# DLR-IB-AT-KP-2022-32

## Technological Level of CMC Components for Stationary Gas Turbines and Aero-Engines

### DLR Report

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# 1 Introduction

Ceramic matrix composites (CMCs) offer promising potential for efficiency improvement of stationary gas turbines and aero-engines [1]. Compared to conventional nickel-based superalloys, CMCs possess a higher temperature capability, hence CMCs are especially predestined for the use in the hot gas section for cooling air reduction and/or process temperature increase. Furthermore, the density of CMCs is about one third lower than that of metals, benefiting the development of lighter aero-engines.

In this report, the technological level of CMCs for stationary gas turbines and aero-engines is presented in order to highlight relevant CMC components and to assess the current development status. Initially, CMC basics are described in terms of material properties, manufacturing and coating (ch. 2). After a historical overview of the CMC development and major research programs (ch. 3), important milestones in CMC component fabrication are detailed, focusing on CMC combustor components (ch. 4), CMC turbine components (ch. 5) and CMC nozzle components (ch. 6).

## 2 Basics

CMCs consist of fibers embedded in a ceramic matrix. The fiber reinforcement is not intended to increase strength, but to improve fracture toughness [2, 3]. Regarding stress-strain behavior, ceramics without fiber reinforcement, so-called monolithic ceramics, are characterized by a steep linear rise with subsequent brittle fracture, whereas CMCs show a quasi-plastic curve (**Fig. 2.1**). Energy dissipating mechanisms, like crack deflection, crack splitting or fiber pull out, leading to an increase in crack resistance of the ceramic fiber-matrix composite, are the main reasons for the quasi-plasticity of CMCs. Compared to monolithic ceramics, CMCs offer a more damage-tolerant material behavior, and thus can be used under higher thermo-mechanical loads, such as in the hot section of gas turbines.

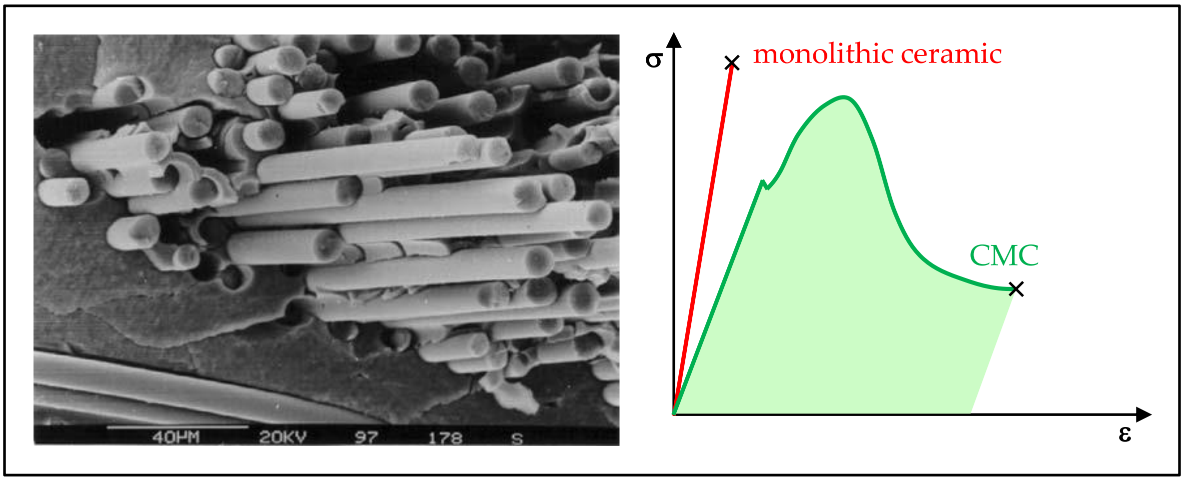


**Fig. 2.1:** Ceramic matrix composite (left) [4]
Schematic stress-strain curves of a monolithic ceramic and a CMC based on [2] (right)

The notation *fiber type/matrix type* is used to specify a CMC. Depending on the chemical composition, CMCs are classified into oxide and non-oxide CMCs. Oxide-fiber-reinforced oxide ceramic is also referred to as *Ox/Ox*. Various techniques can be used to manufacture Ox/Oxs, usually involving a sintering process [5].

In the field of non-oxide CMCs, carbon-fiber-reinforced carbon (C/C), carbon-fiber-reinforced silicon carbide (C/SiC) and silicon carbide-fiber-reinforced silicon carbide (SiC/SiC) are important materials. Particularly SiC/SiC is predestined for hot gas applications in stationary gas turbines and aero-engines due to its high thermo-structural capability [6-9]. Important infiltration processes for creating a SiC matrix around a fiber architecture are as follows: Chemical vapor infiltration (CVI), polymer infiltration and pyrolysis (PIP) and reactive melt infiltration (RMI) [10-12]. CVI is based on chemical vapor deposition, whereas PIP utilizes polymers being infiltrated and finally pyrolyzed. RMI is characterized by infiltrating a prepared fiber preform with molten silicon, leading to reaction with subsequent SiC matrix generation. In terms of RMI, different techniques exist, such as liquid silicon infiltration (LSI), NASA's slurry-cast MI process or the prepreg-MI process developed by General Electric (GE). LSI uses a porous carbon matrix as a preform reacting with the molten silicon to the SiC matrix. In the slurry-cast MI and prepreg-MI process, the fiber preforms are impregnated with special slurries before silicon infiltration (**Fig. 2.2**). However, the two methods differ in the preform fabrication procedure. In the slurry-cast MI process, fiber lay-ups are soaked with a slurry, while in the prepreg MI process, single fibers are treated and formed into lay-ups afterwards.

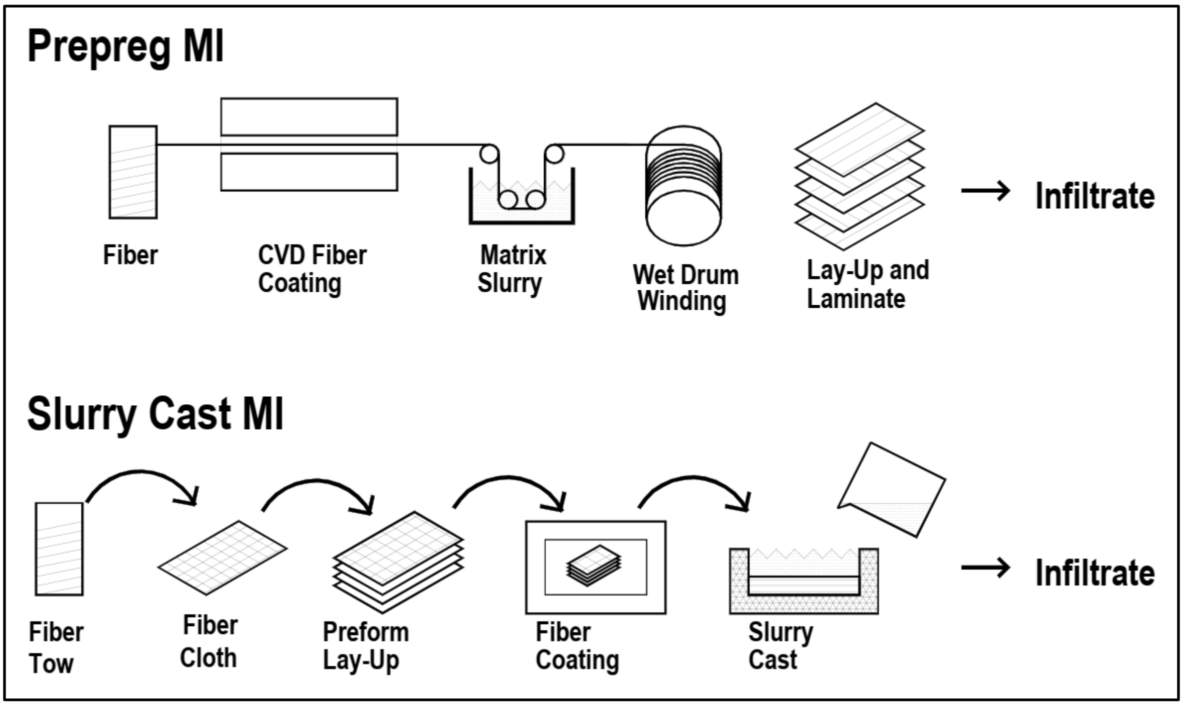


**Fig. 2.2:** Illustration of the preform fabrication in the slurry-cast MI and prepreg-MI process [13]

Current SiC/SiCs produced via slurry-cast MI or prepreg-MI process, like GE's HiPerComp®, provide temperature capability up to 1316 °C [10, 14-17]. This temperature limitation is essentially due to a residual amount of silicon in the matrix which did not react to silicon carbide during infiltration. Novel manufacturing approaches developed by NASA Glenn Research Center (GRC) focus on the creation of a silicon-free and dense SiC matrix by a hybrid CVI/PIP process [16-18]. In the future, the temperature capability of SiC/SiCs is expected to be increased to 1482 °C by using creep-resistant SiC fibers, an advanced 3D fiber architecture and a hybrid CVI/PIP matrix.

Environmental barrier coatings (EBCs) are required to protect SiC matrices from environmental influences, particularly from the damaging reaction with water vapor [19-22]. The use of EBCs significantly increases the service life of SiC-based CMCs. If a CMC component with EBC is cooled, the EBC can additionally cause thermal insulation. In reference to thermal barrier coatings (TBCs) of metallic hot gas components, such EBCs are also called thermal and environmental barrier coatings (TEBCs). NASA GRC is developing and testing EBCs for various applications and temperature requirements [14, 19, 23]. Current SiC/SiCs for gas turbines, produced via slurry-cast MI or prepreg-MI process, could be combined with EBCs of NASA GRC to a system providing a CMC temperature capability of 1316 °C and an EBC surface temperature capability up to 1482 °C.

## 3 Historical Overview

Research has been carried out in the field of CMCs in the USA and Europe since the 1960s. Originally, the development of C/C thermal protection systems was focused on spacecraft, however the potential applications of C/C increased over time [3, 24, 25]. One of the most popular C/C components are high performance brake discs, being initially used in aircraft construction and later in motor sports. Due to the low oxidation stability of C/C at high temperatures, there was a need for CMCs that could also be used in an oxidative atmosphere. Therefore, research was conducted on new matrix materials and the development of CMCs with SiC matrix began in the 1970s.

In 1977, one of the first SiC-based CMC was manufactured by using CVI process under the direction of Prof. Roger Naslain at the University of Bordeaux in France [26]. Two years later, the first industrial furnace for the production of CMCs with SiC matrix was constructed at Société Européenne de Propulsion (SEP, later Snecma Propulsion Solide, now Safran Ceramics).

In the 1980s, research into CMCs and their manufacturing processes was intensified. Furthermore, the first companies, such as GE or Snecma, invested in this technology. GE filed its first CMC patent in 1986 and has been developing CMCs ever since [27, 28].

Snecma also began to evaluate CMC prototypes in cooperation with SEP, focusing on subcomponents of military nozzles [8, 26, 29]. In 1989, the world's first demonstration of CMC nozzle flaps installed in the M53-2 engine of the Mirage 2000 took place at the Paris Air Show.

In the 1990s, CMC outer and inner nozzle flaps were developed and tested as part of the Rafale/M88-2 test program by Snecma and SEP, which later merged into Snecma Propulsion Solide (SPS, now Safran Ceramics) [3, 8, 29]. An important milestone was reached in 1996 when the first CMC outer nozzle flaps for the M88-2 engine of the Dassault Rafale entered series production. During testing of the CMC inner nozzle flaps, it was observed that the SiC/SiC used could not withstand the high thermomechanical requirements of the M88-2 engine. Hence, a new SiC matrix generation based on a self-sealing matrix concept was developed [24, 30].

In the meantime, a wide variety of CMC materials and generations have been developed, which were marketed by Safran under the names Sepcarb® (C/C), Sepcarbinox® (C/SiC) and Cerasep® (SiC/SiC) [24, 31, 32].

Beginning in the 1990s, the US government supported CMC research by initiating several campaigns. The NASA *Enabling Propulsion Materials* (EPM) program (1994-1999) was part of the NASA High Speed Research (HSR) program and was intended to develop high-temperature materials for the High Speed Civil Transport (HSCT) engine, which was researched in the HSR program [33]. NASA, GE and Pratt & Whitney (P&W) cooperated in the EPM program and developed the slurry-cast MI process for the manufacturing of SiC/SiCs [34, 35]. Moreover, EBC systems have been significantly optimized by a three-layer design [36, 37].

