radiation generated by the running rocket engine. This considerably reduces the capability of the enemy's passive protection systems to detect a missile taking off from the carrier.

The major reason for the authors' effort was to prevent the hazard of surge while launching missiles, which may result in setting fire to the aircraft or to the engine(s) shut-down, i.e. to crash.

The above-formulated objective has been reached. In the nearest future, the authors are going to proceed with the modelling of phenomena that take place in the combustion chamber in the decompression region; also, with the optimisation of the structure of the K_2SO_4 charge itself.

References

1. Bagrowski J., Kamiński R., 2011, *Development of the process to Manufacture the K_2SO_4 Salt Charges for Rocket Engines* in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa
2. Bagrowski J., Kamiński R., Klemba T., 2011, *On the Impact of the Nitrocellulose Nitrification Degree on the Oxygen Balance of the Rocket Propellant; Determination of Physical and Chemical Parameters of Combustion Products* (in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa
3. Gajewski S.A., Morozov F.N., Tikhomirov J.P., 1980, *Automatics for Aircraft Gas Turbine Power Systems* (in Russian), Voennoe Izdatelstvo Ministerstva Oborony SSSR, Moskva
4. Hawley E., 2003, *Mk66 Rocket Motor/Helicopter Compatibility Program*, Indian Head Division
5. Kowalski M., Pietraszek M., Szymczak J., Winczura Z., 2011, *Analysis of the Aircraft Engine Stall/Surge in the Course of Launching the NLPR-700 Rockets from the W-3W Helicopter*, (in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa
6. Kurow W., Dołżański J., 1964, *Principles Behind the Solid-Propellant Rocket Design* (in Polish), Wydawnictwo MON, Warszawa
7. Leciejewski Z., 2010, *Analysis and Assessment of Correctness of Methods of Pyrostatic Testing of Single- and Double-Based Powders* (in Polish), Wojskowa Akademia Techniczna (WAT), Warszawa
8. Lubieniecka J., Łukasiewicz J., 2011, Report No. S-1/16/2011 on examination/testing of 02.000"R1" power plants of uncontrolled air-launched rockets (NLPR-70) (in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa
9. Lubieniecka J., Łukasiewicz J., Tchórznicki K., 2013, Report No. S-1/16/2011 on examination/testing of power plants furnished with K_2SO_4 charges in uncontrolled air-launched rockets (NLPR-70), (in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa
10. Niechaev J.N., Fiedorov R., et al., 1980, *Theory of Aircraft Engines* (in Russian), Voennoe Izdatelstvo Ministerstva Oborony SSSR, Moskva
11. NO-10-A232:2009 Uzbrojenie lotnicze. Balistyka wewnętrzna silników rakietowych. Badania naziemne (Air weapons. Interior ballistics of rocket engines. Ground tests)
12. Operation Manuals for Helicopters (in Polish):
    – Mi-24A and Mi-24D helicopters, practical aerodynamics, Lo 2011/80, Poznań 1980
    – Turbojet aircraft engine TW3-117MT series 3, Technical description and maintenance guidelines, Lot 2855/90, Poznań 1991
    – Turbojet engine TWD-10B, Maintenance Manual, Doc. No. 19.0.646 (10B.00.030.RE2), WSK--Rzeszów, 1984

13. Romasiewicz W.F., Samoilow G., 1980, *Practical Helicopter Aerodynamics* (in Russian), Voennoe Izdatelstvo Ministerstva Oborony SSSR, Moskva
14. Safta D., Vasile T., Ion I., 2011, Regarding the influence of high frequency combustion instabilities on operation of solid rocket motors, *Problems of Mechatronics. Armament Aviation, Safety Engineering*, **1**, 3, 7-24
15. Stieczkin B.S., Kazandżan P.K., Aleksiejew Ł.P., Goworow A.N., Konowałow N.E., Nieczajew J.N., Fiodorow R.N., 1961, *Theory of Aircraft Engines* (in Polish), Wydawnictwo MON, Warszawa
16. Torecki S., 1989, *Interior Ballistics of Solid-Propellant Rocket Engines* (in Polish), Wojskowa Akademia Techniczna (WAT), Warszawa
17. Yang V., Brill T., Ren W., 2000, *Solid Propellant Chemistry, Combustion, and Motor Interior Ballistics*, American Institute of Aeronautics and Astronautics, Virginia
18. Żyluk A., 2009, *Simulation Tests of Exterior Ballistics Demonstrated by Air Weapons* (in Polish), Instytut Techniczny Wojsk Lotniczych (ITWL), Warszawa

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