In 1992, the *Ceramic Stationary Gas Turbine* (CSGT) program started in the USA being conducted by Solar Turbines on behalf of the US Department of Energy (DoE) [38]. The aim of the program was to

improve the performance of stationary gas turbines by using ceramic components in the hot gas section. The Solar Centaur 50S engine of Solar Turbines served as the basis for research and testing. Numerous CMC liners were developed and tested in the course of the CSGT program [39]. The research activities of Solar Turbines were later continued in the *Advanced Materials for Mercury 50 Gas Turbine Combustion System* program of the DoE and in an Advanced Technology Program (ATP) of the National Institute of Standards and Technology (NIST) [39-41]. The NIST project involved a team of Solar Turbines, Siemens and ATK-COI Ceramics (ATK-COIC) which developed the concept of hybrid oxide CMCs. In this approach, an Ox/Ox is coated with a protective and insulating layer called friable graded insulation (FGI) [41, 42].
In parallel to the CSGT program, the DoE initiated the *Continuous Fiber Ceramic Composite* (CFCC) program (1992-2005) in cooperation with GE which later transitioned to the *Advanced Materials for Advanced Industrial Gas Turbines* (AMAIGT) program (2000-2010) [10, 13, 43]. As part of these campaigns, GE developed the prepreg-MI process to manufacture a SiC/SiC material that was named HiPerComp®. GE also fabricated and tested prepreg-MI shrouds and liners for the GE 7FA engine. The technology readiness level (TRL) of these CMC components was successively improved during GE's projects.
The research activities of Solar Turbines and GE also demonstrated that the EBC system, which was developed under the NASA EPM program, significantly increased the service life of a SiC matrix [36, 43, 44]. Thus, both companies supported the further development of EBCs.

In the US military, CMCs were primarily researched in the *Integrated High Performance Turbine Engine Technology* (IHPTET) program (1987-2005), which was continued in the course of the *Versatile Affordable Advanced Turbine Engines* (VAATE) program [8, 45-47]. The aim of the IHPTET program was to develop and demonstrate advanced technologies for military engines.
Since the 1990s, the US military has been developing and testing CMC divergent flaps and seals for nozzles of military engines [8, 48, 49]. After SPS had developed a new SiC matrix generation based on the self-sealing matrix concept, the US military evaluated this material in cooperation with Snecma in the 2000s [8, 26, 50, 51]. GE also contributed to military CMC nozzle development and introduced its Ox/Ox material [5, 8, 52].

In the 1990s and 2000s, the CMC development in Japan was promoted in the *Advanced Material Gas-Generator* (AMG) program (1993-2002) and in the *Research and Development of Environmentally Compatible Propulsion System for Next-Generation Supersonic Transport* (ESPR) project (1999-2004) [53-55]. The ESPR project was initiated by the Japanese government as a successor to the *Super/Hyper-sonic Transport Propulsion System* (HYPR) project and was aimed at researching environmentally friendly engine technologies in terms of noise, NOx and $CO_2$ reduction. Under AMG program, the potential of advanced materials in gas turbines was investigated in order to increase efficiency, save weight and reduce pollutants. As part of the two Japanese programs, prototypes of CMC hot gas components were developed and tested.

In the 2000s, the CMC technology was advanced by further programs and numerous research contributions. The NASA continued its CMC development from the EPM program in the *Ultra-Efficient Engine Technology* (UEET) program [56]. The aim of UEET was to reduce fuel consumption and NOx emissions of engines by improving propulsion and combustor concepts. In addition to the research of CMC liners, the development and testing of CMC vanes was also focused. Besides, the NASA slurry-cast MI process and the EBC technology of the EPM program were further developed. By using new SiC fibers and fiber architectures and optimizing the matrix infiltration process, advanced SiC/SiCs for high temperature applications could be manufactured [35, 57]. EBCs were enhanced by rare earth silicates and TEBCs were researched, being a combination of EBC and TBC [19-21, 58, 59].

Since 2010, US CMC research has been promoted in the NASA *Environmentally Responsible Aviation* (ERA) project [14, 60]. The NASA ERA project was initiated to develop and demonstrate future technologies for fuel burn decrease, NOx reduction and noise suppression. A TRL of 4-6 is to be achieved by 2020. The main CMC objectives are the manufacturability of complex CMC parts and the evaluation of the CMC component performance under engine operating conditions. Hence, CMC liners, turbine vanes and exhaust mixers were fabricated and tested by NASA (GRC).
In terms of CMC and EBC development, NASA GRC focused in the 2010s on improving and testing the CMC and EBC systems developed so far. The research activities were mainly supported by the NASA ERA project, the NASA *Fundamental Aeronautics Program* (FAP) and the NASA *Transformational Tools and Technologies* (TTT) project [21, 61-63]. Creep-resistant SiC fibers and advanced 3D fiber architectures were researched in order to improve SiC/SiCs. [7, 16-18, 63]. Moreover, a hybrid CVI/PIP fabrication process was considered to create a silicon-free and dense SiC matrix. In the field of EBCs, the development of advanced EBCs was focused [21, 64]. In addition to rare earth silicates, hafnium oxides were increasingly used for EBC optimization. Furthermore, an improvement of the bond coat technology was intended.

After the successful development of CMC shrouds and liners under the CFCC and AMAIGT program, GE focused on the development of CMC vanes for stationary gas turbines in the mid-2010s [65-68]. Important work is still being done in a DoE project.
Due to the success in the field of stationary gas turbines, GE also aimed at the use of CMC in aero-engines in the mid-2000s, so that mainly in the 2010s CMC components were evaluated in engine tests [8, 27, 69-72]. GE tested stationary CMC parts, such as shrouds and vanes, in the hot gas section of the F136 engine and uncooled CMC blades in the low pressure turbine (LPT) of the F414 engine.

Having gained a lot of experience with CMC in military engines, Safran also focused on CMC use in commercial engines in the mid-2000s and started testing different CMC components in the CFM56 engine [8, 26, 73]. Safran achieved success with a CMC exhaust system that finally received EASA certification in 2015 for a two-year evaluation phase.

Further relevant research contributions to CMC development in the field of civil nozzles were performed in the *Continuous Lower Energy, Emissions, and Noise* (CLEEN) program of the Federal Aviation Administration (FAA) (CLEEN), which started in 2010 [74]. In the CLEEN program, the FAA partners with the aviation industry in order to develop technologies for noise, emission and fuel burn reduction and to promote sustainable aviation.

In the 2010s, research was also carried out on the use of Ox/Ox in the hot gas section. Important contributions were achieved in the German *High Performance Oxide Ceramic* (HiPOC) program, the DLR project *Low NOx Oxide Ceramic Combustors for Aero-Engines* (LOCCA) [75-77] and by Siemens in a DoE project (2014-2018) [78]. In the HiPOC program and LOCCA project, DLR and Rolls-Royce Deutschland, among others, were involved in order to develop and evaluate Ox/Ox materials and liners for hot gas applications. Siemens continued to focus on the concept of hybrid oxide CMCs and its testing.

In Asia, CMC research was further intensified in the 2010s. On behalf of Ishikawajima-Harima Heavy Industries Corporation (IHI), CMC turbine components were developed in Japan, especially vanes and blades [79]. Furthermore, CMC turbine vanes were tested in China [80].

The first series applications of CMC components in commercial engines were realized by GE and CFM International in the 2010s. In the LEAP engine of CFM, a 50/50 joint venture of GE and Safran, CMC shrouds are used in the first stage of the high pressure turbine (HPT) [15, 27]. Moreover, a CMC exhaust mixer is installed in GE's Passport 20 engine[5, 81].
The first engine with multiple CMC components is GE's GE9X, which will power the Boeing 777X [15, 27]. In total, the engine will feature an outer and inner CMC liner, CMC vanes in the first and second HPT stage and CMC shrouds in the first HPT stage. The GE9X received FAA certification in September 2020 and the Boeing 777X debuted at the Dubai Airshow in November 2021 [82, 83].
Future CMC engine applications are planned in the next-generation LEAP engine and GE's *Adaptive Versatile Engine Technology* (ADVENT) military engine [27, 71, 84]. Rolls-Royce has also announced the use of CMCs in the HPT of their UltraFan engine [85].

## 4 CMC Combustor Components

The first CMC liners were developed and tested as part of a number of US programs in the 1990s. Under the NASA EPM program, various components of a rich-burn, quick-quench, lean-burn (RQL) combustor were manufactured using slurry-cast MI process (**Fig. 4.1**). The SiC/SiC combustor components were then tested in the RQL sector rig at NASA GRC in order to research the thermomechanical behavior of the CMC components under operating conditions [86].

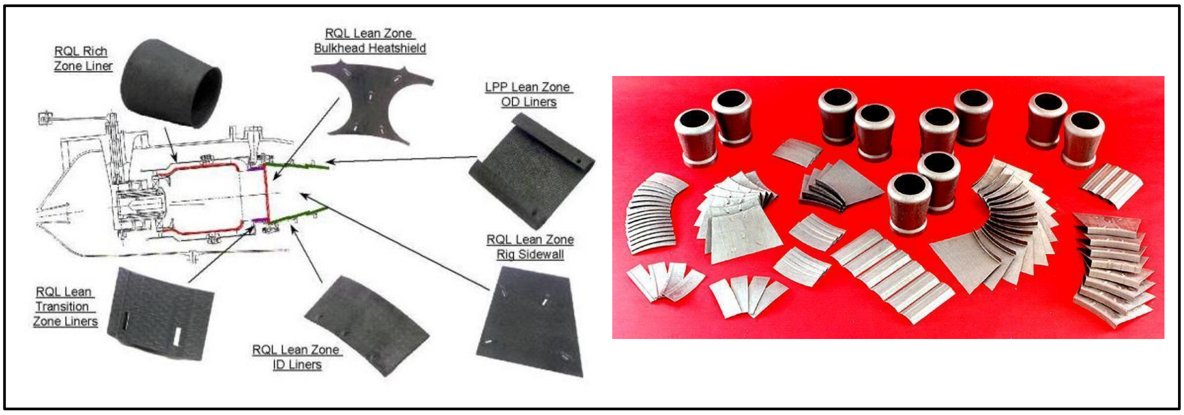


**Fig. 4.1:** Schematic of NASA GRC's RQL sector rig (left) [87]
Slurry-cast MI combustor components fabricated in the NASA EPM program (right) [87]

With the start of the CSGT program in 1992, various outer and inner SiC/SiC liners were evaluated by Solar Turbines in test rigs and Solar Centaur 50S engines (**Fig. 4.2**). The results obtained are summarized in [39]. In the course of the tests, a potential for NOx and CO reduction could be demonstrated. It was also shown that the service life of SiC/SiC liners could be significantly increased by using the EBC technology developed in the NASA EPM program [36, 44]. The EBCs enabled different test runs with a duration of 13 937 h and 15 144 h. Hybrid oxide liners were also researched in further investigations. The longest test run of a hybrid oxide liner lasted 25 404 h [88].

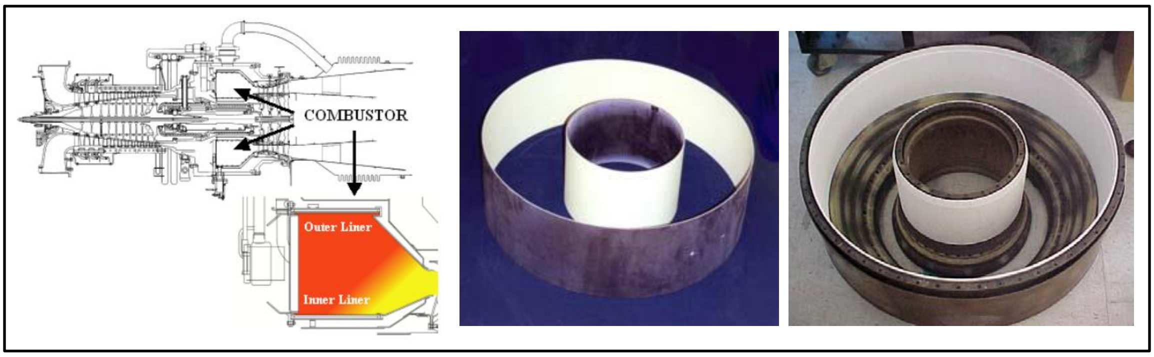


**Fig. 4.2:** Schematic of Solar Centaur 50S combustor with CMC liners (left) [88]
SiC/SiC liner set before 13 937 h test (center) [39]
Hybrid oxide liner set in combustor housing (right) [41]

In parallel to Solar Turbines, GE also researched SiC/SiC liners for stationary gas turbines as part of the CFCC and AMAIGT program. GE fabricated prepreg-MI liners for the GE 7FA engine (**Fig. 4.3**) and the Solar Centaur 50S engine [27, 43]. In a test run by Solar Turbines a GE liner with EBC achieved a test duration of 12 822 h.

**Fig. 4.3:** Uncoated prepreg-MI liner for the GE 7FA engine [43]

In addition to the previously described research activities by Solar Turbines and GE, the development of CMC liners was supported by many other contributions in the 2000s. Important work was performed by United Technologies Research Center (UTRC) by testing CMC liner prototypes (**Fig. 4.4**). First, a slurry-cast MI liner with EBC was fabricated for the FT8 engine and tested in a FT8 sector rig [89]. The SiC/SiC liner was designed for an unchanged flow rate in order to enable comparisons with the metallic baseline liner. Instead of injecting the cooling air through many small film cooling holes, the cooling air was mixed with the hot gas through enlarged dilution holes in the CMC liner walls. In the test runs it was demonstrated that the CO production near the liner wall could be significantly reduced, since elimination of film cooling layers leaded to complete combustion.

Later a three-piece SiC/SiC liner prototype with EBC for a full annular reverse flow combustor was manufactured via slurry-cast MI [90]. No film cooling was used and the liner was only cooled by backside impingement air. The CMC liner was tested in the PW206 combustor rig. In comparison to the metallic baseline liner, the emission of CO and NOx could be reduced and the pattern factor improved. The main reason for this was better mixing within the SiC/SiC liner which was caused by the absence of film cooling and the increase in dilution air.

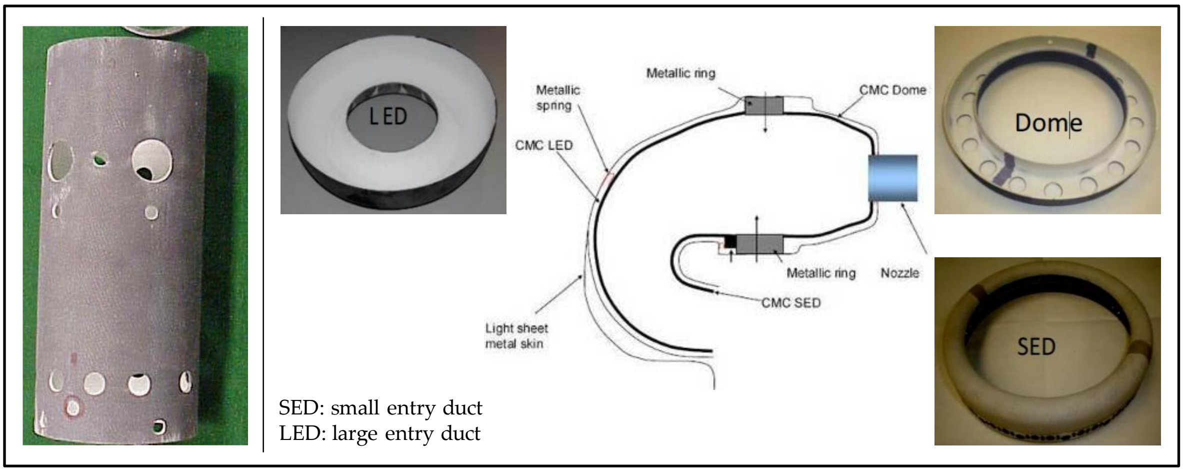


**Fig. 4.4:** Slurry-cast MI liner for the FT8 engine with enlarged dilution holes (left) [89]
Three-piece slurry-cast MI liner with EBC for a full annular reverse flow combustor (right) [90]

In Japan, the first CMC liner prototypes were developed as part of the AMG program and ESPR project [91-93]. Under the AMG program, a SiC/SiC liner for the AMG combustor was manufactured via PIP and then tested for three hours at a combustor outlet gas temperature of 1600 °C. A liner with a hybrid matrix of CMC and glass matrix composite (GMC) was also published later. The CMC part was infiltrated using PIP and the GMC part using MI. The combustion test of the hybrid CMC/GMC liner was performed for 15 hours at a gas outlet temperature of 1600 °C. In addition, a SiC/SiC liner was fabricated via PIP in the ESPR project, which was tested in short time cycle combustion tests.

In the 2000s, GE and Safran developed and tested full-scale CMC liners for the CFM56 engine in order to increase TRL and to focus the use of CMC liners in civil engines [6, 8, 27, 32]. The SiC/SiC liner of Safran was tested for 180 hours including 100 h of maximum conditions and achieved in an engine demonstration a cooling air reduction of 35% and a significant decrease in NOx (**Fig. 4.5**). As part of the UEET program, GE also fabricated a SiC/SiC liner via slurry-Cast MI which was successfully tested in rig and engine tests.

**Fig. 4.5:** Cerasep® A415 liner for CFM56 combustion chamber (left) [8]
Cerasep® A415 liners tested on a CFM56 combustion chamber (right) [8]

In the 2010s, the CMC liner development was advanced by further research activities. Under the NASA ERA project, SiC/SiC liner prototypes were manufactured by NASA GRC using GE's HiPerComp® [14, 94, 95]. Altogether, two uncoated outer liners and two inner liners including EBC and film cooling holes were fabricated (**Fig. 4.6**). An outer and inner SiC/SiC liner were then tested in the NASA High Pressure Burner Rig (HPBR), with focus on the EBC, the investigation of the CMC/EBC system and durability tests. The pair of CMC liners accumulated 250 h in the HPBR and was able to successfully demonstrate its rig durability. The combustion chamber pressure was approximately 16 bar. Based on chemical equilibrium analysis codes an average gas temperature of 1649 °C was calculated.

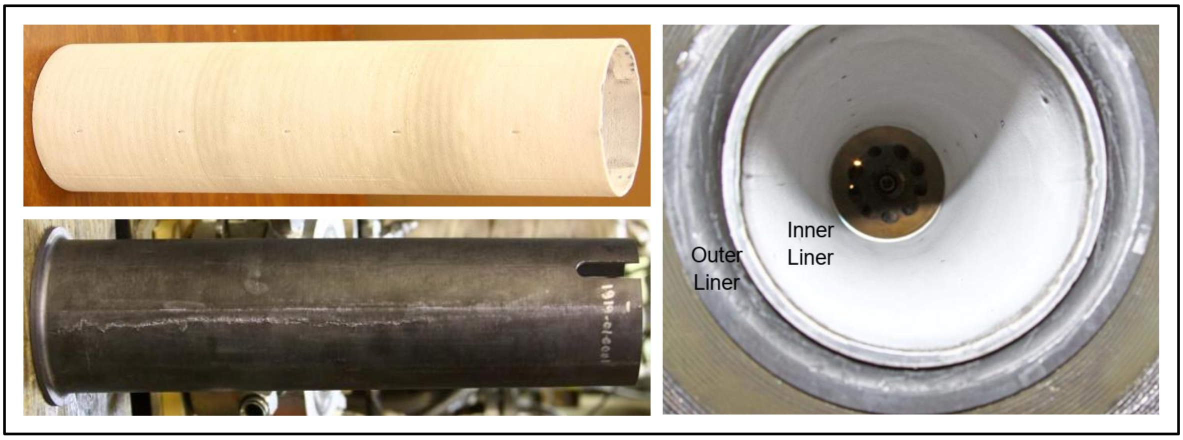


**Fig. 4.6:** Inner prepreg-MI liner with EBC and film cooling holes (top left) [14]
Outer uncoated prepreg-MI Liner (bottom left) [14]
Pair of SiC/SiC liners installed in the NASA HPBR (right) [14]

Another contribution to the CMC liner development was achieved by GE in the course of the NASA ERA project [96]. GE engineered a 5-cup CMC combustor sector rig for the NASA Advanced Subsonic Combustion Rig (ASCR) (**Fig. 4.7**). CMC liner materials were used to reduce cooling air and to enable a high mixer air flow split.

**Fig. 4.7:** GE's 5-cup CMC combustor sector rig for the NASA ASCR [96]

In the field of Ox/Ox liner development, further research activities can be observed in the 2010s. In Germany, the development and testing of Ox/Ox liner prototypes was initiated under the HiPOC program and continued in the LOCCA project [76, 77]. Several Ox/Ox liners with laser drilled effusion cooling holes were fabricated (**Fig. 4.8**). The liner prototypes were also coated with TEBC to provide thermal protection from the hot gas and to protect against water vapor corrosion. After manufacturing, the Ox/Ox liners were tested in a rig under realistic engine conditions. One liner was tested at conditions up to 80% take-off.

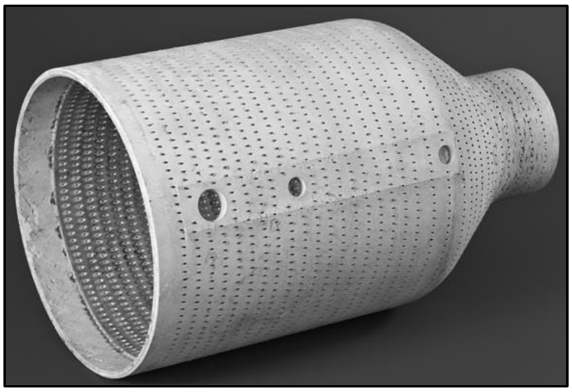

**Fig. 4.8:** Ox/Ox liner with effusion cooling holes and TEBC [76]

Furthermore, Siemens developed an advanced transition cone based on hybrid oxide CMC as part of a DoE project [78]. Important objectives were design and testing in order to increase TRL.

The first series application of CMC liners is intended by GE for its GE9X engine. According to [15], an inner and outer SiC/SiC liner are installed in the GE9X.

# 5 CMC Turbine Components

CMCs are especially predestined for turbine applications due to their high temperature capability. The technological level of CMC turbine components is presented below, divided into shrouds (ch. 5.1), vanes (ch. 5.2) and blades (ch. 5.3).

## 5.1 CMC Shrouds

Under the CFCC and AMAIGT program, the first CMC shrouds were developed by GE via prepreg-MI process [13, 27, 43]. The first HPT stage of GE's 7FA engine was used as design baseline for the shrouds. During these programs, the design and TRL of the CMC shroud system could be successively improved (**Fig. 5.1**). Testing of the SiC/SiC shrouds started with laboratory and rig tests and culminated in 7FA field engine tests with full EBC-coated SiC/SiC shroud sets. A CMC shroud set achieved a test duration of 21 740 h.

Due to the positive results in the course of the 7FA field engine tests, GE also focused the use of SiC/SiC shrouds in future engine developments in the mid-2000s [27]. In 2009, SiC/SiC shrouds were tested for the first time by GE in the hot gas section of the F136 engine [70, 71]. In the meantime, prepreg-MI shrouds with EBC are installed as standard in the first HPT stage of the CFM LEAP engine, which entered into service as LEAP-1A on an Airbus A320neo in 2016 (**Fig. 5.2**) [15, 27, 97]. GE's SiC/SiC shrouds are the first CMC turbine components being used in a civil engine. A further application of the shrouds is planned in the first HPT stage of the GE9X engine. The industrialized production of prepreg-MI shrouds mainly takes place in a CMC facility in Asheville (North Carolina) that opened in 2014 [98, 99]. In July 2021, GE's 100 000$^{th}$ CMC shroud was manufactured and shipped.

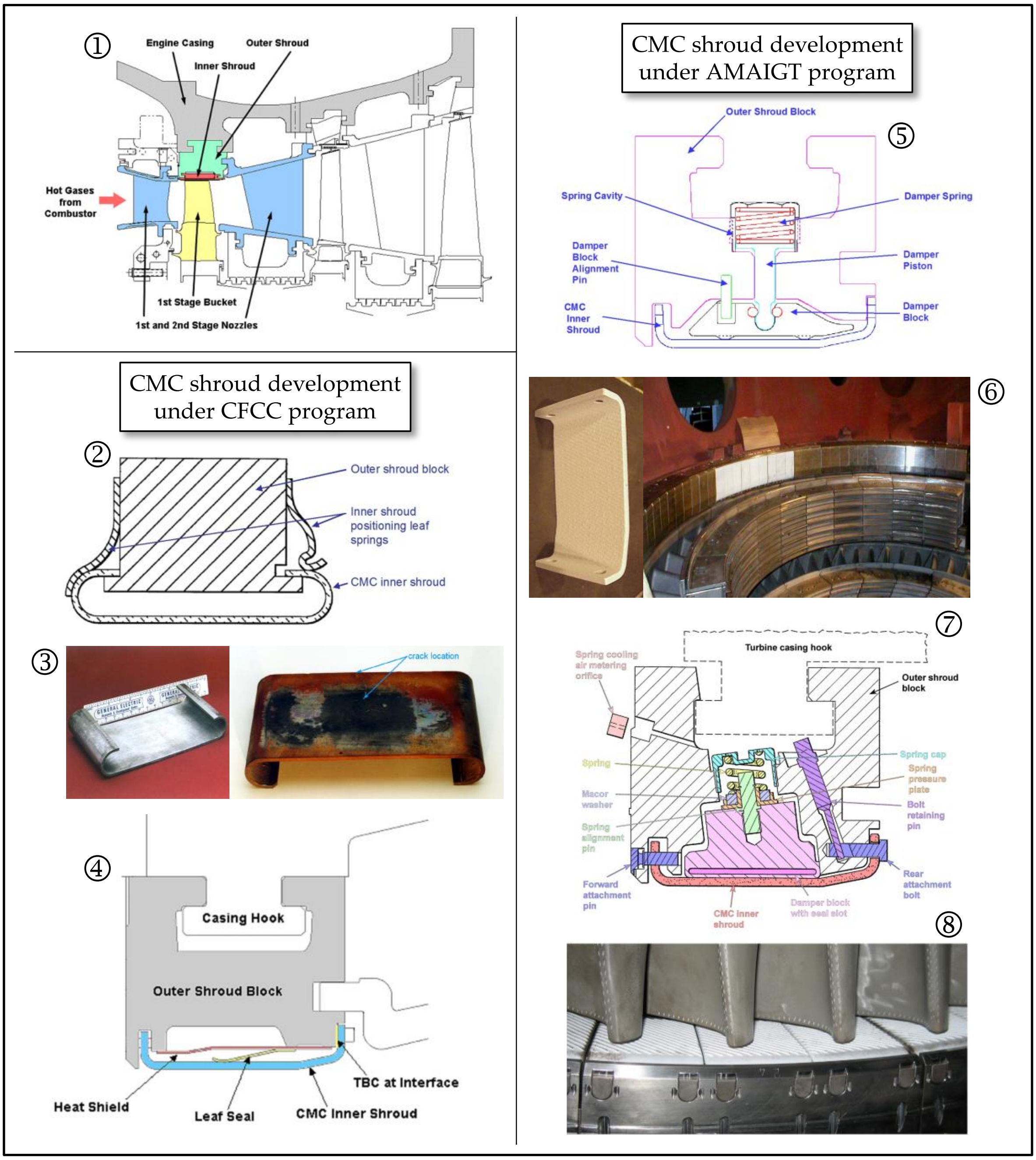


**Fig. 5.1:** Development of GE's prepreg-MI shrouds during CFCC and AMAIGT program

① Position of 1st stage inner shroud and outer shroud block in the GE 7FA engine [43]

② Tested SiC/SiC shroud system in the CFCC program [43]

③ SiC/SiC shroud before and after rig testing [13]

④ Final SiC/SiC shroud system design generated in the CFCC program [43]

⑤ SiC/SiC shroud system design with a spring-loaded damper block [43]

⑥ 7FA field engine test with nine EBC-coated SiC/SiC shrouds [43]

⑦ New SiC/SiC shroud system design after partial shroud set engine test [43]

⑧ 7FA field engine test with full EBC-coated SiC/SiC shroud set [27]

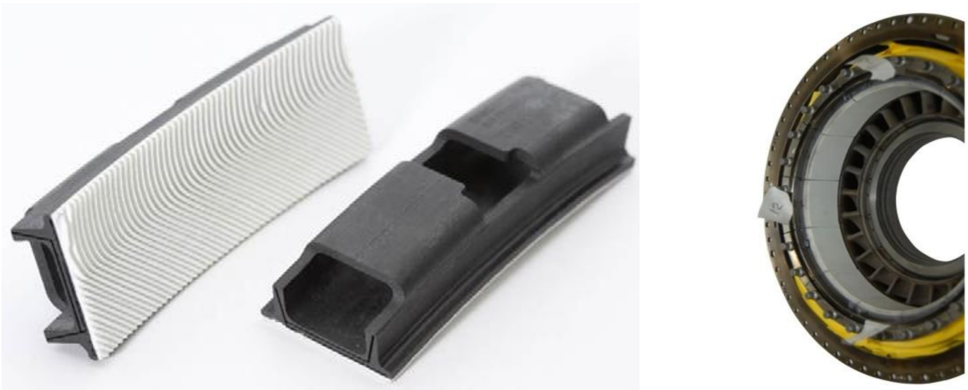

**Fig. 5.2:** EBC-coated prepreg-MI shrouds for CFM LEAP-1A engine (left) [27]
Prepreg-MI shrouds in 1$^{st}$ HPT stage of CFM LEAP engine (right) [100]

The first Japanese contributions to CMC shroud development were introduced in the 2000s. Under the ESPR project, SiC/SiC shroud prototypes were fabricated and coated with a heat- and oxidation-resistant layer [101]. The SiC/SiC shrouds were then tested in a gas burner rig to evaluate their durability.

### 5.2 CMC Turbine Vanes

Before the use of CMCs in turbines was focused, the substitution of metallic turbine components by monolithic ceramic components (without fiber reinforcement) was intensively researched in the 1980s and 1990s, especially in the field of stationary gas turbines. Relevant contributions were achieved by Solar Turbines in the US CSGT program [38, 102, 103], by MTU Aero Engines AG in Germany [104] and in a Japanese research program by Tokyo Electric Power Company, Inc. (TEPCO) [105-107]. In the course of these research activities, important design and integration concepts for ceramic vanes and blades were developed being a basis for the CMC vane development. Similar integration concepts for cooled ceramic turbine vanes are described in [104] and in [107], following the principle of split loads (**Fig. 5.3**). The thermal load is carried by an impingement cooled CMC shell in the shape of an airfoil, whereas a metallic spar within the CMC shell serves as support and jet plate. In the following, this concept is referred to as *shell & spar concept*.

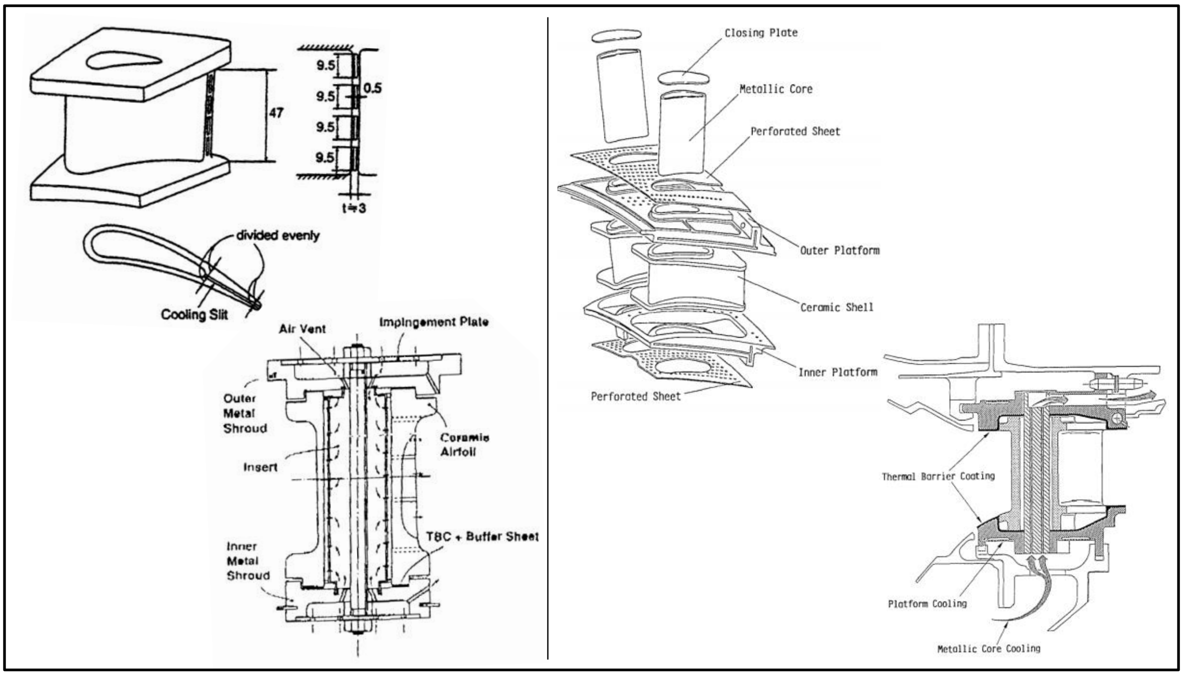


**Fig. 5.3:** Illustration of integration concepts for cooled ceramic vanes (left: [107, 108], right: [104])

The first prototypes of CMC turbine vanes were published in the 2000s [109-111]. Important contributions to the development and testing of SiC/SiC vanes were achieved in the NASA UEET program. Under the UEET program, a team from NASA GRC manufactured slurry-cast MI vanes with EBC (**Fig. 5.4**). The fabrication of these vanes is reported in [110]. NASA GRC developed a new Y-cloth fiber architecture in order to form the sharp vane trailing edge with brittle SiC fibers. The cooling holes on the pressure side were machined by laser drilling. Testing of the SiC/SiC vanes was performed under hot gas conditions using the NASA HPBR [112]. The test runs consisted of 50 h of steady state operation and 102 thermal cycles of two minutes with minimum gas temperatures between 900 and 1050 °C and maximum gas temperatures between 940 and 1440 °C. On the EBC surface a temperature of approximately 1300 °C was measured. During the tests, the cooling air entered the vanes through both ends and then flowed through the cooling holes on the pressure side into the hot gas flow. In comparison to the destroyed superalloy baseline vanes, the SiC/ SiC vanes survived the hot gas tests largely well. Burst tests were also performed later to demonstrate the design and fabrication maturity of the SiC/SiC vanes [113].

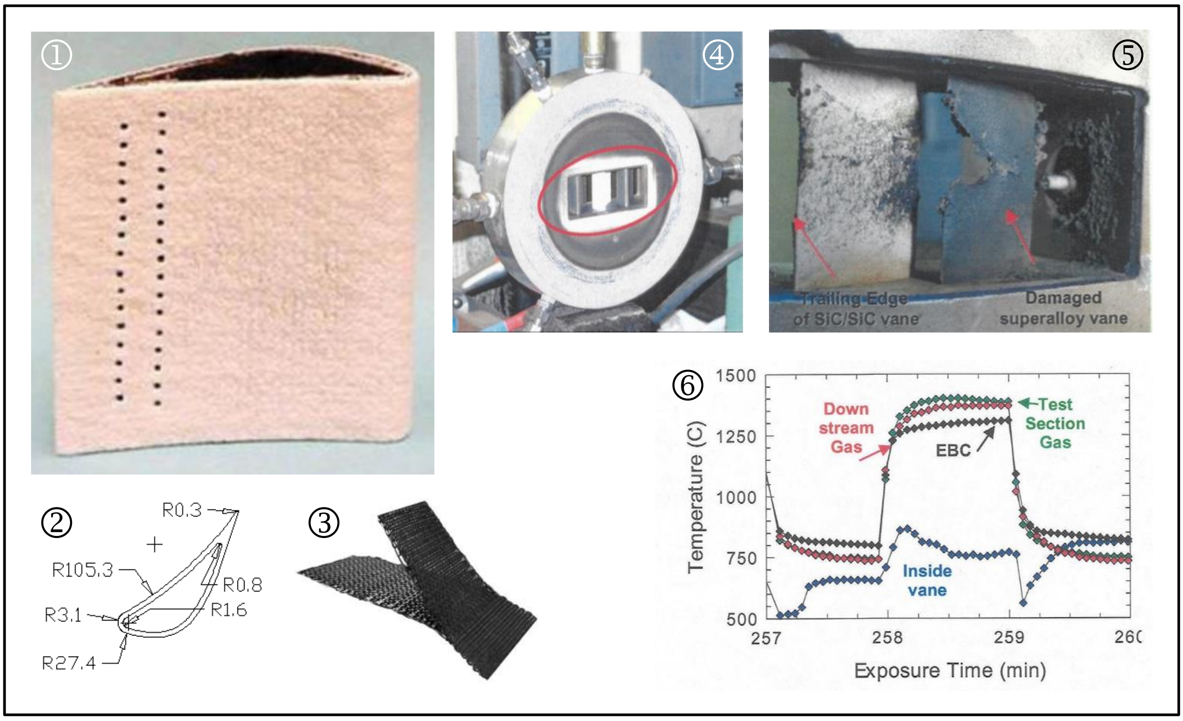


**Fig. 5.4:** NASA GRC's development and testing of slurry-cast MI turbine vanes under the UEET program

① Slurry-cast MI turbine vane with EBC [8]

② Cross-section geometry of the SiC/SiC turbine vane (dimensions in mm) [112]

③ NASA GRC's Y-cloth fiber architecture for vane trailing edges [113]

④ Cooled SiC/SiC turbine vane installed in the test section of NASA's HPBR [112]

⑤ Comparison between SiC/SiC vane and superalloy vane after cycle testing [112]

⑥ Temperatures measured during cycle 100 [112]

Another slurry-cast MI turbine vane with EBC was manufactured by UTCR as part of the UEET program (**Fig. 5.5**) [109]. P&W's FT8 engine was chosen as design baseline for the vane. Like the NASA GRC team, the UTCR team also used the Y-cloth fiber architecture to form the sharp vane trailing edge and laser drilling to machine the cooling holes. The design of the cooled CMC vane is based on the shell & spar concept, which has already been described at the beginning of this chapter. The SiC/SiC vane was tested in a burner rig with a steady state operation time of six hours at 1316 °C and 100 thermal cycles of two minutes between 482 °C and 1316 °C. No serious vane damage could be detected in the first inspection after testing.

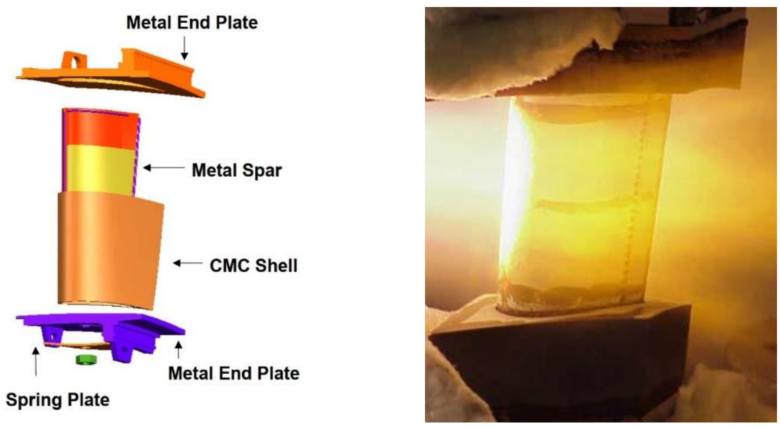


**Fig. 5.5:** Shell & spar concept of a CMC turbine vane (left) [109]
Burner rig test of a slurry-cast MI turbine vane with EBC by UTCR (right) [109]

Under the ESPR project, the first Japanese CMC vane prototype of an exhaust guide vane was developed (**Fig. 5.6**) [101, 111]. The SiC/SiC vane was manufactured using a combination of CVI and PIP processes. The fiber architecture consisted of an airfoil and platform fiber part which were fabricated separately and then stitched together. For durability evaluation of the SiC/SiC exhaust guide vane, it was tested in 1000 thermal cycles between 300 and 1150 °C.

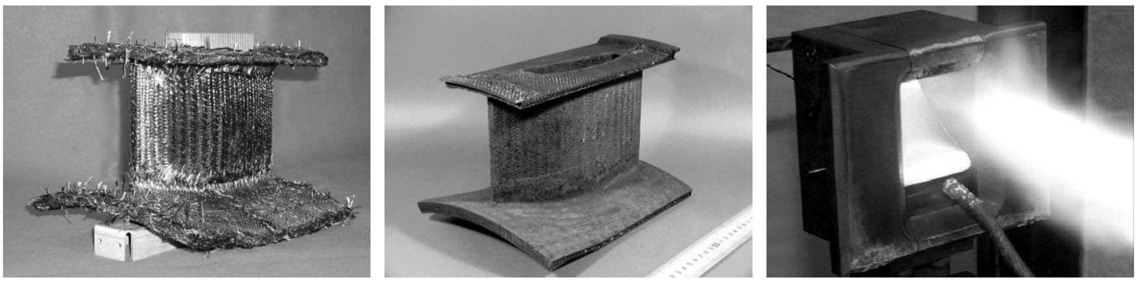

**Fig. 5.6:** Fiber architecture with stitched airfoil and platform parts (left) [101]
SiC/SiC exhaust guide vane (center) [101]
Thermal cycle testing of a SiC/SiC exhaust guide vane using a burner flame (right) [101]

Based on the findings of the NASA UEET program, a new generation of CMC turbine vane prototypes was developed by NASA GRC in the 2010s as part of the NASA ERA project (**Fig. 5.7**) [14, 18, 63, 95, 114]. Two approaches were investigated in the manufacturing of the SiC/SiC vanes: GE's prepreg-MI process and the CVI process. Moreover, the vanes were coated with EBC. The testing of the SiC/SiC vane prototypes was performed at NASA HPBR. During the test runs, the research of advanced EBCs was particularly focused. Further tests with EBC-coated SiC/SiC vane subelements fabricated via hybrid CVI/PIP process were conducted by a collaboration between NASA, P&W and UTCR in a UTCR test rig. The SiC/SiC vanes were able to demonstrate their durability at a TRL of 5 in a 1482 °C turbine environment.

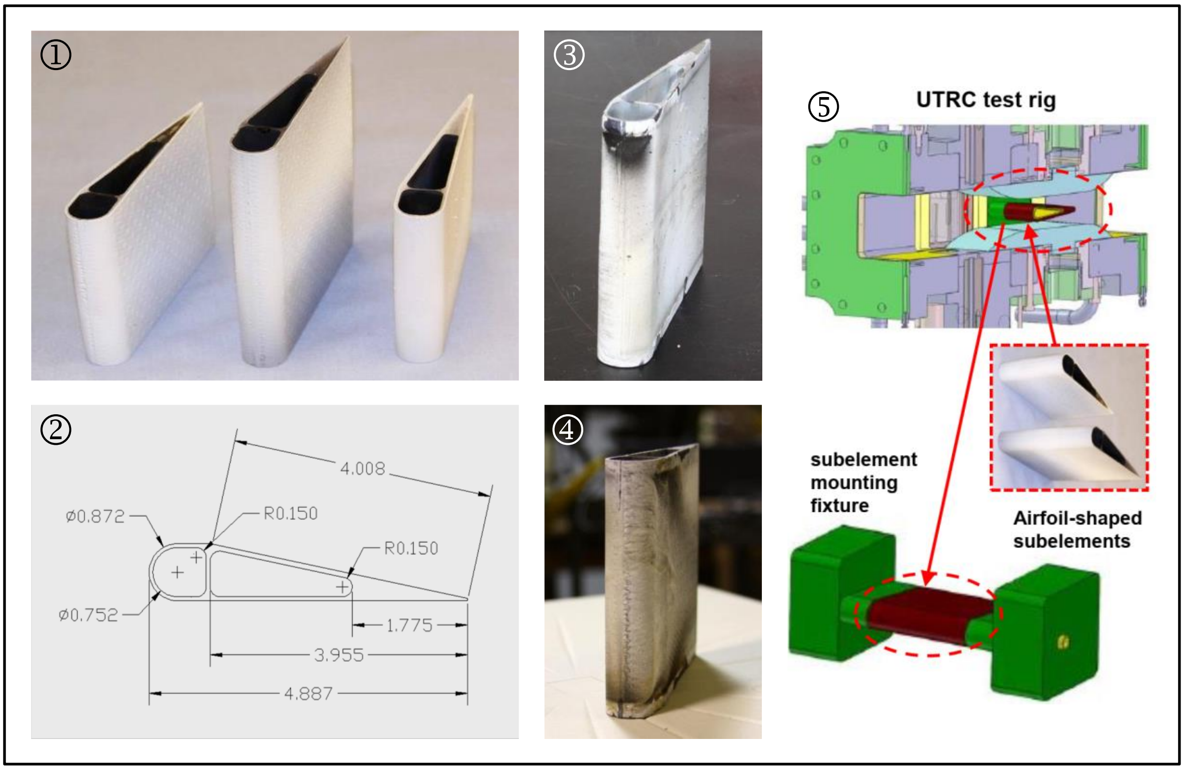


**Fig. 5.7:** Development and testing of SiC/SiC turbine vanes under the NASA ERA project
① SiC/SiC turbine vanes with advanced EBC [14]
② Cross-section geometry of the SiC/SiC turbine vane (dimensions in inches) [14]
③ Prepreg-MI vane with EBC after 21 h testing [95]
④ CVI SiC/SiC vane with EBC after 31 h testing [95]
⑤ Testing of hybrid SiC/SiC vanes with EBC in a UTRC test rig [18]

In Asia, the development of CMC turbine vanes was enhanced in the 2010s. SiC/SiC vanes with integrated end walls were engineered on behalf of the Japanese concern IHI using the IHI-IM270 engine as a baseline (**Fig. 5.8**) [79, 115]. Vane testing started with burner rig tests and thermal shock tests, followed by engine tests. First, four metallic vanes of the first stage were substituted by SiC/SiC vanes which collected 415 operating hours at a measured inlet gas temperature of approximately 1050 °C [115]. Later, six SiC/SiC vanes of the third stage were evaluated for 862 hours at a gas temperature of about 700 °C [116].

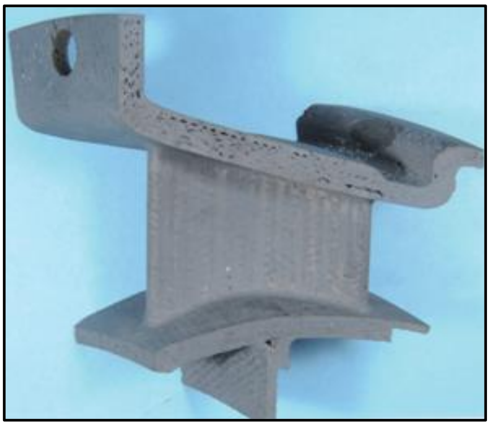

**Fig. 5.8:** SiC/SiC turbine vane for the 1st stage of the IHI-IM270 engine [79]

In China, a high temperature test facility with cooling air control for CMC turbine vanes was designed [80]. A first test run has already been completed. A CMC vane was subjected to five thermal cycles of 10 min with gas temperatures between 50 and 1200 °C. After a failure of the coating, thermal cycle testing was stopped.

Since the mid-2000s, GE has considered the use of CMCs in future engine programs [8, 27, 72]. The first experience with CMC vanes was gained in 2009 by a GE Rolls-Royce Fighter Engine Team during the development of the F136 engine for the Joint Strike Fighter. It was planned to install SiC/SiC vanes manufactured via prepreg-MI process in the third LPT stage of the engine. When the development of the F136 engine was stopped, the prepreg-MI vanes had successfully passed over 1200 hours of testing.

In the field of stationary gas turbines, GE has been developing CMC turbine vanes as part of a DoE project since the mid-2010s (**Fig. 5.9**) [37-40]. First, various CMC vane integration concepts were evaluated. Finally, a concept similar to the previously described shell & spar concept was adopted in the further development. The first prototypes of CMC shells with integrated end walls have already been fabricated. Moreover, a CMC vane cooling system consisting of impingement cooling and trailing edge cooling is intended. Regarding trailing edge cooling, two approaches were considered: small diameter cooling holes made by electric discharge machining and meandering internal cooling passages. Engine tests of the CMC vanes are scheduled for the future. In addition, EBCs were investigated in durability tests.

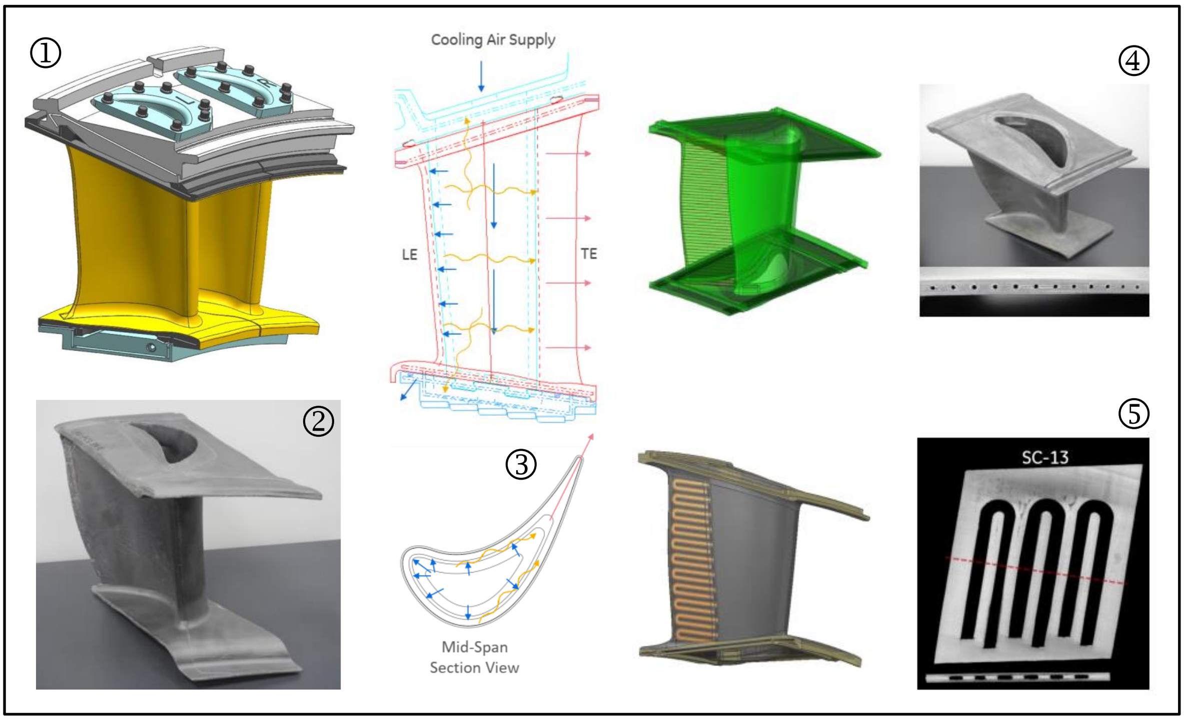


**Fig. 5.9:** GE's CMC turbine vane development for stationary gas turbines as part of a DoE project
① CMC vane design similar to the shell & spar concept [68]
② Uncoated CMC shell with integrated end walls [68]
③ CMC vane cooling system with impingement and trailing edge cooling [66, 68]
④ Trailing edge cooling with small diameter cooling holes [68]
⑤ Trailing edge cooling with meandering internal cooling passages [68]

The first series application of CMC vanes is planned by GE in the GE9X engine [15, 27]. In the GE9X, SiC/SiC vanes are used in the first and second HPT stage.

Current research activities concerning CMC vanes can also be observed at UTRC [117]. Under a project initiated by the DoE, impingement cooled CMC vanes are being researched which are designed according to the shell & spar concept and feature a monolithic ceramic layer between the CMC shell and the TEBC (**Fig. 5.10**).

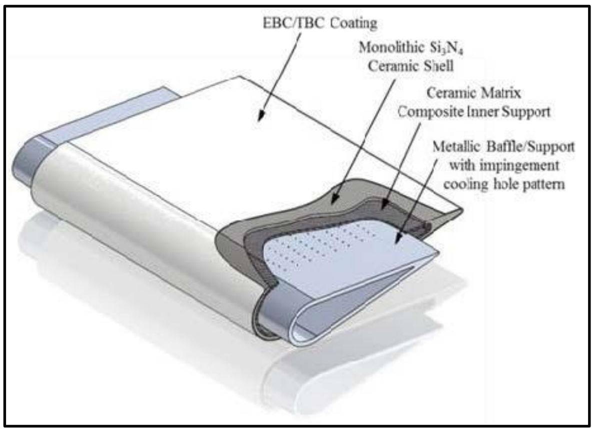


**Fig. 5.10:** Impingement cooled CMC shell with metallic spar, monolithic ceramic layer and TEBC [117]

### 5.3 CMC Turbine Blades

The first uncooled CMC blade prototypes for low pressure turbines were developed and tested by a cooperation between GE and the US Army Aviation Applied Technology Directorate (AATD) in the mid-2000s [118]. GE's Advanced Affordable Turbine Engine (AATE), a turboshaft engine for military helicopters, was chosen as a baseline engine. The CMC LPT blades were manufactured based on a technology developed by GE (**Fig. 5.11**). First, a uniaxial tape lay-up fiber architecture in the shape of an airfoil was created which is then used in the prepreg-MI process. By means of this method, GE was able to fabricate twisted uncooled SiC/SiC blades with small thickness, thin trailing edges and dovetails. More than 200 CMC LPT blades were produced in order to demonstrate the feasibility and applicability of GE's manufacturing process. After the successful fabrication of the prepreg-MI LPT blades, they were tested in several test rigs. Dovetail and witness panel pull testing, overspeed and high cycle fatigue (HCF) testing, tip rub testing and FOD testing were performed to achieve a TRL of five.

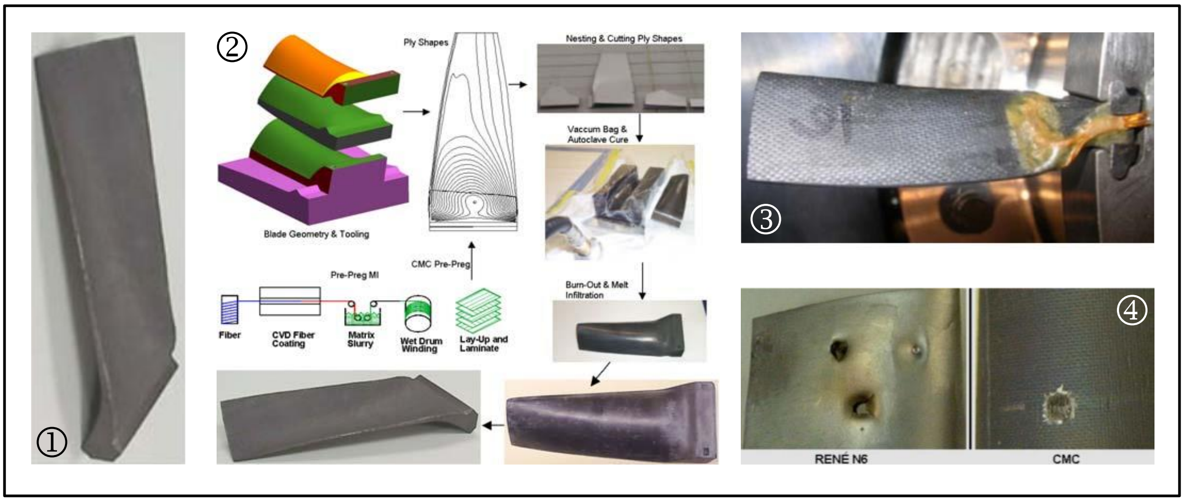


**Fig. 5.11:** Development and testing of uncooled prepreg-MI LPT blade prototypes by a cooperation between GE and AATD [118]

① Prepreg-MI LPT blade prototype

② Manufacturing process of prepreg-MI blades developed by GE

③ SiC/SiC blade installed in the HCF test rig

④ Comparison between SiC/SiC blade and superalloy blade after FOD testing

The first successful tests of rotating uncooled CMC LPT blades in a military engine were performed by GE between 2010 and 2015 [27, 69, 71, 119]. The SiC/SiC blades were fabricated via prepreg-MI process and tested in the second LPT stage of a F414 turbofan demonstrator engine (**Fig. 5.12**). During testing, it was possible to validate the applicability of CMC components under rotating conditions. CMC blades offer great potential for the design of lighter turbine disks, shafts and bearings because of lower centrifugal forces than conventional superalloy blades.

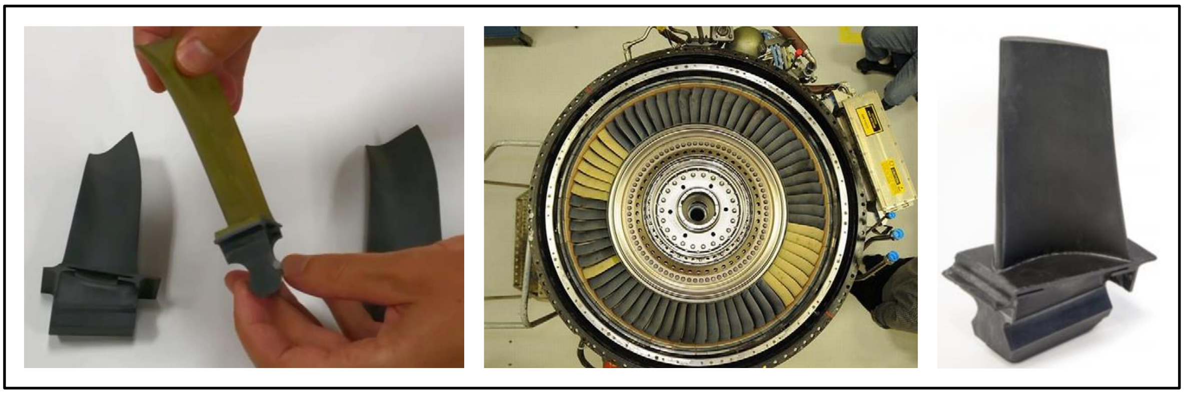

**Fig. 5.12:** Uncooled prepreg-MI LPT blades for F414 engine manufactured by GE (left) [27]

Full LPT stage of SiC/SiC blades before engine testing (yellowish blades are EBC-coated, gray blades are uncoated) (center) [27]

CMC turbine blade presented by GE at the Paris Air Show 2015 (right) [71]

Like GE, Safran also focused on testing uncooled CMC LPT blades for civil engines in the early 2010s [8, 26]. SiC/SiC blades were manufactured and then tested in a CFM56 engine in order to demonstrate the viability of CMC blades (**Fig. 5.13**).

**Fig. 5.13:** Uncooled Cerasep® A40C LPT blades for CFM56 engine [8]

Furthermore, Japanese research activities regarding CMC blades can be observed in the 2010s [79]. On behalf of IHI, SiC/SiC blades were fabricated and evaluated in the course of several tests under laboratory conditions.

In addition to the development of CMC blades, the feasibility of CMC bladed disks (blisks) has also been investigated in recent decades. In the NASA SIMPLEX Turbopump Blisk program, C/SiC blisk prototypes for turbopumps were manufactured being used in rocketry (**Fig. 5.14**) [120, 121] Moreover, two CMC blisks were tested in a rig under relevant environmental conditions.

Another CMC blisk prototype was developed and tested in Japan as part of the AMG program [55].

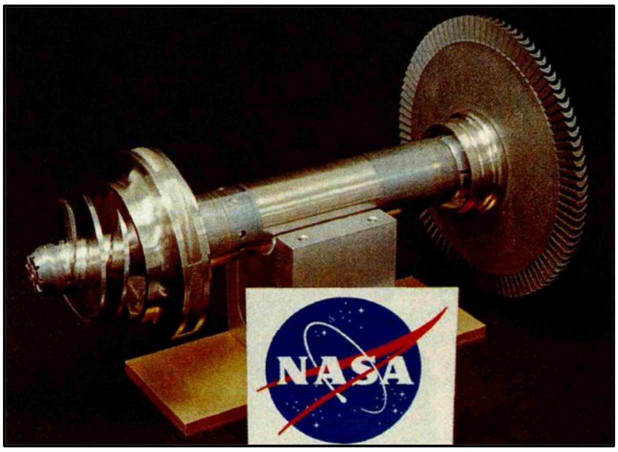


**Fig. 5.14:** NASA's Simplex Turbopump C/SiC blisk installed on a shaft [120]

## 6 CMC Nozzle Components

Since military and civil nozzles differ in design and thermomechanical requirements, the technological level of CMC nozzle components is presented below classified into military nozzle applications (ch. 6.1) and civil nozzle applications (ch. 6.2).

### 6.1 Military CMC Nozzle Applications

In the 1980s, the evaluation of CMC parts for military nozzles was focused by SEP and Snecma (later SPS, now Safran Ceramics) [8, 29]. The first CMC nozzle flaps were demonstrated in the M53-2 engine of a Mirage 2000 at the Paris Air Show in 1989.

After successful testing in the M53-2 engine, Snecma continued the development of outer and inner nozzle flaps in the Rafale/M88-2 test program. In 1996, C/SiC outer nozzle flaps were ready for series production and used as standard in the M88-2 engine of the Dassault Rafale (**Fig. 6.1**) [3, 8]. The C/SiC outer nozzle flaps were fabricated by using CVI process.

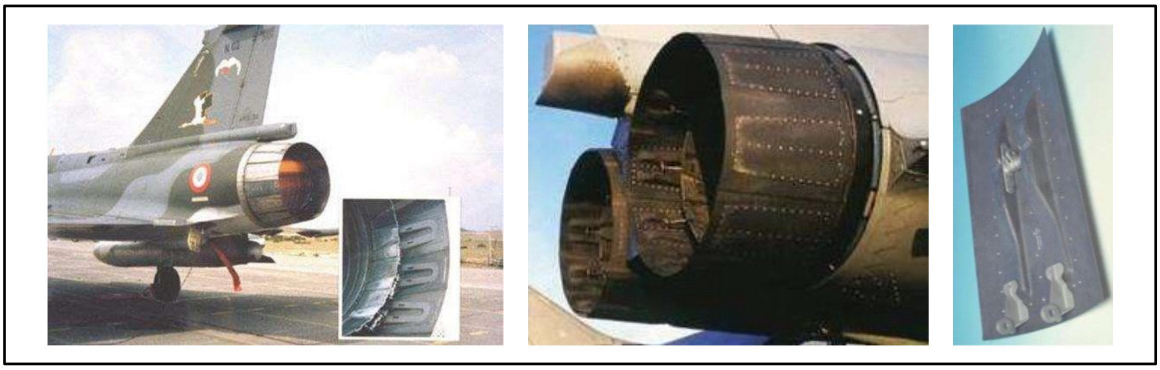

**Fig. 6.1:** Outer an inner nozzle flaps of the M53-2 engine (left) [8]
Sepcarbinox® A262 outer nozzle flaps of the M88-2 engine (center) [8]
Sepcarbinox® A262 outer nozzle flap (right) [3]

Since the SiC/SiC inner flaps inspected in the Rafale/M88-2 test program did not withstand the high thermomechanical requirements of the M88-2 engine [8, 29], SPS developed a new SiC matrix generation based on the self-sealing matrix concept [24, 30]. In the 2000s, this material was tested in military nozzles in cooperation with the US military, first in ground engine tests and later also in flight tests [8, 26, 50, 51]. Mainly SiC/SiC divergent seals were manufactured via CVI, which were tested in the F100-PW-229 engine of the F-15 Eagle and F-16 Falcon (**Fig. 6.2**). During the flight tests, the SiC/SiC divergent seals accumulated more than 1000 h, thus demonstrating the potential of CMCs for military nozzle applications.

**Fig. 6.2:** Sepcarbinox® A500 divergent seal for F100-PW-229 engine (left) [8]
Nozzle of a F-16 Falcon with five SiC/SiC divergent seals (center) [50]
SiC/SiC divergent seals installed in the right nozzle of a F-15 Eagle (right) [50]

Other nozzle applications used for testing the CMCs developed by SPS are shown in (**Fig. 6.3**) [8]. A CMC mixer nozzle was manufactured for military helicopters. Besides, CMC flame holders for the afterburner of the M88-2 engine were fabricated and tested. In thermal cycle tests, the CMC flame holders successfully completed 2000 hours at a maximum temperature of 1180 °C.

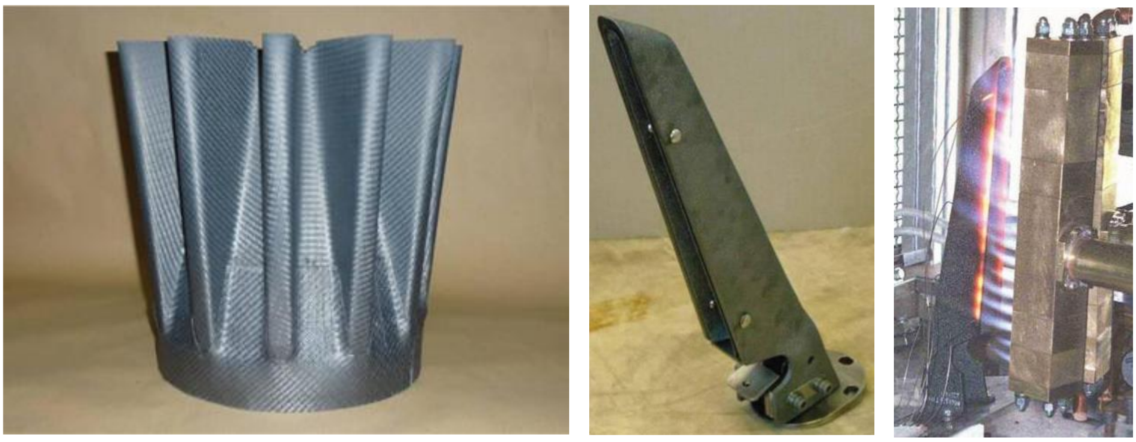

**Fig. 6.3:** CMC mixer nozzle for military helicopters (left) [8]
CMC flame holder for the M88-2 engine (center) [8]
Thermal cycle testing of a CMC flame holder(right) [8]

Independent of the research activities with SPS, the US military has been evaluating CMC divergent flaps and seals for military nozzles since the 1990s [8, 48, 49]. Important tests were performed in GE's F110 engine.
Later, the US military focused on the qualification of CMC divergent flaps and seals made of SiC fiber-reinforced carbon (SiC/C) for GE's F414 engine of the F/A18-E/F Super Hornet (**Fig. 6.4**) [5, 8, 52]. Furthermore, GE developed Ox/Ox divergent seals which were ready for production in 2011 and can alternatively be installed in the nozzle of the F414 engine.

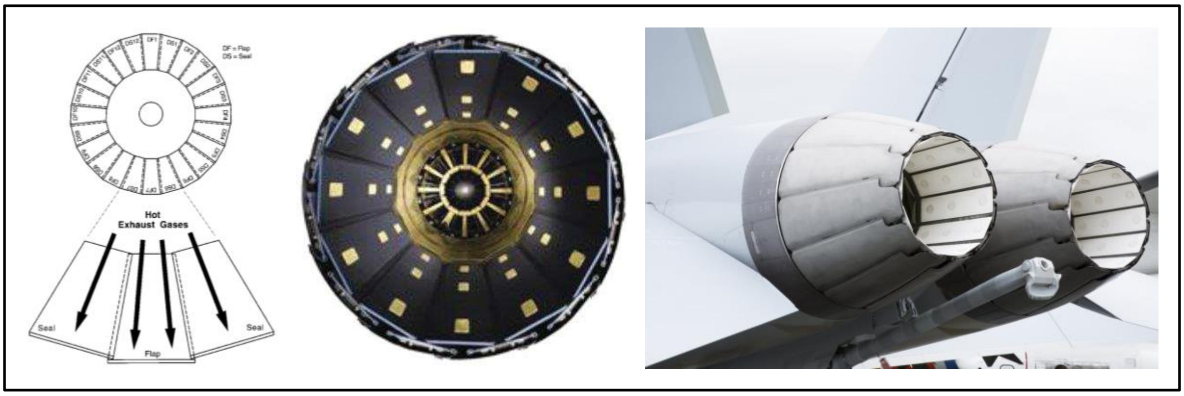


**Fig. 6.4:** Schematic of an open military nozzle consisting of divergent seals and flaps (left) [48]
Closed nozzle of a F414 engine with SiC/C divergent flaps and seals (center) [8]
Closed nozzles of two F414 engine with Ox/Ox divergent seals (right) [122]

## 6.2 Civil CMC Nozzle Applications

The first CMC nozzle components for civil engines were developed by Safran in the mid-2000s [8, 26, 32, 73]. Safran manufactured a CMC exhaust system for the CFM56 engine which consists of a SiC/SiC mixer and a SiC/SiC centerbody and offers a significant weight reduction compared to the metallic counterpart (**Fig. 6.5**). In 2007, the CMC exhaust system was evaluated in ground engine tests. 700 engine cycles including 70 take-off hours were performed without material damage. The first flight test of the CMC exhaust system was started on an Airbus A320 in 2012. The SiC/SiC mixer and the SiC/SiC centerbody achieved 18 flight hours. In 2015, Safran received EASA certification for a two-year in-service evaluation phase of the SiC/SiC exhaust system on a commercial aircraft.


**Fig. 6.5:** Safran's SiC/SiC exhaust system for the CFM56 engine consisting of a Cerasep® A40C mixer and a Cerasep® A40C centerbody (left) [9]
Ground engine testing of the SiC/SiC exhaust system (center) [8]
Flight testing of the SiC/SiC exhaust system on an Airbus A320 (right) [8]

In the 2010s, the use of Ox/Ox in civil nozzles was focused [5, 8, 123]. A first Ox/Ox exhaust nozzle prototype for the Rolls Royce Trent 1000 engine was developed by a cooperation between Boeing, ATK-COIC and Albany Engineered Composites (AEC) in the CLEEN program of the FAA (**Fig. 6.6**). Compared to titanium components, the CMC exhaust nozzle offers a weight saving of 15%. According to [123], the centerbody is the longest Ox/Ox part and the nozzle outer ring is the largest diameter Ox/Ox part ever made. The testing of the CMC exhaust nozzle began in 2012 with ground engine tests on a Rolls Royce Trent 1000. In 2013, the first flight test on a Boeing 787 was planned.

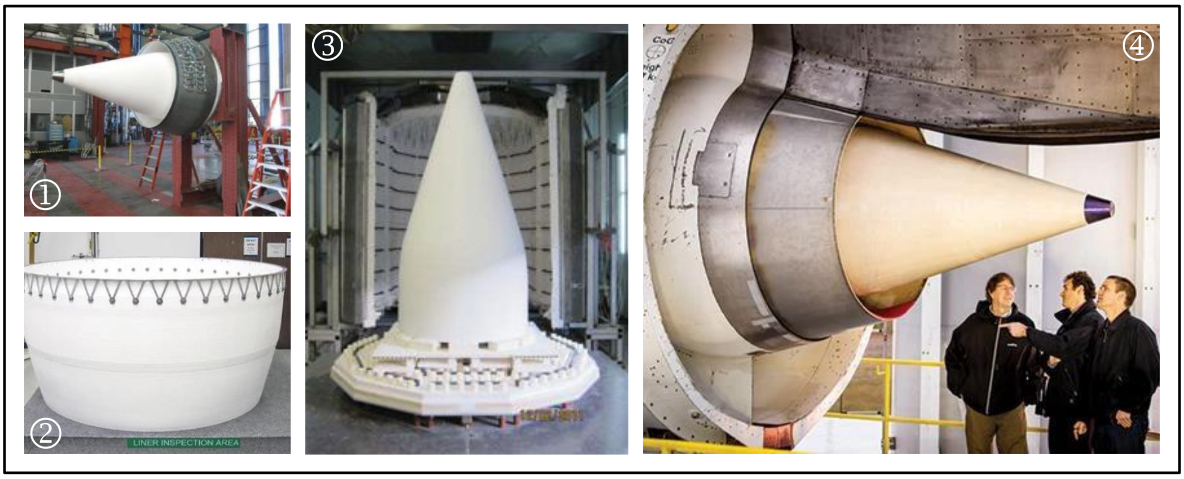


**Fig. 6.6:** Development and testing of an Ox/Ox exhaust nozzle prototype for the Rolls Royce Trent 1000 engine under the CLEEN program of the FAA
① Ox/Ox exhaust nozzle assembly [5]
② Ox/Ox outer ring of the CMC exhaust nozzle [123]
③ Ox/Ox centerbody of the CMC exhaust nozzle [5]
④ Ground engine testing of the Ox/Ox exhaust nozzle on a Rolls Royce Trent 1000 [123]

In 2010, a cooperation between NASA GRC, Rolls-Royce LibertyWorks® (RRLW) and ATK-COIC started the development of an Ox/Ox exhaust mixer for subsonic jet engines within the NASA ERA project (**Fig. 6.7**) [5, 14, 124]. The aim was to achieve a TRL of 6 in 2015. A subscale Ox/Ox exhaust mixer with 16 lobes and a diameter of approximately 200 mm was manufactured and then tested in several rigs. Aerodynamic performance testing was performed by ASE FluiDyne at a Hot/Cold Dual Stream Static Nozzle Test Facility. In the course of the performance tests, various combinations of temperature and pressure ratios were operated and an excellent mixing efficiency was measured. The acoustics of the subscale Ox/Ox exhaust mixer were evaluated in the AeroAcoustic Propulsion Laboratory (AAPL) of NASA GRC and a potential for noise reduction could be detected. After the successful testing of the subscale Ox/Ox exhaust mixer, two full-scale Ox/Ox mixers were fabricated being sized for the Rolls-Royce AE3007 engine. These were then evaluated in vibration tests at the Structural Dynamics Laboratory of NASA GRC. In 2014, the second mixer achieved over 2 million cycles of vibration testing with only three minor defects. As a result, ground engine tests were planned at Rolls-Royce in Indianapolis.

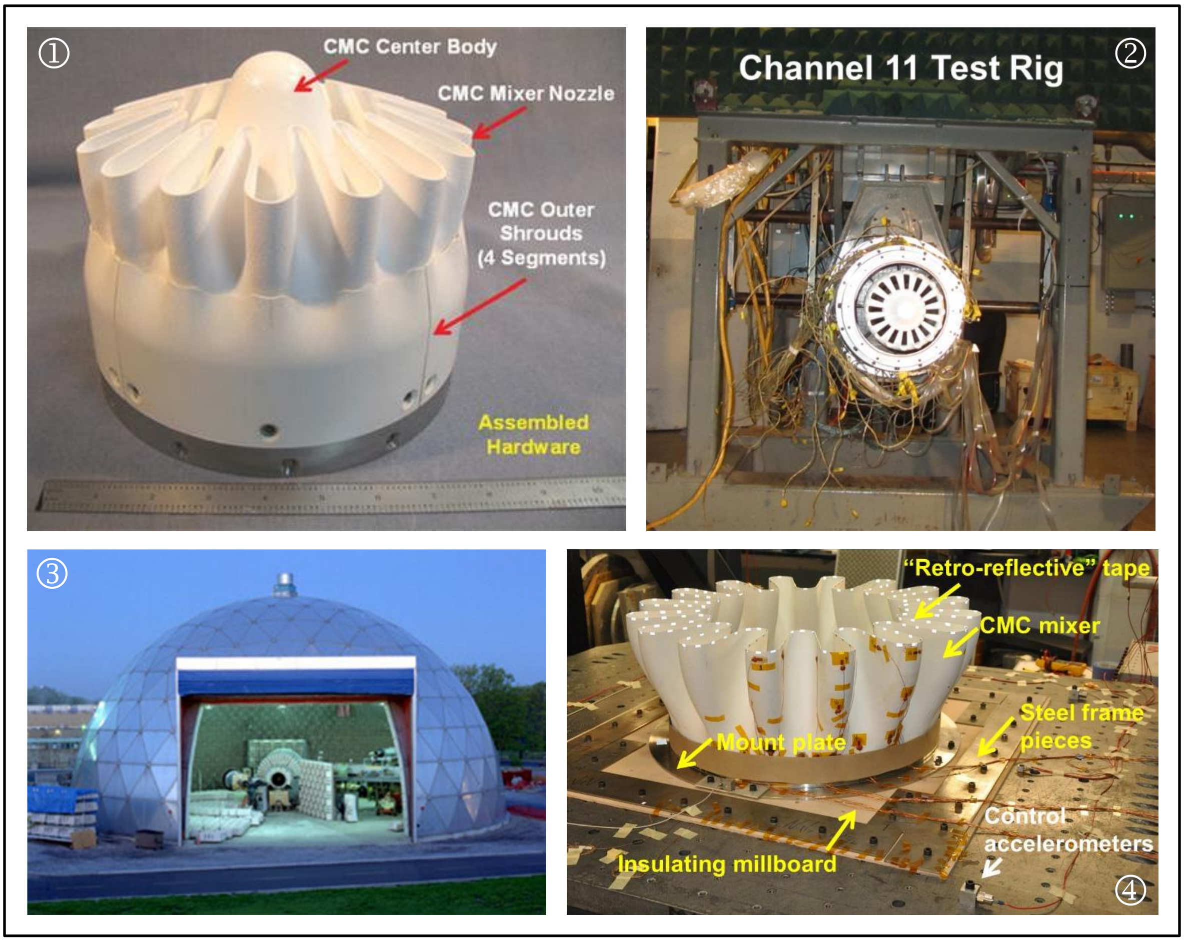


**Fig. 6.7:** Development and testing of an Ox/Ox exhaust mixer under the NASA ERA program
① Subscale Ox/Ox exhaust mixer assembly [5]
② Aerodynamic performance testing of the subscale Ox/Ox exhaust mixer [124]
③ Acoustic testing of the subscale Ox/Ox exhaust mixer [124]
④ Vibration testing of the 2nd full-scale Ox/Ox mixer in 2014 [124]

After GE had gained a lot of experience with Ox/Ox by the development of the divergent seals for the F414 engine, GE's Ox/Ox material made its commercial debut in the Passport 20 engine [5, 52, 81, 125]. The engine powered the Bombardier Global 7500 and 8000 and entered into service in 2018. An Ox/Ox exhaust mixer and a five-part Ox/Ox core cowling are installed in the Passport 20 engine which are manufactured by Composites Horizons LLC in Covina (California) and enable a weight saving of approximately 20 kg compared to the metallic baseline. Thus, GE is the first company using an Ox/Ox exhaust mixer as standard in a commercial engine. The exhaust mixer consists of an Ox/Ox mixer and an Ox/Ox centerbody which are fabricated separately and assembled later (**Fig. 6.8**). The Ox/Ox mixer assembly is 610 mm long and 965 mm in diameter.

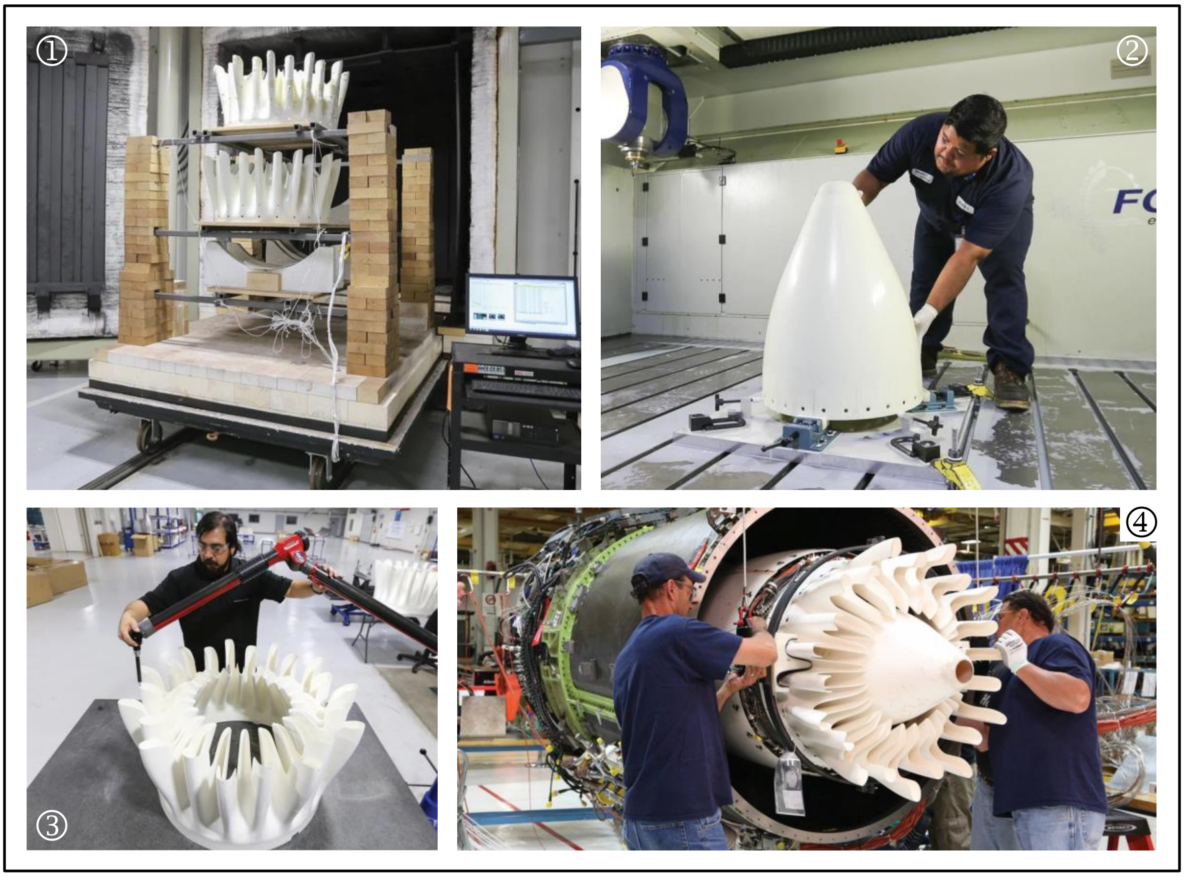


**Fig. 6.8:** Manufacture and installation of an Ox/Ox exhaust mixer for GE's Passport 20 engine

① Ox/Ox mixers in front of the sintering furnace [81]

② Machining of an Ox/Ox centerbody on a CNC milling machine [81]

③ Quality control of an Ox/Ox mixer [81]

④ Installation of an Ox/Ox exhaust mixer in a Passport 20 engine [52]

# 7 Summary

This report summarizes the technological level of CMC components for stationary gas turbines and aero-engines. After a historical overview of CMC research over the past decades, the development of various CMC components is described in detail, focusing on CMC components for combustor, turbine and nozzle applications. The introduction of GE's CMC shrouds in the LEAP engine of CFM and the debut of a CMC exhaust mixer for GE's Passport 20 engine are important milestones in the field of civil engines. CMC nozzle flaps and seals have been intensively researched for military engines, first series applications have already been achieved. The entry into service of the GE9X with several CMC components is scheduled for this decade, the use of CMC liners, shrouds and stators in the hot gas section have been announced. An important design concept for cooled CMC stators is the shell & spar concept, following the principle of split loads. The thermal load is carried by an impingement cooled CMC shell, whereas a metallic spar within the shell serves as support and jet plate. In terms of rotating CMC components, research is being conducted on uncooled CMC rotors for low pressure turbines. First prototypes have already been developed and successfully tested.

## Nomenclature

| | |
|---|---|
| AAPL | AeroAcoustic Propulsion Laboratory of NASA GRC |
| AATD | US Army Aviation Applied Technology Directorate |
| AATE | Advanced Affordable Turbine Engine |
| ADP | Advanced Technology Program |
| AEC | Albany Engineered Composites |
| AMAIGT | Advanced Materials for Advanced Industrial Gas Turbines |
| AMG | Advanced Material Gas-Generator |
| ASCR | Advanced Subsonic Combustion Rig |
| ATK-COIC | ATK-COI Ceramics |
| Blisk | Bladed Disk |
| C/C | Carbon-Fiber-Reinforced Carbon |
| C/SiC | Carbon-Fiber-Reinforced Silicon Carbide |
| CFCC | Continuous Fiber Ceramic Composite |
| CFM | CFM International (joint venture 50/50 joint venture of GE and Safran) |
| CLEEN | Continuous Lower Energy, Emissions and Noise |
| CMC | Ceramic Matrix Composite |
| CNC | Computerized Numerical Control |
| CO | Carbon Monoxide |
| $CO_2$ | Carbon Dioxide |
| CSGT | Ceramic Stationary Gas Turbine |
| CVI | Chemical Vapor Infiltration |
| DLR | German Aerospace Center |
| DoE | US Department of Energy |
| EASA | European Union Aviation Safety Agency |
| EBC | Environmental Barrier Coating |
| EPM | Enabling Propulsion Materials |
| ERA | Environmentally Responsible Aviation |
| ESPR | Research and Development of Environmentally Compatible Propulsion System for Next-Generation Supersonic Transport |
| FAA | Federal Aviation Administration |
| FAP | Fundamental Aeronautics Program |
| FOD | Foreign Object Damage |
| FGI | Friable Graded Insulation |
| GE | General Electric |
| GMC | Glass Matrix Composite |
| GRC | Glenn Research Center |
| HCF | High Cycle Fatigue |
| HiPOC | High Performance Oxide Ceramic |

| | |
|---|---|
| HPBR | NASA High Pressure Burner Rig |
| HPT | High Pressure Turbine |
| HSCT | High Speed Civil Transport |
| HSR | High Speed Research |
| HYPR | Super/Hyper-sonic Transport Propulsion System |
| IHI | Ishikawajima-Harima Heavy Industries Corporation |
| IHPTET | Integrated High Performance Turbine Engine Technology |
| LOCCA | Low NOx Oxide Ceramic Combustors for Aero-Engines |
| LPT | Low Pressure Turbine |
| LSI | Liquid Silicon Infiltration |
| NASA | National Aeronautics and Space Administration |
| NIST | National Institute of Standards and Technology |
| NOx | Nitrogen Oxide |
| Ox/Ox | Oxide-Fiber-Reinforced Oxide Ceramic |
| P&W | Pratt & Whitney |
| PIP | Polymer Infiltration and Pyrolysis |
| RMI | Reactive Melt Infiltration |
| RQL | Rich-Burn, Quick-Quench, Lean-Burn |
| RRLW | Rolls-Royce LibertyWorks® |
| SEP | Société Européenne de Propulsion |
| SiC/C | Carbon-Fiber-Reinforced Silicon Carbide |
| SiC/SiC | Silicon Carbide-Fiber-Reinforced Silicon Carbide |
| SPS | Snecma Propulsion Solide |
| TBC | Thermal Barrier Coating |
| TEBC | Thermal and Environmental Barrier Coating |
| TEPCO | Tokyo Electric Power Company, Inc. |
| TRL | Technology Readiness Level |
| TTT | Transformational Tools and Technologies |
| UEET | Ultra-Efficient Engine Technology |
| UTRC | United Technologies Research Center |
| VAATE | Versatile Affordable Advanced Turbine Engines |

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